Technical Memorandum 4170

OFFICE OF EXPLORATION

Exploration Studies Technical Report

FY 1989 STATUS

Volume VI: Special Reports, Studies, and Indepth Systems Assessments

August 1989

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NASA

National Aeronautics and Space Administration

Lyndon B. Johnson Space Center
Houston, Texas

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This publication is one of seven documents describing work performed in fiscal year 1989 under the auspices of the Office of Exploration. The first in the series, titled, "Journey Into Tomorrow...Beyond Earth's Boundaries II" provides an overall programmatic view of the goals, opportunities, and challenges of achieving a national goal for human exploration. The technical details and analyses are described in a six-volume set titled: "Office of Exploration: Exploration Studies Technical Report (FY 1989 Status)." Volume I is Mission and Integrated Systems; Volume II is Space Transportation Systems; Volume III is Planetary Surface Systems; Volume IV is Nodes and Space Station Freedom Accommodations; Volume V is Technology Assessment; and Volume VI is Special Reports, Studies, and In-Depth Systems Assessment. These six volumes document the status of Exploration Technical Studies at the conclusion of the FY 1989 study process in August 1989, and, therefore, do not contain any analyses, data, or results from the NASA 90-Day Study on Human Exploration of the Moon and Mars.

NASA Technical Memorandum 4170

Office of Exploration
Exploration Studies Technical Report
FY 1989 Status

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Space Administration

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Disclaimer Statement

The Exploration Studies Process, as explained in detail in Section 2 of Volume I, was a requirements driven, iterative, and dynamic process developed for case study analysis. This process consisted of three parts: (1) requirements generation, (2) implementation development, and (3) integrated case study synthesis.

During the final step of the process, an integrated mission was developed for each of the case studies by synthesizing the implementations developed earlier into a coherent and consistent reference mission. These are presented in Section 3 of Volume I of this annual report. Given the iterative and dynamic nature of this process, there are two important items to note:

- **The integrated case studies do not always reflect a mission that has a direct one-to-one correspondence to the requirements specified in the March 3, 1989, *Study Requirements Document*. Many changes were made to these requirements prior to and during the synthesis activities when warranted.**

- **The integrated case studies presented in Volume I represent the results of the synthesis process. Volumes II, III, and IV are the implementation databases from which the integrated case studies were derived. Therefore, the implementations outlined in Volumes II, III, and IV are generally reflected in the integrated case studies, but, in some cases, the implementations were changed in order to be effectively included in the integrated case studies. These modifications are only briefly discussed in Volumes II, III, and IV.**

VOLUME VI: SPECIAL REPORTS, STUDIES, AND INDEPTH SYSTEMS ASSESSMENTS

CONTENTS

VOLUME VI: SPECIAL REPORTS, STUDIES, AND INDEPTH SYSTEMS ASSESSMENTS
CONTENTS (CONT.)

VOLUME VI: SPECIAL REPORTS, STUDIES, AND INDEPTH SYSTEMS ASSESSMENTS

CONTENTS (CONC.)

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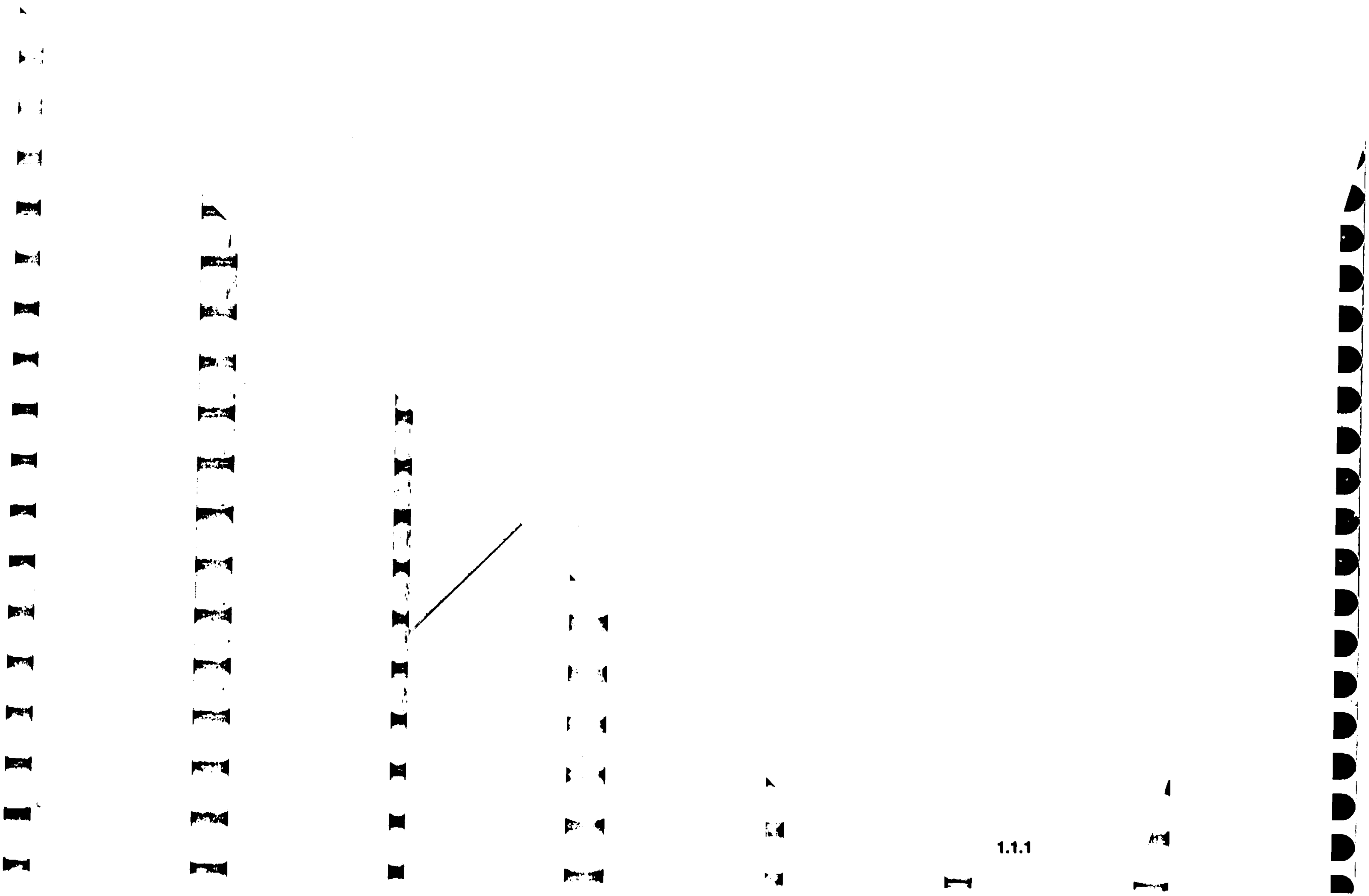
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Lockheed
Engineering & Sciences Company

Contract NAS 9-17900
Job Order YP-202

PROCEEDINGS OF THE GOVERNMENT MEETING

ON THE PLANETARY ENVIRONMENT ANALOGUE

FACILITY

April 18-19, 1989
Houston, TX

edited by
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December, 1989

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Section	CONTENTS	Page

SECTION 1

Overview

On April 18-19, 1989, a meeting was held at the Lunar and Planetary Institute in Houston to determine the government's requirements for a Planetary Environment Analogue Facility (PEAF). It was considered of particular importance by the meeting sponsor (Planetary Surface Systems Integration Agent) and the steering committee to establish needs for such a facility from government entities outside of NASA as well as from within the agency.

Representatives from several NASA field centers and headquarters program offices as well as the Bureau of Mines and the US Army Corps of Engineers were present. This report synthesizes the discussion, conclusions and recommendations reached at that meeting. The Appendices provide a list of attendees and a summary of proposed concepts for PEAFs.

SECTION 2

Planetary Environment Analogue Facility Concept

A Planetary Environment Analogue Facility (PEAF) is envisioned as a physical installation on Earth which could provide simulations of various characteristics of the environment on the Moon or Mars. It could be a testbed dedicated to developing, verifying, validating, and/or integrating technologies for large scale planetary operations. Environmental factors which could be simulated include, among many others, the rock and dust of a planetary surface, illumination, temperature extremes, vapor, vacuum or low pressures, and ionizing radiation.

A wide variety of operations and equipment could be researched, tested, and evaluated in such an analogue facility. These could include delicate electronic instrumentation, large-scale mining or construction equipment, closed ecological life support systems (CELSS), man-machine interactions in a hostile environment, tele-operated robotic machines, and simulations of crew operations in stressful, isolated environments.

Five separate lunar base analogue facilities have been proposed from other sectors. They are Moon Park, Lunar Base One, the Antarctic Planetary Testbed, Center for Extraterrestrial Engineering and Construction (CETEC), and the Lunar Base Simulator. Each has a different proponent, proposed location, sponsor, and targeted technology. Appendix B provides more information about these proposals.

SECTION 3

Requirements for a PEAF

3.1 NASA NEEDS FOR A PEAF

Support for a PEAF exists for the development of lunar/Mars hardware and operations. Support also can be found for intensive analysis of terrestrial operations in construction and mining.

3.1.1 Science

The Office of Space Science and Applications (OSSA) was indirectly represented at the meeting by Kyle O. Fairchild, the convenor. Fairchild, in a presentation based on comments from John Rummel (OSSA, NASA Headquarters) and Andy Cameron (Science Applications International Corporation), stated that OSSA has expressed a strong interest in the concept of "terrestrial analogues." He added that OSSA has also stated that "analogues that cannot clearly translate themselves into real scientific goals will run without Code E support or interest."

In Fairchild's presentation, specific OSSA needs which could be supported by an analogue were identified. One need is to discover whether factors such as instrument size, power requirements or crew involvement have different impacts on short-term and long-term missions. A second need is a demonstration of the simulated lunar surface as a location for instrument and technology development for Mars/Phobos sample return missions. Both of these requirements could be fulfilled by the large-scale simulation capabilities of a PEAF.

A third need is the establishment of a closer relationship between OSSA and the Office of Exploration (OEXP) in early definition of long range science programs and associated technologies (including man-machine issues). The PEAF, because of its complexity and need for integration, could contribute to a closer relationship.

3.1.2. Exploration

The Office of Exploration (OEXP) was represented at the meeting by Woody Lovelace. He explained that OEXP is tasked to provide recommendations and alternatives for an early 1990s national decision on a focused program for human exploration of the solar system. It must also steer present agency investments to arrive at well-defined choices in the early 1990s.

According to Lovelace, OEXP has five explicit requirements which a PEAF might help support:

- a testbed for development and demonstration of systems and technologies
- a means to gain experience in lunar surface materials processing
- a means to realistically simulate missions prior to their occurrence
- a way to perform real-time anomaly investigations during missions
- a facility for developing crew operations and procedures

A PEAF could support all of these requirements through its realistic, long-term simulation of planetary surfaces. It could provide an environment for the development of systems and technologies, practice in materials processing, simulation of missions in training and in realtime, and development of crew operations and procedures for use on a planetary surface.

3.1.3 Transportation

Bruce Wiegmann of Marshall Space Flight Center discussed requirements for a PEAF from a space transportation viewpoint. Wiegmann cited three specific needs a PEAF could fulfill in developing space transportation vehicles and ground servicing facilities. One need is as a testbed for developing systems and subsystems. A second need is testing operational procedures to service transportation systems. According to Wiegmann, a facility for checking out operational mission procedures is *mandatory* for safety. And third, a need exists to increase integration and cooperation between different centers and directorates within NASA. A PEAF could meet this requirement.

3.1.4 Human Factors/Operations/Training

To support human factors/operations/training, Yvonne Clearwater of Ames Research Center (ARC) discussed the needs for a PEAF. First, a high fidelity, highly rigorous simulation of isolated, hazardous environments over varying periods of time is required for the study of unpredictable personality dysfunctions which occur after 30 days in any isolated, stressful environment.

Another need is to orient the crew to the hazards and stresses involved in long-term isolation. A third requirement is to train crew under highly realistic conditions, for example, communications breakdowns, lockouts, lack of medical intervention, true isolation from any aid, and difficult EVA activities.

A variety of studies are required which a PEAF could support. These studies could include habitability factors, crew interaction under realistic conditions, environmental control, food management, power, and waste and water management. Crew operations, both human and human-machine, also need to be tested.

3.1.5 Life Support

From the perspective of life support, John Sager of Kennedy Space Center discussed the needs for a PEAF. According to Sager, a PEAF could support tests of closure levels of air, water, food, waste, energy and mass within a Controlled Ecological Life Support System (CELSS). Sager and other attendees considered the interaction of these life support functions critical to simulate.

A PEAF could support study of the habitation aspects of a CELSS, such as psychological impacts, dietary preferences and allergic reactions of the crew, and contaminant control within the CELSS. According to Sager, there is a requirement for the modular development of habitat systems and life support systems, and a PEAF could serve as a forum for this development.

3.2 NON-NASA NEEDS FOR A PEAF

Representatives from both the Bureau of Mines (BOM) and the US Army Corps of Engineers were present at the meeting. Respectively, those representatives

were Egons Podnieks for the BOM and Jim Waddell and Al Smith for the Corps of Engineers. Their presence represents an effort to form a consensus of support for a PEAF among several interested US Government agencies.

3.2.1 Bureau of Mines

The BOM is charged with providing the nation with its necessary mineral resources, whether the nation seeks them on the Earth, the Moon, or Mars. However, the Bureau is also interested in using findings on extraterrestrial minerals and conditions to improve mineral recovery and mining efficiency here on Earth. Considerable research was done during the Apollo period by the Bureau on lunar simulants, and interest is now increasing in doing such research again.

According to Podnieks, a PEAF can be used as a simulator to study mining conditions and equipment performance with an eye to improving terrestrial operations. Surface friction, rock deformation, soil characteristics (such as compaction), and surface material adhesion and cohesion are among the conditions which need study. A PEAF can be used to evaluate and interpret environmental effects. It can be used to perform research in fundamental areas of mining, such as fragmentation, ground stability, materials handling, and resource processing. It can help develop novel mining and processing methods and systems, as well as aid prototype mining equipment analysis and testing for long-term reliability and ease of maintenance.

A PEAF could also yield important information about operational aspects of terrestrial mining equipment, according to Podnieks. For example, it could provide information about power sources, human operator factors, robotics, automated systems, and the effects of hazardous conditions on equipment (for example, dust). These studies could provide information for use on Earth.

However, Podnieks stated that the Bureau is also interested in using a PEAF to test prototype systems for use in space, under conditions such as the lack of atmosphere, extremes of temperature, solar flares, dust and extreme dryness (electrostatic adhesion), and the long lunar day-night cycle. A PEAF could also allow study of equipment assembly in a full-scale realistic setting.

3.2.2 US Army Corps of Engineers

The US Army Corps of Engineers (USACoE) shares many concerns with the BOM, especially in the area of equipment operations and development. The Corps also has unique requirements for a PEAF, both in terms of terrestrial applications and in terms of its mandate to meet the national challenges of the US space program.

According to Jim Waddell, the construction industry suffers from a low level of investment in research and development. A PEAF could augment this investment, inevitably improving the industry. Improvements could be felt in commonplace projects as well as in more exotic projects occurring in extreme or hostile environments.

Waddell indicated that a PEAF could improve terrestrial construction practices by allowing for detailed study of construction in a controlled setting. The PEAF could support the study of wearing surfaces in dynamic load-bearing equipment components, such as joints, couplers, bearings, etc. It could support the study of equipment components which restrain the flow of particles of gases, such as seals, filters and other components. It could aid the development of infrastructure systems such as water supply, waste processing, heating, ventilation and air conditioning, lighting, and shielding. A PEAF can permit the verification of analytical studies of terrestrial building assembly procedures.

Looking to more exotic applications, Waddell stated that a PEAF could enable the study of construction procedures in highly controlled, perhaps hostile environments. It could aid in the assessment of vacuum effects on the assembly of building components. Servicing equipment in hostile environments (as the Army Material Command has already done) could be studied, as could the decontamination of equipment, tools and personnel equipment. A PEAF could enable the development and testing of a new family of light-weight power and hand tools which have particular application to undersea construction and services.

According to Waddell, a PEAF can further Construction Productivity Advancement Research (CPAR) in a laboratory of sufficient scale to build full-size structures. Advanced construction management, including scheduling, quality control, and life cycle project management can be studied as part of CPAR. Also as part of CPAR, a PEAF could permit study of new techniques in modular, rapidly deployable facilities and utility systems (another strong interest of the military) and developing and testing robotics at all levels and scales.

SECTION 4

PEAF Scope, Function, and Development

This section summarizes the scope, functional characteristics, operational characteristics, and development concepts of a PEAF identified by the attendees.

4.1 SCOPE OF PEAF

As a research facility, a PEAF could encompass all of the areas described above. Therefore, as a research facility, it could meet requirements from a vast range of areas, ranging from human factors to instrumentation development, mining equipment studies to CELSS evolution, and so forth. However, a PEAF is also envisioned as a mechanism to address a number of concerns which are not currently being considered, and which the attendees considered essential.

One such concern is the building of cooperative endeavors for space exploration and development. As the list of proposed analogues in Appendix B indicates, the concept of an analogue of an extraterrestrial environment has already generated interest in both the public and private sectors. It was the consensus of the attendees that a PEAF offers an

opportunity to involve many other constituencies, ranging from private companies who are interested in technology development to the military to even the entertainment industry. Furthermore, it was also the consensus of the attendees that, considering the increasing cost and complexity of space endeavors, such opportunities for enabling cooperative ventures must be taken advantage of whenever they arise.

An allied part which a PEAF can play is that of a mechanism to help fulfill the President's directives concerning commercial space policy. Specifically, a portion of that directive calls for NASA to create opportunities for US commerce in space. The PEAF is envisioned by the attendees as an ideal vehicle, comparable to the Commercial Centers for Development of Space (CCDS), for creating such opportunities.

A PEAF, like a CCDS, could allow a modest capital investment in space development, with an expectation of a return on investment within five years or so -- a normal period for venture capital operations. The return on investment could be real technology development which could be marketable. It was suggested that such a "modest investment" could be as small as $50,000 to $100,000.

An additional advantage to this investment, according to the attendees, is that it distributes the financial burden of space technology development, which until now NASA has carried alone. Also, encouraging private sector investment with genuine marketable results could feed back into the process of creating cooperative ventures.

Sharing investment is not without its pitfalls, however. One concern of the participants is that sharing investment inevitably means sharing control, something which NASA historically has been unwilling to do. However, the participants also concluded that such obstacles are effectively challenges. If a PEAF includes cooperation from international partners, both government and private, as well as from domestic commercial enterprises, the problem of control and cooperation becomes even more acute -- and even more important to resolve, as such cooperation is essential if lunar and Martian settlements are to become realities.

The consensus of the attendees is that a PEAF is an opportunity to learn how to cooperate with international partners in such large-scale projects. It represents an unparalleled occasion for integrating diverse constituencies, ranging from foreign government entities to domestic private companies, into a large-scale, long-range project. The attendees felt that such experience could well be essential in establishing similar large-scale, long-range projects such as extraterrestrial space colonies.

The PEAF could also fulfill the requirement for mission risk assessment. The attendees felt that this sort of assessment is not clearly understood by the general public nor by the news media, who are NASA's interpreters to the public. However, it was the opinion of the attendees that without a facility such as a PEAF,

prior simulations of missions on the lunar or Martian surface cannot take place with the required realism. Such simulations are considered by the attendees to be mandatory if missions are to be as safe as possible. If a mishap then occurs during a mission, prior simulations could allow at least some experience with failure isolation and reconfiguration to provide a safe return of crew. And the attendees felt that a PEAF can allow realtime simulation of the existing mission anomaly, which has significantly aided in solving such anomalies in the past.

According to the attendees, a PEAF can also serve the function of demonstrating long-term planning by public servants. Establishing goals for the next 15 to 30 years is something which the attendees felt citizens expect of their government. Attendees therefore saw a PEAF as an opportunity for the government to be seen as society's tool for converting visions of the future into reality.

4.2 FUNCTIONAL AND OPERATIONAL CHARACTERISTICS

Attendees envisioned a PEAF as a testbed for developing technologies for large scale planetary operations. As such, they felt that one could range in actual physical characteristics from small, isolated, modular units to a huge, centralized research park the size of the Astrodome.

Modular laboratory areas were seen by the attendees as characteristic of any PEAF design. Such modular units are considered particularly essential for development of habitat and life support systems. An advantage of modular development is that the technology can be easily transferred to mission scenarios. A second advantage is that modules avoid operational conflicts over simultaneous use of equipment.

A few other proposals for a PEAF were made by attendees. Los Alamos National Laboratory was suggested as one model. Parts of such national laboratory facilities could be leased out to interested researchers, as in the case of Bureau of Mines research facilities or the Langley wind tunnels. Another model was an industrial park where buildings and common facilities can be easily compartmentalized. A developmental center similar to Arnold Engineering Development Center was a third suggestion.

The largest-scale PEAF which was proposed was a facility with a common area the size of the Astrodome. The central common area could be administered by NASA. Radiating from this central area could be independent laboratory units connected via airlocks. Whether these units were owned by the government or by private industry, they could be used by the owners or leased out to other users.

The actual daily functioning of such a facility was of great concern to the attendees, both in terms of policy implications and in terms of practical details. Daily operational facility control was perceived as an especially sensitive policy issue. Although NASA has historically desired control of its facilities, it is considered unlikely by the attendees

that potential advocates, investors or partners can be interested unless they are guaranteed control over at least their portion of the facilities. Stennis Space Center was cited as a possible model for resolving some of these difficulties since a program of private/NASA use of the facilities there has been instituted.

With respect to operational practices, a consensus was reached that both permanent, full-time researchers and visiting researchers could use the facility on a regular basis. Since all versions of a PEAF involve modularity as either free-standing lab modules or modules connected via airlocks to common areas, both resident and visiting researchers can work there.

It was proposed at the meeting that NASA take responsibility for and operate common areas. Modules connected to common areas could either be owned by the government (not necessarily NASA) or by other partners. Government-owned independent units could be leased to interested researchers. Modular units could contain independent experiments which could access the common area but not contaminate it. They could also be isolated from other experiments. Thus many research projects could occur simultaneously without fear of cross-contamination.

4.3 DEVELOPMENT ISSUES OF A PEAF

Attendees identified four issues involved in the actual construction of a PEAF. They are scientific support, political support, visibility, and initial size.

To obtain scientific and industrial support, it was the meeting consensus that it is essential to attract "champions" whose reputations are well-established. These "champions" must be willing to support a PEAF and promote its establishment.

According to the attendees, political support for national facilities has recently been problematical unless the funding could be distributed among many constituencies. However, if a PEAF is to be a distributed facility, attendees had serious questions about the capability of performing large-scale integration activities. Such integration was perceived as essential to both planetary surface mission simulations and spinoff commercial benefits.

Visibility is also perceived by the attendees as essential to attract full-scale political support. But according to Kenneth Cashion of Stennis Space Center (SSC) and other attendees, visibility and significance are not necessarily identical. Although no specific "highly visible" components were identified, attendees felt that it might be necessary to start a PEAF with such a visible component to ensure full-scale, long-term public support. Then the rest of a PEAF could be built once that support had been gained.

The consensus was that whatever the size of the initial construction, a PEAF must be given genuine organizational support by NASA. The question of phased development was left open, in light of

the questions of visibility and political support.

SECTION 5

Implementation Issues

This section discusses benefits, disadvantages, and possible consequences of a failure to build a PEAF.

5.1 BENEFITS AND DISADVANTAGES OF A PEAF

Attendees considered an analogue facility to be necessary for performing mission simulations in a high-fidelity environment, thereby reducing mission risks and aiding mission planning. Such a facility can also be considered essential to developing the technology needed for such missions and for large-scale planetary facilities. A PEAF, in other words, could allow for a long-term, focused study of the complexity of large scale human exploration of extraterrestrial bodies. The attendees saw this as the direct and most obvious benefit of a PEAF.

However, they believed that a PEAF could also bring many other benefits. Simply by being constructed, it could produce many industrial spinoffs from the technology development necessary for that construction. It could, of necessity, force the creation of mechanisms necessary for cooperation between NASA and the private sector and NASA and possible foreign partners (whether governmental or private). Its sheer size and complexity could force the development of integrative skills which are essential to any large-scale activity.

The cooperation which is deemed essential for its construction could lessen NASA's fiscal burden. This could also feed back into the development of cooperative mechanisms, since financial investment dictates some measure of control. It is anticipated that through small investments, companies could be able to take advantage of a PEAF to create genuinely marketable products over a reasonable period of time, instead of the 20-year lead time usually cited as necessary for a return on a space investment.

Another advantage is that, by bringing in private sector, academic and other partners, public support for space could be considerably broadened, something which is also essential to large-scale extraterrestrial projects. Through its visibility, a PEAF could perhaps provide a source of inspiration for the space program, something which the attendees indicated is missing now.

There are some disadvantages to a PEAF. Depending on selection of approach and capability, it could be expensive. It can raise questions about national priorities and the way public resources are utilized. There may be resulting difficulties in defining its scope and the required level of fidelity.

There could also be problems with obtaining NASA funding for a PEAF because it is currently perceived as a technology exploration activity for a program which does not exist, i.e., extraterrestrial exploration. If a lunar base program is established by the President, then NASA could require such a facility for engineering development and verification.

If international partners are involved, technology transfer issues can present problems which can be difficult to resolve and which may have political repercussions. Issues of control can also arise, whether partners are international or domestic. There may also be problems over control from within different NASA program offices, within other government entities, and among government entities.

5.2 CONSEQUENCES IF A PEAF IS NOT DEVELOPED

It was a consensus that a PEAF is mandatory for testing if we are to go to the Moon and Mars. Reliability of equipment, mission planning and so forth are absolutely critical to mission success.

The absence of a PEAF implies that there can be no integrated, large-scale parallel testing of mission components, either in terms of hardware or mission planning. According to the attendees, this means that risk assessment, a critical part of mission planning, cannot be as complete or as reliable as it could be. It also means that any problems which occur during a mission must be solved in realtime rather than simulated in a realistic training environment prior to the mission.

Attendees felt that development of habitat and life support systems must be modular and integrated into a PEAF for ultimate testing. Without such development and testing, they thought that humans can be severely limited in their ability to function in a planetary environment.

Finally, according to the participants, the absence of a PEAF can mean fewer opportunities to involve the private sector. This can reduce public space support. It can also hinder the development of techniques for managing large-scale projects.

SECTION 6

Summary and Conclusions

A PEAF is seen as a terrestrial facility which will simulate the lunar and/or Martian environments. Both NASA and non-NASA needs were identified. Representatives from the BOM and the US Army Corps of Engineers discussed non-NASA needs in the government sector.

NASA needs are:

- short/long-term mission distinctions
- technology development for Mars/Phobos sample return missions
- better understanding of launch requirements
- systems/technology testbed
- materials processing laboratory
- real-time anomaly analysis
- development facility for crew operations and procedures
- testbed for transportation system servicing procedures
- integration of different areas within NASA
- high fidelity/highly rigorous simulation of hazardous environments
- crew orientation
- crew training
- habitability/human factors/life support studies
- testbed for CELSS closure

- testbed for habitability features of CELSS

Non-NASA needs are:

- simulation of mining conditions to improve terrestrial mining operations
- research in fundamental mining areas
- development of new mining and processing methods
- study and testing of prototype mining equipment and equipment operations
- testbed for space mining equipment and its assembly
- research and development in construction operations to improve terrestrial operations
- verification of analytical techniques for studying building assembly
- study of construction and equipment servicing in hostile environments
- support of Construction Productivity Advancement Research (CPAR)

A PEAF is seen as a possible mechanism for addressing concerns about large-scale space exploration programs. Those concerns are:

- cooperation between the public and private sector
- fulfilling Presidential directives regarding commercial space development
- cooperation with international partners
- mission risk assessment
- fulfilling citizen expectations of government planning.

A PEAF's physical plant could be a number of distributed laboratory modules or it could be one large, central facility with separate facilities radiating from it. PEAF development was thought to be contingent upon the resolution of four issues:

- scientific support
- political support
- visibility
- initial size.

Benefits of a PEAF could be substantial. Those include reducing mission risk, improving mission planning, and aiding large-scale technology development. Others include industrial spinoffs, creation of mechanisms for cooperation between NASA and other public/private entities, and increased public support for the space program.

Disadvantages also exist. A PEAF could be expensive, which raises questions about sources of funding, facility control, national priorities, and utilization of public resources. If a PEAF develops as a cooperative venture with international partners, technology transfer becomes an issue.

However, if a PEAF is not developed, attendees felt that private sector participation in space exploration could be reduced, thus reducing public support for space programs. The absence of a PEAF could seriously impair the possibility of integrated, large-scale mission simulations. Such simulations are needed for risk assessment and training prior to a mission and for anomaly analysis during them. The consensus was that a PEAF is mandatory for testing for lunar and Martian missions.

APPENDIX A

PARTICIPANTS

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APPENDIX B

EXISTING CONCEPTS OF PLANETARY ENVIRONMENT ANALOGUE FACILITIES

Five separate lunar base analogue facilities have already been proposed from several sectors, including universities, a government research institution, and the private sector. Each of the five has a different proponent, proposed location, sponsor and targeted technology.

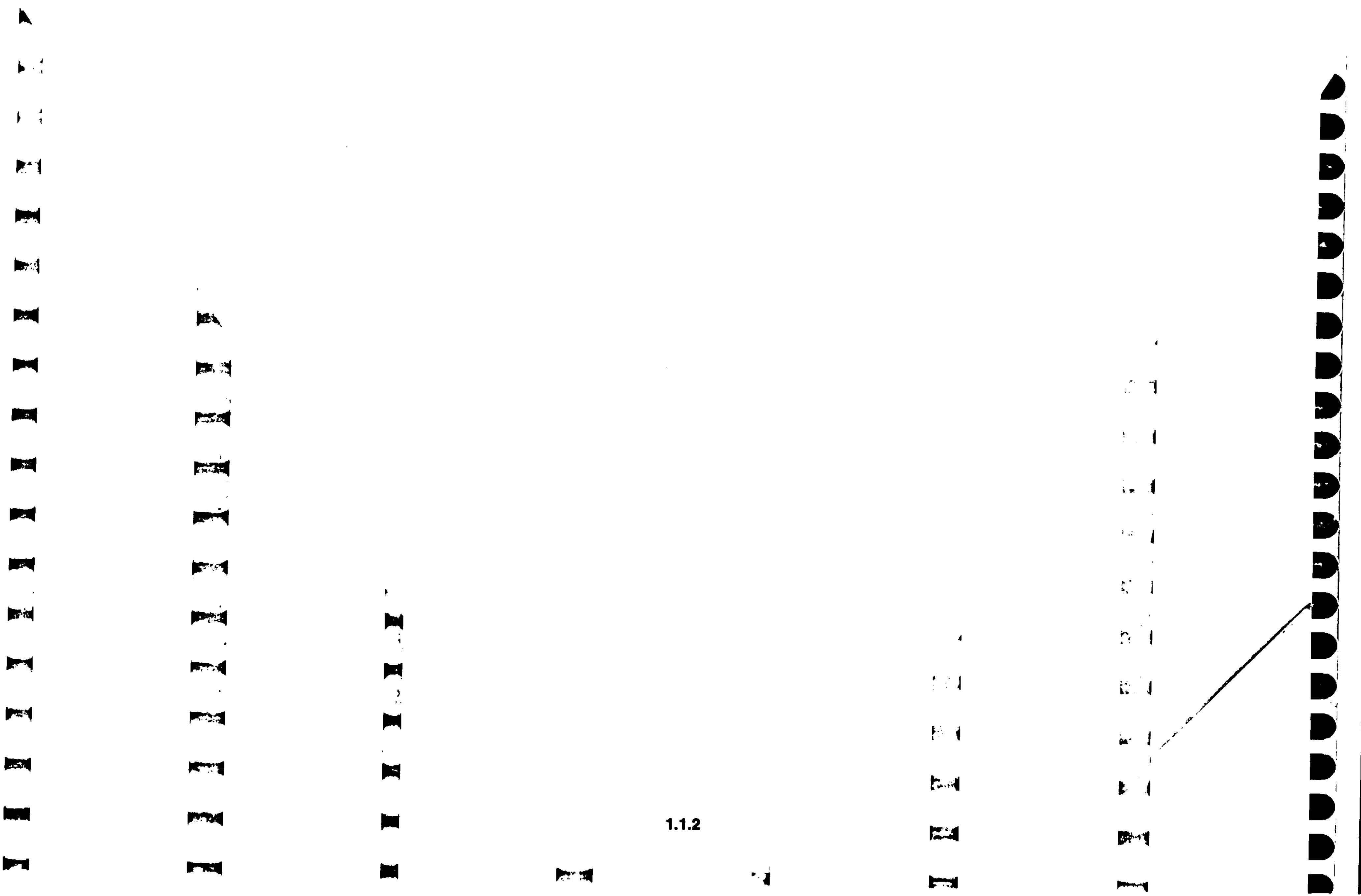
Moon Park, proposed by Professor Kyoichi Kuriki of the University of Tokyo, is sponsored by the Ohbayashi Corporation. The location is still undetermined. Two technologies would be emphasized, a demonstration of CELSS and the investigation of human behavior in a controlled environment.

Lunar Base One is a proposal by Dr. Lawrence Udell of the RELCOR Corporation. Both sponsor and location are undetermined. Infrastructure, building technologies, life support, and the social sciences are the targeted areas of study.

The Antarctic Planetary Testbed was proposed by Dr. Larry Bell of the Sasakawa International Center for Space Architecture (SICSA) at the University of Houston. SICSA would be the sponsor, and as the name indicates, the facility would be located in Antarctica. A wide range of study areas would be targeted, including human physiology and medicine, prolonged isolation and confinement, extraterrestrial food production, field mission analogue and simulations, facility planning, construction demonstrations, automation and robotics, infrastructure development, and international cooperation.

Center for Extraterrestrial Engineering and Construction (CETEC) is a proposal by Dr. Steve Howe of the Los Alamos National Laboratories (LANL). Sponsors would be LANL and the University of New Mexico in Albuquerque (UNM). CETEC would be located at UNM. It would support large scale experiments and development work in construction, mining, processing extraterrestrial materials, and facility operations in vacuum and other non-terrestrial environments.

Lunar Base Simulator, proposed by Dr. Willy Sadeh of Colorado State University (CSU) would be located at Fort Collins, Colorado. CSU would be the sponsor. It would model, study, investigate, and test functions of all subsystems.



1.1.2

Proceedings of the Workshop on Extraterrestrial Mining and Construction

May 2-4, 1989
Golden, Colorado

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Section CONTENTS Page

Section		Page
3.7	CONSTRUCTION SITE DEVELOPMENT SESSION SUMMARY	3-10

TABLES

FIGURES

SECTION 1

Executive Summary

1.1 INTRODUCTION

Lunar and Martian manned bases are among the next major proposed goals for NASA. It is probable that building such bases will require the use of local, planetary resources. Therefore, both mining and construction skills are likely to be necessary. Because these areas lie outside of NASA's areas of expertise, the Planet Surface Systems Integration Agent (PSSIA) at Johnson Space Center (JSC) called for a workshop to convene experts in those areas for the purpose of providing inputs to current PSS studies.

The Extraterrestrial Mining and Construction Workshop occurred May 2-4, 1989 at the Colorado School of Mines (CSM) in Golden, CO. It was co-sponsored by NASA, US Army Corps of Engineers Research Laboratory (USA-CERL), Bureau of Mines (BOM), and the CSM. The goals of the workshop were to:
1) identify problems with extraterrestrial mining and construction and suggest solutions
2) identify linkages and expert resources
3) identify goals for the next workshop

1.2 WORKSHOP FORMAT

The workshop was structured to last two and a half days. Plenary half-day sessions began and ended the workshop. The remaining time was devoted to three concurrent sessions concerned with mining, construction site development, and facilities construction. A schedule is shown in Appendix G.

The opening plenary session consisted of a series of papers dealing with the lunar environment, the lunar base, oxygen production methods, and suggested methods of extraterrestrial mining and construction. Appendix D summarizes those papers. The Mining Session (MS) considered the problems of mining and processing in situ extraterrestrial materials. The Construction Site Development Session (CSDS) examined infrastructure and logistics requirements for a lunar base. The Facilities Construction Session (FCS) considered construction requirements of lunar base facilities, especially the habitat. Though Mars was considered in each session, effort focussed on the Moon due to the far greater body of lunar knowledge. The final half day session was dedicated to summaries of results of the three concurrent sessions.

Approximately 45 individuals attended the workshop. Their backgrounds and interests were diverse, ranging from aerospace engineering and lunar science to terrestrial mining, from conventional and exotic terrestrial construction to heavy equipment manufacturing, from contractors to academic researchers. Appendix F lists the participants.

1.3 PROCEEDINGS FORMAT

This document records the proceedings of the workshop. The Executive Summary, Section 1, overviews the workshop's genesis, goals, structure, participation, central topics, and conclusions. Sections 2, 3, and 4 report the proceedings of the MS, the CSDS, and the FCS, respectively. Conclusions and recommendations of the sessions are assembled in Section 5.

Appendix A lists expert linkages and resources as identified by the attendees, and the goals for the next

workshop are listed in Appendix B. Appendix C contains the "strawman scenarios" which provided descriptions of the lunar and Mars bases used in identifying mining and construction requirements. Appendix D contains brief summaries or abstracts of the plenary papers. It also contains a summary of a third strawman scenario presented by Mr. Ray Leonard, which was not possible to reproduce because of its length. Appendix E lists Proceedings references, Appendix F lists participants and their affiliations, and Appendix G gives the schedule. Appendix H is a complete listing of all topics introduced in the CSDS.

1.4 WORKSHOP TOPICS

Many topics were discussed in the concurrent sessions, but two major commonalities emerged. The first such commonality was the lack of extraterrestrial engineering experience; the second was the lunar base facilities requirements.

1.4.1 The Lack of Extraterrestrial Engineering Experience

The lack of existing engineering knowledge about the Moon was perceived as a central problem by all three sessions. The scientific data which exist are perceived as insufficient, inappropriate, or inaccessible for the detailed engineering needed for lunar base development. Several approaches to this problem emerged.

One approach, agreed upon by all sessions, was an intermediate base phase following the establishment of the earliest lunar base facilities. The intermediate phase would be used for engineering experiments in which terrestrial techniques would be tested for adaptability to the Moon.

A second approach was a MS recommendation of the establishment of a large-scale lunar simulator on Earth. It could be used to simulate mining operations in a lunar environment.

The CSDS recommended that lunar soil samples be acquired for engineering experiments. To judge the validity of engineering experiment proposals, they suggested the establishment of an engineering lunar sample analysis team. That session also recommended that where lunar data exist, they should be codified into the type of design manuals common to terrestrial engineering practice. Three types of manuals were specified: a lunar building code, an equipment and machinery design manual, and a regolith mechanics manual. Where no data exist, the CSDS felt that terrestrial engineering knowledge could be applied, at least initially, to the lunar case.

1.4.2 Key Base Facilities Problems

Each session was concerned with the key base facilities and the problems associated with them. The key facilities of the lunar base, shown in Figure 1.4.2-1, were identified as the habitat, the mine, the power supplies, the transportation network, and the equipment inventory.

The MS focussed on the mining and processing operations and the equipment associated with those operations. The FCS examined all major facilities except the mine operations, but concentrated on the habitat. The CSDS concentrated on the planning, logistics, and base infrastructure necessary to the establishment of the base.

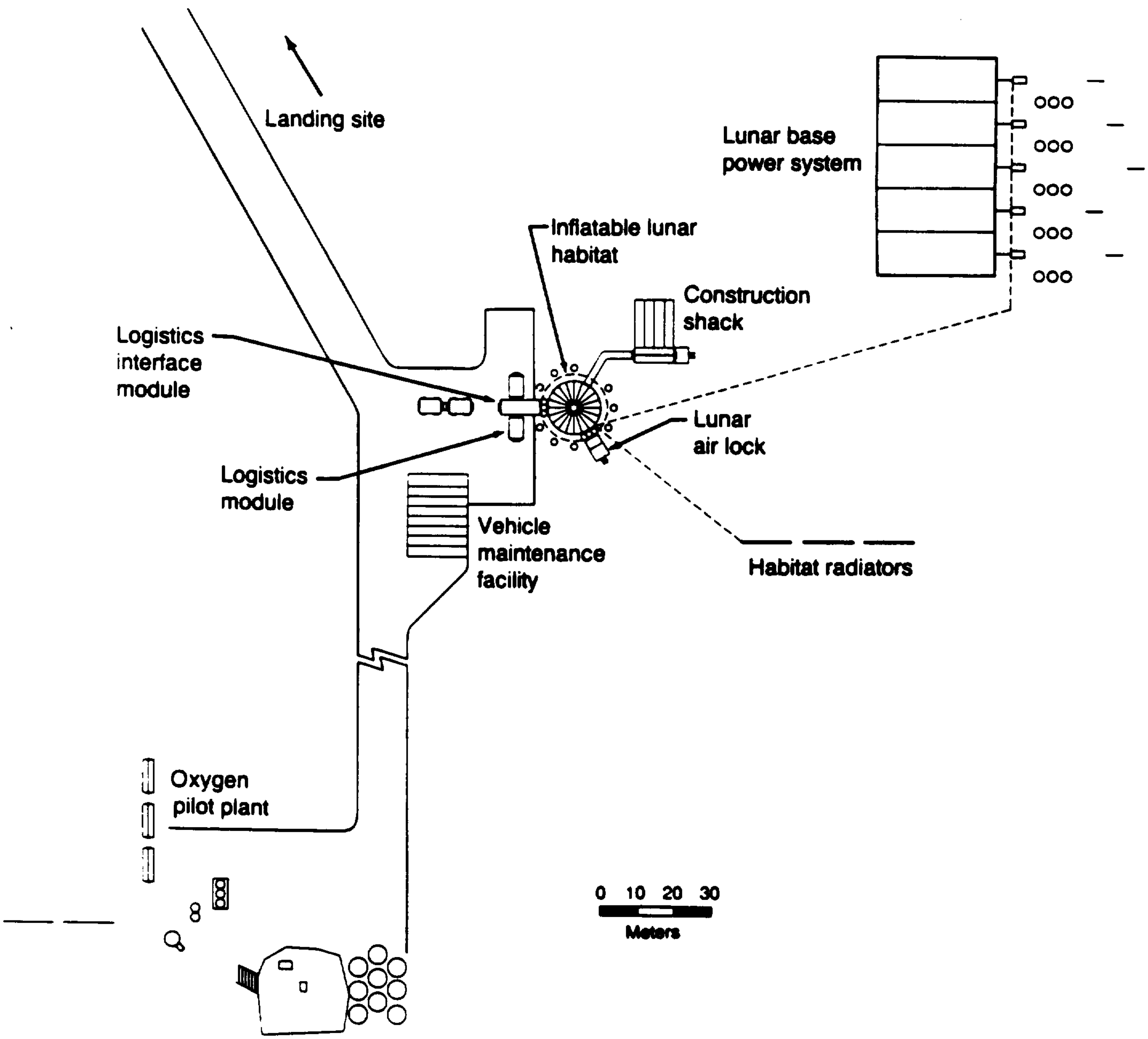

Figure 1.4.2-1.-Lunar Base Layout

The key MS mining concerns were feedstock to yield ratios, leveraging of productivity due to automation and robotics, and power requirements for mine operations. Special attention was given to hydrogen extraction, which attendees perceived as so power-intensive that it should be deferred to later stages of the lunar base. However, oxygen extraction from raw feedstock was seen as equivalent to a small terrestrial mining operation. Therefore, it was thought to be completely practicable. Automation and robotics are perceived to be essential to operating a lunar mine where no crew member is a full-time miner. However, much research and development remains to be done. Though the lunar processing facility will consume large quantities of the available one MW power, attendees felt that power allotments were sufficient for the early mining requirements.

The FCS examined habitat alternatives. They looked closely at options for above and below-ground habitats, concluding that the crew should go below-ground as soon as possible. One reason for this is safety. A second is because below-ground habitats conform to terrestrial construction experience more closely than above-ground ones do. It was felt that known techniques would be safer in the lunar environment.

The CSDS focussed on the lunar base infrastructure. Staging the base in increments was considered vital, so that lunar construction experience can be incorporated into base development. Logistics is also critical to base success, because it deals with construction schedules, launch manifests, and site layout.

Concerns about mining and construction equipment revolved around synergistic use of the equipment. Three issues were involved. The first was whether equipment should be dedicated or multi-purpose. The second was whether multi-purpose equipment should have modular, detachable implements or multiple, permanent implements. The third issue was equipment ruggedness and versatility versus redundancy and singularity. Each choice affects reliability, maintenance, equipment backup, and launch manifest requirements. Further, if mining equipment must be absent for construction tasks, then raw material must be available to keep the processing plant operating. Whatever equipment design philosophy is chosen will have significant strategic consequences for base development.

Concerns about power were concentrated on lunar base requirements and power distribution to mobile equipment. Workshop participants were divided on the issue of sufficiency of power for the base. Some participants felt that approximately one megawatt was sufficient to operate the base, if equipment and operations were scaled to the lunar gravity. Other attendees disagreed. Power distribution to mobile equipment was considered a significant problem, and the CSDS considered it to be the most critical issue affecting mobile equipment.

The impacts of base layout, mining operations, and equipment inventory on road construction were targeted as the main transportation concerns. Both the FCS and the CSDS were concerned with roads. Dust and degradation of regolith roads are perceived as serious problems. To alleviate those problems, participants proposed paving the roads with gravel from beneficiated raw feedstock from the processing plant. Road-building requirements are contingent upon site layout.

Equipment availability will determine feasibility.

1.5 CONCLUSIONS AND RECOMMENDATIONS

Participants concluded that extraterrestrial engineering knowledge is inadequate for detailed lunar/Mars base planning. To rectify that, they recommended that more research should be conducted in such areas as lunar environmental effects, oxygen processing operations, automation and robotics, power distribution, materials, thermal control systems, and pressurized vessel development. Terrestrial experience should be used where necessary, but opportunities to gain lunar experience should be sought out.

To gain that experience, a variety of approaches were recommended. One approach is to perform engineering experiments on existing lunar regolith samples. A second is to codify present knowledge into lunar design engineering manuals. Another mechanism for gaining extraterrestrial engineering experience is a full-scale Earth-based lunar simulator.

Another recommendation is to use incremental base phasing at the lunar base to gain additional engineering experience. Wherever possible, present engineering knowledge should be used. Simple solutions to problems should be sought in preference to high-tech, complex, elegant, and difficult ones.

A second conclusion was that key base facilities definition entails many problems and issues. To resolve those, participants generated a number of recommendations. Participants recommended that realistic power requirements be established for both the base and equipment. An equipment design philosophy and the resulting design parameters must be established. Base layout must be determined so that logistics can be planned. Safety factors for habitation must be established. Hydrogen extraction should be deferred until the later stages of base development, unless the process is less power intensive than first thought.

A third conclusion was that the workshop had identified the general character of the problems of extraterrestrial mining and construction. However, a better resolution of problem specifics and solutions is needed. The recommended next step is a second workshop which builds upon the results of the first. A second workshop is currently being planned.

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SECTION 2

Mining the Surface of the Moon: Issues and Problems

2.1 INTRODUCTION

The Mining Session (MS) identified three topics of concern to mining the lunar surface: lunar surface conditions, mining operations, and mining equipment. In each case, the participants identified the topical problems and then, where solutions were available, recommendations were made.

2.2 LUNAR SURFACE CONDITIONS

The MS was concerned with the lunar temperatures, vacuum, dust, and regolith density.

2.2.1 Surface Temperatures

The surface temperature at the Apollo 17 landing site varied from -171°C to 111°C, and the Apollo 15 site temperatures were about 10°C cooler [Taylor, 1982]. MS participants felt that these temperature extremes represent significant obstacles to equipment operations, especially during the lunar night. The primary concern is that the structural metal could become so cold that it would become brittle and crack. The primary recommendation was that operations should take place only during the lunar day. A secondary recommendation was that waste heat could be used to heat equipment.

2.2.2 Vacuum

Bureau of Mines (BOM) research with lunar simulants indicates that surface friction will be much greater on the Moon than on Earth. This is due to the lack of a monolayer of water vapor between surfaces, which is absent on the Moon due to the lunar vacuum.

On Earth, the monolayer acts as a lubricant. In the absence of such a lubricant, rock and tool surfaces are expected to adhere, creating a problem with chip formation, removal, and clogging during drilling. Also, ordinary lubricants cannot be used because they will vaporize in the vacuum.

Another effect of the vacuum is that lunar material will compact when moved. This is due to particulate cohesion caused by the absent water vapor monolayer [Podnieks and Roepke, 1986].

No specific solutions were recommended for the lubrication problem. The compaction problem is discussed further in Section 2.2.4.

2.2.3 Dust

The particles in the lunar regolith are extremely fine-grained, half less than 74 microns in size. They also have a positive electrostatic charge, because the solar wind strips electrons from the particles [Carrier, 1989]. Electrostatic adhesion of fine-grained dust particles to all surfaces is the result, and it is a serious problem for lunar operations.

Developing equipment surfaces which are antistatic or anticharged was one recommended solution to solving this problem. Another recommendation was the development of a chemical applicant which could be applied to surfaces and which could serve as a "grounding film" for spacesuits and equipment.

2.2.4 Lunar Regolith Density

Lunar regolith is packed loosely at

the surface. At a depth of 10 to 20 cm, however, it becomes tightly packed. It is five to 20 percent more compressed than terrestrial soil can become when compacted by heavy road-building equipment [Carrier, 1989]. This high density is due to continual meteoritic impacts, which keep the surface loose but also produce shock waves that compact the underlying regolith [Taylor, 1982].

The compaction of the regolith is a serious concern for mining equipment design. Equipment can have lower mass because of the lower lunar gravity. It must also have sufficient power and cutting capability to break and move heavily compacted material. This represents a design conflict. However, if explosives are used to dislodge the regolith, the conflict is avoided. Explosives use was therefore the recommended solution.

2.3 MINING OPERATIONS

Mining operations discussions focussed on processing and automation. Processing topics included expected yields, recovery efficiencies, and processing power. Automation topics included operation size and productivity.

2.3.1 Processing

Workshop target yields for a lunar mine were given by a strawman scenario originated by the Planetary Surface Systems (Office of Exploration) and published in the Planetary Surface Systems Requirements Document (PSSRD). The strawman scenario is reproduced in Appendix C. A lunar mine based on concepts in the strawman scenario is shown in Figure 2.3.1-1. Suggested target yields are given in Table 2.3.1-I:

TABLE 2.3.1-I.-MINE TARGET YIELDS

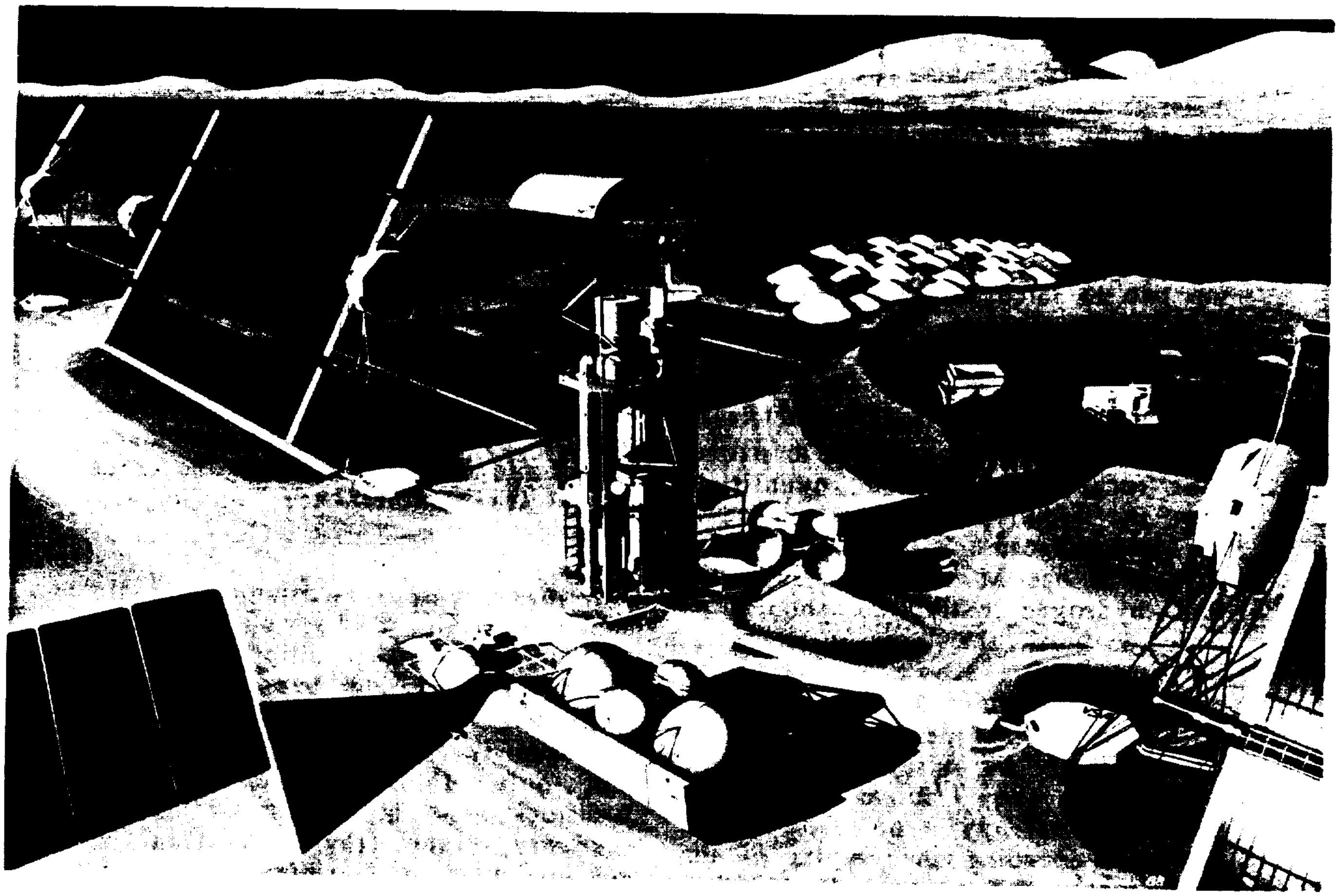
Product	Amount (t/yr)
O2	150
H2	15
C	5
Other volatiles	Capture all available
Metals	50
Ceramics	100

Three assumptions were involved in setting the targets. First, the targeted yields include all product quantities needed for life support. Second, the required purity of products for all purposes is achieved in processing. Third, the raw feedstock necessary for the oxygen yield is estimated to be 45,000 metric tonnes (t) per year, based on recovery of one percent of the available oxygen.

Mining session attendees pointed out that the oxygen recovery rate depends upon the efficiency of the process chosen. During the workshop, the assumed efficiency of the baseline ilmenite reduction process was one to three percent.

Processing power for 150 t of oxygen per year was scaled to a proprietary process which requires three to four megawatts (MW) for the production of 1000 t of oxygen per year. Based on this, the production of 150 t per year was estimated by the participants to require up to 0.5 MW. The PSSRD scenario stipulated a one-megawatt-electric (MWe) nuclear power plant for the lunar base. The plant would supply all power for the habitat, laboratories, materials processing, and mining equipment.

The one MWe power supply constrains the possible mining processes, operational size, and available equipment power. For



example, materials processes such as carbochlorination [Waldron, 1988] require high process power levels, so participants ruled them out. Also ruled out were various high temperature production processes which have 40-50% efficiencies, but which require more than one megawatt.

To meet the strawman's requirement for a H_2 return of 15 t/yr, participants estimated that approximately 330,000 t/year of raw material would have to be mined, and all of the available hydrogen would have to be recovered (100% efficiency). The calculated quantity of raw material is based on an average H_2 content of 50 ppm in the top two to three meters of regolith.

One opinion expressed during the workshop was that if it was necessary to heat the top two or three meters of the regolith to 600 or 700°C to extract the targeted hydrogen quantities, then the required energy could equal 20 MWe. Because initial base power is limited to one MWe, it was suggested that hydrogen extraction should be delayed until the base was better developed. However, because attendees considered the available data on hydrogen extraction processes to be insufficient at the workshop, they chose to delay recommendations on hydrogen extraction pending the results of the In Situ Resource Utilization Workshop in June.

2.3.2 Automation

The extent of automation of the mine will be important to both its size and productivity. On Earth, a 300,000 t per year operation is considered small next to the average mine production of 2,000,000 t per year. A 300,000-t operation can be handled with three people and a loader, truck, and crusher. Such an operation can be managed with minimal crew and equipment because terrestrial mining productivity rates now equal 40 t per manhour. Whether an operation of similar size is possible on the Moon depends upon the level of productivity which can be expected there.

None of the initial eight crew members will be full-time miners. Participants estimated that a lunar robotic mining system employing eight to ten robotic scrapers or backhoes might increase productivity over current terrestrial achievements by merely 10 to 15%. Initial lunar base equipment allocations will comprise one to two pieces, not eight to ten. Telemetered control from Earth could increase productivity by permitting equipment to operate when no lunar operators are available. However, such leveraged productivity will be limited by the available equipment. If robotic equipment must be shut down for maintenance or repair, work time is decreased.

For these reasons, participants again concluded that it was unlikely that the 300,000 t of regolith needed for the hydrogen could be mined in a year. If the requirement to recover hydrogen is deferred to a later time, and only 45,000 t per year of feedstock need be mined for oxygen recovery, then the operation is considered more feasible.

Because the potential impacts of automation on a lunar mining operation are so important, it was recommended that more research in automation and robotics be conducted. Teleoperated systems and automated mining systems were considered especially important for lunar mining.

2.4 EQUIPMENT

2.4.1 Equipment Power and Distribution

The one MWe base reactor is the source of all power for the base. Approximately 0.5 MWe will be required for oxygen processing; the remaining power must suffice for all other base facilities, as well as mining equipment.

To estimate required equipment power, participants scaled terrestrial front-end loaders (FEL) to the lunar environment. The Caterpillar 966E wheel loader's flywheel power is 216 hp or 161 kW; the Caterpillar 980C wheel loader's flywheel power is 270 hp or 201 kW [Caterpillar, 1988]. During the workshop, the power of each of these machines was estimated to be 300 horsepower. Their productivity was estimated at 800 t/hr. It was assumed that lunar loaders would most likely be built of aerospace materials, thus reducing their weight compared to machines made of conventional terrestrial materials. Participants expected a lighter machine to require less power, even on Earth, and with the loading factor cut by 6 on the Moon, they felt that a lunar loader could operate with minimal power. Also, as discussed in Section 2.2.4, Lunar Regolith Density, the use of explosives dispels the need to break heavily compacted regolith prior to digging. Therefore, a scaled power approximation for a 150 t/hr lunar FEL was 50 hp or approximately 37 kW.

Power can be distributed to mining equipment in several ways. Participants discussed both non-conventional microwave and solar systems and conventional cable, battery, and trolley/battery systems, but conventional systems were emphasized.

On Earth, trucks which are travelling long distances use trolley systems during hauling runs. Trucks run on batteries either in the mining pit or when hauling ore long distances where the trolley is unavailable. Digging equipment which permanently resides in the mining pit is powered from cables which attach to a substation in the pit. When digging equipment must be moved to another pit, it operates from batteries during the move. Underground equipment operates entirely from cables, but only for short distances of 150-200 feet. This power distribution system was viewed by the participants as adaptable to the lunar mine.

2.4.2 Equipment Synergy

A constraint imposed by the PSSRD was one of maximum synergy between mining and construction equipment. This was to be achieved by attaching construction implements to mining equipment, reducing the need for specialized construction equipment.

Participants felt that while such synergy could be desirable, it could also be difficult. The required mining equipment could differ from the required construction equipment to such an extent that simply adding attachments would be insufficient for conversion. Participants foresaw other problems with stockpiling, maintenance, backups, scheduling, and specialization.

If mining equipment which normally feeds the process machinery is operating elsewhere as construction equipment, then a stockpile of feedstock must be available at all times. So time must be regularly scheduled to build such a stockpile.

Equipment which has multiple uses must be reliable and easily maintained and/or it must have backups. If backups are necessary, the original goal of eliminating additional machinery is contravened.

Construction and mining operations are already difficult to schedule because of the lunar day-night cycle and scarce crew time. If machine use must be shared, scheduling becomes even more difficult.

Multi-purpose equipment serves no single function well. This is true even when limited to the basic mining functions of loading and hauling. The central aspects of equipment optimization were detailed by Caterpillar engineer Charles Scheidle in "Mobile Mining System Strawman," prepared for the workshop.

Scheidle remarked that a mining machine can perform one or both hauling or loading functions. "By limiting a machine to a single function," he stated, "its design can be optimized for that function. This optimization is most evident in the ratio of the payload the machine handles to the empty weight of the machine." Those ratios are shown in Table 2.4.2-I.

TABLE 2.4.2-I COMPARISON OF FUNCTION VS. REPRESENTATIVE PAYLOAD TO WEIGHT RATIO

Function	Payload to weight ratio
Hauling	>1.0
Loading	<0.3
Loading and Hauling	~0.6

Scheidle commented that where loading conditions are difficult, dedicated loaders are indicated. If hauls are long, dedicated hauling machines are needed. If loading is

TABLE 2.4.2-II MOBILE MINING SYSTEM ADVANTAGES VS. DISADVANTAGES

Advantages	Disadvantages
Flexibility in face location, processing location, and route between	Mobile systems require roads. Road construction and maintenance requirements depend on the mine site environment and the particular mining machine being used.
Wide flexibility in capacity by adding or removing machines from system	Traveling machines can generate dust.
Assembly, servicing, maintenance, and repair of mobile equipment in a shop instead of the field	If mine production is continually increased by adding machines, at some point, machine congestion will become a problem.
Equipment is easily moved to new location	
No extensive site preparation required	
Failure of one machine will not disable entire operation unless that machine is the entire system	
Backups or spares can easily be placed into the system, insuring against decreases in production	

easy and hauls are short, a machine which combines both functions may be the best solution. Single-machine mining systems have advantages in reducing the required spare parts inventory and the number of backup machines. [Scheidle, 1989]

Synergistic use of equipment may also favor mobile mining systems over stationary ones. Part of Scheidle's Mobile Mining System Strawman was an overview of the advantages and disadvantages of mobile mining systems. Those advantages and disadvantages are summarized in Table 2.4.2-II.

Scheidle stated that mobile and stationary equipment are often combined into a single mining system. Mobile equipment is usually used at the mine face and for short distance transport to the stationary system. Stationary equipment then transports ore to the processing plant. He continued:

> Such a [combined] system can often maximize advantages and minimize disadvantages. For either 100 t or 1,000 t annual oxygen production (10,000 t and 100,000 t regolith respectively), any of the above systems [mobile, stationary, combined] are feasible. By terrestrial standards, these are low production rates and can easily be achieved with a small number of machines. The number of machines is probably best determined by the amount of production decrease allowable upon the failure of any single unit. Once the desired number of machines is determined and the number of operating hours per year specified, the size of the machines can be quickly derived. Of course, load times, travel speeds, and travel distances will not be included in this calculation. [Scheidle, 1989].

Though available information appeared to support mobile systems, participants chose not to recommend them over stationary systems. They felt that insufficient information existed to make a choice.

2.4.3 Baseline Equipment Systems

The final goal of the mining session was to develop one baseline system, which included all the subsystems of breaking, digging, loading, unloading, hauling, storage, retrieval, sizing, beneficiation, and disposal. Three sub-goals were listed:

- Identify mining system concepts
- Select priority systems and components
- Assign parameters to baseline concept.

A variety of options were considered for each mining subsystem. These options are shown in Table 2.4.3-I. Terrestrial versions of four of these options are shown in Figure 2.4.3-1a-1d.

In comparing the options for each subsystem, certain assumptions about the entire mining operation were made. These were:

- Diggable regolith
- Material for oxygen and hydrogen
- Excavation rate of 333,000 t/yr
- 4000 hr/yr or 8000 hr/yr operation
- Fixed processing plant
- Daylight operations only

Several equipment attributes were also identified as critical or important. These are shown in Table 2.4.3-II.

For the digging and loading subsystem, the FEL and the shovel were recommended. FELs were recommended because they can load

TABLE 2.4.3-I-MINING SUBSYSTEMS AND OPTIONS

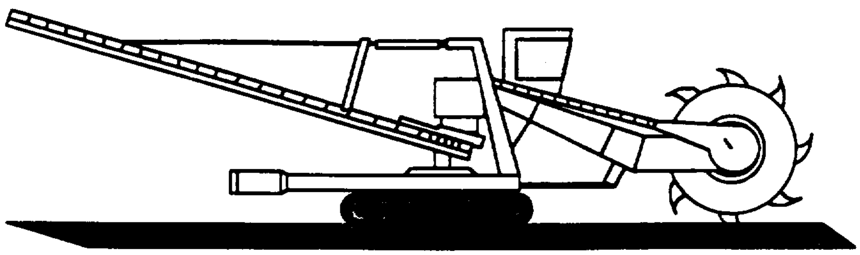
Subsystems	Options
Breaking	Blasting, Mechanical: Scooping, Ripping, Hammers, Electrical, Fluids, Thermal
Digging and Loading	Shovels, Front-End Loaders (FELs), Scrapers, Bucketwheel Excavator, Dozers, Augers, Draglines, Containers/Boxes, Slusher, Electrostatic, Electromagnetic
Hauling	Trucks, Pipe Conveyors, Catapult, Pipeline, Railcar, FEL, Scraper, Aerial Tram, Auger, Gravity, Trailer, Containers/Combination, New-Technology Electric Methods
Unloading	Fork-lift, Selfdump, Auger, Blower, Catcher, Bucket
Storage	Piles, Containers, Bins: Atmospheric, Insulated, Pressurized
Retrieval	Conveyor, Auger, Chute, Bucket, FEL
Sizing	Grizzly, Screening, Casting, Cyclone
Beneficiation	Magnetic Separation, Electrostatic Separation, Thermal, Flotation, Gravity, Grinding, Abrasion

TABLE 2.4.3-II. EQUIPMENT ATTRIBUTES

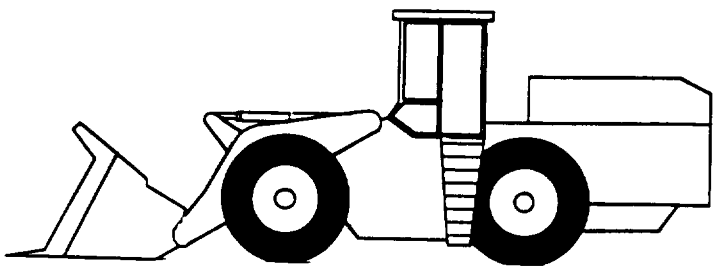
CRITICAL	IMPORTANT
Mass	Technology Availability
Efficiency per unit mass	Technological Risk
Reliability	Development Time
Maintenance Requirements	Flexibility within the Mining System
Energy	Applicability to Other Bodies
Labor Dependency	
Versatility	
Applicability for Expansion	
Automation	

and haul material. This flexibility was considered especially desirable in early stages of base development, when dedicated machines are unlikely to be feasible.

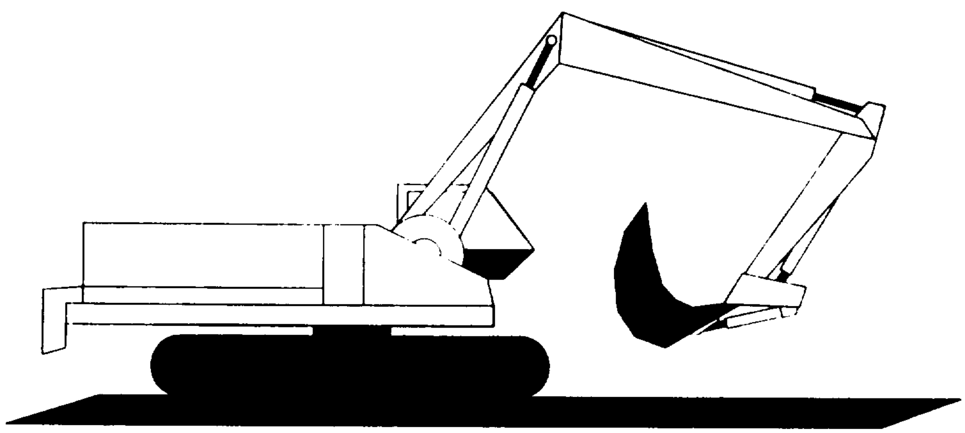
A shovel/backhoe and truck combination was also thought to offer many advantages. It has a long lifespan. It is flexible in that many implements such as blades, buckets and hydraulic hammers can be attached. It can move itself by grabbing the regolith and sliding without engaging the power train. A shovel with a small bucket can cycle in one-third the time of a FEL, because a FEL must move itself. Although a shovel is more massive than a FEL, that can be addressed by setting the shovel on a tripod. Anchored like this, a shovel has a 20-meter, 300-degree reach, permitting it to dig the same amount of material as a FEL with one-tenth the mass. The shovel function itself is easy to automate. However, a truck is



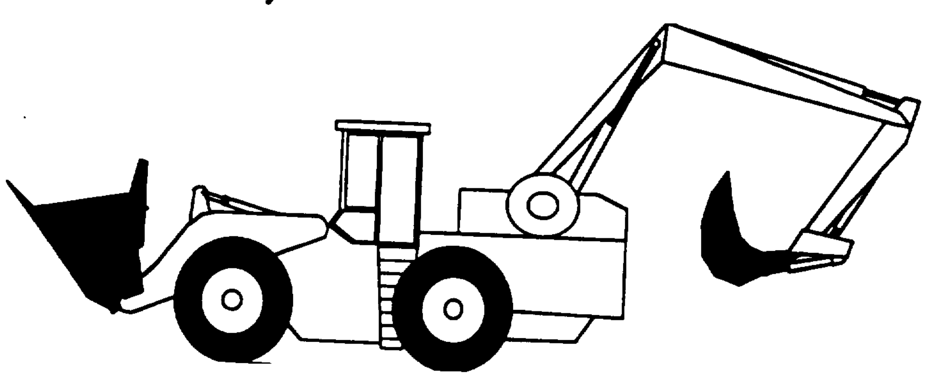
a-Bucketwheel Excavator



b-Front End Loader



c-Hydraulic Hoe



d-Front End Loader and Backhoe

Figure 2.4.3-1.-Terrestrial Earth-Moving Equipment

required to move the material dug by the shovel, which decreases the mass advantage of the shovel.

The bucketwheel was rejected because participants considered it to be too heavy and complicated. Also, bucketwheels are designed for larger volumes than expected at the lunar base. The auger was rejected because it is both massive and fixed in place. Slushers are also fixed, but they are too limited for the required tonnages per year. Dozers were eliminated because participants considered them to be energy inefficient and limited in use to excavation. Electrostatic and electromagnetic methods were considered too unfamiliar for discussion. Participants felt that it was better to consider only known systems for the early stages of lunar development. However, they recommended additional research on these and other components, as shown in Table 2.4.3-III.

TABLE 2.4.3-III.-PROCESSES DESERVING FURTHER STUDY

Electromagnetic Separation (junk-yard magnet)
Ballistic Thrower (to throw buckets of material in ballistic trajectories)
Auger in later base phases
Electrostatic Processes
Containers in Trucks

Trucks and pipe conveyors were recommended for hauling. Trucks can be used with any digging machine. Pipe conveyors were recommended because participants thought the required non-metallic pipe could be produced early at the base from in situ materials, eliminating the need to transport additional mass from Earth. Although pipe conveyors would require a fluid medium such as air to move material through the pipes, participants felt this medium could be reclaimed and re-used. Fixed rail was rejected because it would require significant initial installation and frequent movement of the track. The aerial tramway was rejected because elevation is its main advantage; however, participants considered that advantage to be inappropriate for the small size of the lunar mining operation. The aerial tramway also requires significant installation and permanent structure. Pipeline also requires heavy initial installation and permanent structure.

Stockpiles are a simple storage technique, and were recommended on that basis. The FEL and the pipe conveyor, recommended above for digging, loading, and hauling, were recommended as retrieval components for the same reasons. Recommended sizing and beneficiation systems included the grizzly (coarse sieve), vibration screen, cyclone and magnetic separator. One participant suggested that junkyard-type electromagnets could be used to separate magnetic lunar agglutinates from the non-magnetic regolith. However, since this is not a mining technology, participants were hesitant to pursue it at the workshop. The cyclone was felt to be the best technique for sizing material, especially for extracting hydrogen. Then the underflow from the cyclone could be channeled into an air crusher. Fifty percent of the flow would be rejected, so that it would not run through the extractors. Thus the concentration of hydrogen in the air flow would be improved.

The recommended baseline mining system is summarized in Table 2.4.3-IV.

2.5 MINING SESSION SUMMARY

Three topics concerned the Mining Session: lunar surface conditions, mining operations, and mining equipment.

TABLE 2.4.3-IV.-BASELINE MINING SYSTEM

MINING SUBSYSTEM	MINING OPTION
Digging and Loading	FEL, Shovel
Hauling	Trucks, Pipes, Conveyors
Storage	Stockpiles
Retrieval	FEL, Conveyor
Sizing and Beneficiation	Grizzly, Vibration Screen, Cyclone, Magnetic Separator

Important lunar surface conditions included surface temperatures, vacuum, dust, and the lunar regolith density. Participants feared that the cold lunar night temperatures would embrittle equipment metal, causing it to fracture. To solve embrittlement problems due to the cold lunar nights, participants recommended daytime operations. Explosives will dislodge compacted regolith, dispelling the need for high-powered regolith-moving machinery.

Mining operations discussion focused on yields, recovery efficiencies, and processing power. The mining requirements for 150 t/yr lunar oxygen products were considered achievable and comparable to a small terrestrial sand and gravel operation. Participants tentatively concluded that hydrogen extraction should wait until late base development, because of the large quantities of required feedstock. However, the consensus was that available hydrogen-extraction information was insufficient.

Automation is crucial to achieving desired mining goals. Robotic productivity is uncertain. Participants' recommendations were to increase research in automation and robotics. Special research emphasis was placed on teleoperated systems and automated mining systems.

Equipment concerns included equipment power and distribution, mining/construction equipment synergy, and the establishment of a baseline mining system. Participants concluded that the one MWe nuclear plant provided sufficient power for oxygen extraction and equipment operations

A conventional power distribution system using trolleys, cables, and batteries was emphasized over unconventional microwave or solar systems.

Mining/construction equipment synergy offers advantages in terms of reduced mass, but it entails other problems. These include specialized design needs, equipment availability, scheduling, and reliability and maintenance.

A baseline mining system would include the subsystems of breaking, digging and loading, hauling, unloading, storage, retrieval, sizing, and beneficiation.

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SECTION 3

Developing a Lunar Construction Site: Issues and Problems

3.1 INTRODUCTION

The Construction Site Development Session (CSDS) identified five topics and more than 50 sub-topics associated with lunar site development. The five topics were:

1)Definition of a lunar construction site
2)Identification of required exploration documents
3)Definition of site development's relationship to mining and facilities construction
4)Definition of assumptions to be used in discussion of lunar construction site development
5)Identification of challenges and problems involved in lunar construction site development.

The complete list of subtopics is shown in Appendix H.

The CSDS focused on topic 5, construction site development challenges and problems. Thirteen challenges and problems were identified. the 13 were re-combined into five categories for discussion:

- general design principles
- lunar base development philosophy
- lunar base construction site requirements
- construction site operations
- lunar design manuals.

The 13 challenges and problems and their re-categorizations are tabulated in Table 3.1-I.

CSDS participants broke into three subgroups to consider the challenges and problems. For each one, the subgroup completed a Technology Requirements Form. An abbreviated version of this form is shown in Table 3.1-II. The completed forms captured the discussion for each challenge or problem.

Section 5 of the form requested an assessment of technology readiness level for each challenge or problem. Seven numerical levels were indicated. These are shown in Table 3.1-III.

TABLE 3.1-I.-CONSTRUCTION SITE DEVELOPMENT CHALLENGES AND PROBLEMS

Challenge/Problem	Category
1)Simplicity 2)Learning from experience	General Design Principles
3)Lunar base development philosophy	Lunar Base Development Philosophy
4)Construction logistics 5)Site layout 6)Foundation design and/or construction 7)Road serviceability & surface transport	Lunar Base Construction Site Requirements
8)Equipment 9)Mobile Power 10)Illumination	Construction Site Operations
11)Equipment standards and guidelines 12)Lunar design code 13)Regolith behavior	Lunar Design Manuals

3.2 GENERAL DESIGN PRINCIPLES

Simplicity was one of two general design principles participants identified as crucial to successful lunar site development. The other was learning from experience as it occurs on the moon.

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TABLE 3.1-II.-TECHNOLOGY REQUIREMENTS FORM

1) Identify technology area.
2) State its need or requirement.
3) Identify possible sources of information.
4) Identify any required performance parameters.
5) Define present technology readiness level.
6) How long will it take to bring the technology in question to technology readiness level 6?
7) Identify any adaptable terrestrial technology equivalents.
8) What is the criticality of this technology area?
9) What principal conditions would be required to test an outcome/product of this technology development in a terrestrial facility?

Participants considered the simplest solutions often to be the most reliable ones. They feared that NASA and the science community might aim for absolute rigidity, precision repeatability, and use of robotics, etc., achieving "technological overkill." They believed looser tolerances would be desirable, and that primitive technologies should be examined as role models for development. This approach should not be limited to machinery, but should also be applied to planning, design and operations of the entire lunar complex. Participants were especially concerned that valid, but technologically simple, solutions to problems could be overlooked or ignored because they are not cutting-edge technology.

Learning from past and on-going experience will be important to lunar base development. To achieve this, information from new experiences must be captured as they occur. Specific technical information about lunar construction clearly must be documented. However, participants also felt that it was important to document organizational, management, and personal experiences.

TABLE 3.1-III.-TECHNOLOGY READINESS LEVELS

Readiness Level	Definition
1	Basic principles observed and reported
2	Technology concepts and/or application formulated
3	Analytical and experimental critical function or proof-of-concept
4	Component and/or breadboard equivalent validation in lab
5	Component and/or breadboard equivalent demonstration in relevant environment
6	System validation model demonstrated in a simulated environment
7	System validation model demonstrated in space

Lunar construction technology readiness is presently at Level 1. To gain some experience with lunar construction before going to the moon, participants felt that a lunar simulator should be built on Earth to test and validate experimental lunar construction methods. Other studies and experiments which do not require a simulator are also necessary, especially those dealing with the mechanical properties of the lunar regolith.

3.3 LUNAR BASE DEVELOPMENT PHILOSOPHY

Two lunar base development philosophies were identified. The first is a philosophy of gradual, incremental development. The second is one of large, discrete change.

The chief functional difference between these two philosophical approaches is an interim facility for construction. The gradual approach would have such a facility following the initial base establishment. The discrete approach would not.

Subsequently, the discrete approach entails the use of pre-fabricated structures imported from Earth. Such structures would include all working, living, and storage areas. The only construction activities required for these structures would be fitting them together on the Moon. Therefore, experience in handling lunar materials could be gained only through activities other than building, such as road construction and mining. Once such experience had been gained, the crew would build and inhabit structures fabricated from lunar materials. No interim steps would occur.

The gradual approach entails the use of prefabricated, preassembled modules only in the initial outpost. Then an interim assembly facility would be constructed. This facility which would be first used to assemble imported modules, which would be shipped in pieces. With time, this assembly facility would be used to construct facilities partially and then almost totally from lunar materials.

The use of an interim assembly facility could enable gradual education in the use of lunar materials for construction. Participants felt this approach could enable the construction of intermediate living and working facilities comparable to those used in the Antarctic. The crew would not have to wait until the final stage of base development to build improved facilities. And by building intermediate facilities, the crew would have the opportunity to experiment with in situ materials for construction before building the final base facilities.

Participants considered incremental development to be more realistic than the discrete approach. They thought this gradual development would furnish much more experience with using and deploying lunar resources. The final goal of making full use of lunar resources could be achieved more efficiently and economically .

The incremental philosophy was thought to be highly critical to the logistics and planning of a lunar base. By extending terrestrial construction technology processes, participants thought that Technology Readiness Level 6 could be achieved in two to three years.

3.4 LUNAR BASE CONSTRUCTION SITE REQUIREMENTS

Four requirements were identified for developing the lunar base construction site. These were:

- logistics
- site layout
- foundation design and/or construction
- road serviceability and surface transportation system.

3.4.1 Construction Logistics

Participants considered logistics critical to the success of the lunar base. Logistics involves a construction schedule, a detailed manifest, and a resource list for the base. To develop the schedule, manifest, and resource list, it is

essential to understand design materials, site layout, facilities erection, and facilities placement.

To help define a construction schedule, participants identified three stages of initial base development. These were:

- •Construction site mobilization
- •Initial habitation facility or construction camp
- •Initial lab/work/storage facilities

In developing the schedule, manifest, and resource list, participants identified four logistics concerns. These were:

- •Positioning of equipment and materials prior to first crew's arrival
- •Identifying required facilities components and relative priorities
- •Defining design and assembly considerations of components
- •Defining equipment requirements.

Participants considered maximum simplification and streamlining of the first crew's activities critical. To achieve that, equipment and materials must be properly positioned prior to the first crew's arrival.

Identifying required facilities components and their priorities is also critical. One participant cited a cautionary example in which 85% of an order was delivered for a pipe layout project. That percentage consisted only of straight sections of pipe. Because there were no joints or elbows, the pipe could not be laid. Participants warned against repeating such experiences on the moon.

Design and assembly considerations of the components affect equipment design, delivery schedules, destinations, priorities, etc. For example, the size, shape and weight of a component determine some parameters of construction equipment. Component destination dictates unloading, placement, hauling, radiation shielding and others. Interfaces with other components must be considered for their proper planning and placement. These component considerations are involved with virtually all base facilities which are not entirely pre-fabricated and ready for use upon arrival.

Logistics considerations for equipment can influence design, delivery, task, optimization, etc. These considerations overlap the larger area of equipment design and are therefore discussed as part of Section 3.5.

The readiness level of lunar base construction logistics was believed to be at Technology Level 4. To achieve Level 6 for a specific construction simulation would require one year from the start time of the simulation design. The procedure would entail:

- •Identification of components for the simulation
- •Schedule and mobilization of transportation
- •Delivery and assembly of components.

More detailed planning would require more specific information.

Information sources for lunar base construction logistics exist in the form of terrestrial experiences in harsh and remote environments. Other sources of experience and/or solutions to logistics problems include NASA, contractors with equipment and construction experience, and schedule consultants. However, to test concepts of lunar logistics, participants felt a construction simulation is required.

TABLE 3.4.2-I.-LAYOUT CONSIDERATIONS

	dust	vibration	haulage	safety	module spacing	hazardous operations	flight trajectories
habitat	1	1	2	1	1	3	1
service facility	2	3	2	1	1	1	1
power plant	1	3	3	1	1	1	1
manufacturing plant	3	3	1	1	2	1	2
processing plant	3	3	1	1	2	1	2
mine site	3	3	1	1	1	1	3
space port	2	3	2	1	2	2	NA

3.4.2 Site Layout Considerations

Site layout is considered critical for lunar base success. Participants cited seven factors which must be assessed for site layout. They examined the impact of these factors for seven types of sites. Factors and sites are shown in Table 3.4.2-I. Rankings of impact are:

- 1-Important consideration
- 2-Secondary consideration
- 3-Insignificant consideration

No technology readiness level was stated for site layout considerations, but participants felt three to six years would be required to achieve Level 6. Many terrestrial technologies may offer experience and knowledge applicable to lunar base site layout. Conversely, because so many technologies are involved, participants felt much more research and development is needed. A principal need is to test site layout concepts in an Earth-based lunar simulation.

3.4.3 Foundation Design and/or Construction

Foundation design and/or construction is important for safety and cost-effective design. A geophysical profile of the base area must be obtained. Criteria must be defined, such as:

- Allowable differential settlement beneath the base nuclear reactor
- Appropriate explosives use
- Excavation requirements
- Dynamic and seismic loadings
- Required anchorages.

Knowledge of foundation design/construction for a lunar base is at Technology Readiness Level 1. An adaptable terrestrial technology does exist. However, to achieve Level 6, a large-scale simulation in an Earth-based lunar simulator is required. Participants felt that such a simulation could be completed three years earlier than the lunar base, if both were begun simultaneously. To accomplish the simulation will require the interdisciplinary coordination of foundation engineers, mechanical engineers, and structural engineers.

3.4.4 Road Serviceability and Selection of Surface Transport System

Participants identified six requirements for road serviceability and the selection of a surface transport system. They were:

- Defining the allowable extent of rutting, dust, washboarding
- Characterizing degree of required stabilization vs. maintenance of roads and launch/landing facilities

- Obtaining reliable knowledge of serviceability of natural lunar roads
- Evaluating the use of various mine tailings for road improvement
- Evaluating the location of the landing pad relative to habitats or mines to minimize road building
- Evaluating use of berm systems to reduce blast effects from the landing pad and thus minimize road building.

A portion of the strawman scenario states that blast effects extended one km for rugged equipment and 10 km for delicate equipment. This affects the extent of road building, because equipment must be located a sufficient distance from the landing pad to prevent blast effects. However, the equipment must also be accessible by means of roads. Participants considered the blast protection distances arbitrary. They suggested that methods other than distancing equipment from the launch pad be considered for blast protection. If other methods were used, it could reduce the required road-building.

Some suggested performance parameters requiring definition for road and surface transport systems were identified. They were:

- Wheel vehicle performance
- Tracked vehicle performance
- Walking vehicle performance
- Number of vehicle passes.

Adaptable bodies of design engineering experience and mobility models exist for the areas of road serviceability and surface transport. The technology readiness level was considered to be Level 1. As with foundation design/construction, a large-scale simulation in an Earth-based lunar simulator is required to achieve Level 6. This simulation could also be completed three years prior to the lunar base where both were begun simultaneously. This technology area is considered important for effective lunar base operation, but not critical to safety.

3.5 LUNAR BASE CONSTRUCTION OPERATIONS

Lunar base construction operations included the topics of equipment matched to tasks, mobile power, and worksite illumination. Before equipment can be matched to tasks, a design philosophy must be selected.

3.5.1 Equipment Matched to Tasks

Matching equipment to tasks was believed to be highly critical in order to develop efficient and reliable systems to sustain long-term missions. Participants felt that a well established terrestrial technology base exists for different environments which can be adapted to the lunar base. Three to six years were thought to be required before technology readiness would achieve Level 6.

Choosing an equipment design philosophy is important. Participants identified three dominant design dichotomies:

- One machine vs. many
- Dedicated vs. multi-tasked
- Detachable implements vs. permanently attached ones.

Participants identified seven types of tasks and many sub-tasks which must be properly matched to equipment for successful lunar base site development. These tasks, sub-tasks, and equipment are shown in Table 3.5.1-I.

3.5.2 Mobile Power

Participants considered the issue of mobile power to be the most critical

TABLE 3.5.1-I.-TASKS MATCHED TO EQUIPMENT

Task	Sub-task	Equipment
Exploration	Surface Mapping (site and local area)	•orbital surveying •site surveying •set datum benchmarks
	Subsurface Mapping (site and local area)	•seismic •radar •drilling - hard rock, soil, drive tube •sample acquisition and analysis
Soil Engagement (Machine-Soil Interfaces)	Soil Extraction	•drilling •digging •trenching •scraping, grading •micromechanisms •blasting •rock breaking
	Soil Transportation	•hauling (wheels, tracks, walkers, hybrids) •fixed track (rails, roads, cables, conveyors, pipeline) •trajectory (loose, bagged, "snow-blowing" - compacted or glazed
	Soil Deposition	•dumping •compaction •containerization (bagging) •backfilling (trenches, caves) •shielding
Assembly	Unload Lander	
	Rigging	
	Lifting	
	Lowering	
	Erection	
	Rotation	
	Pitching	
	Connecting	
	Fastening	
	Inspection and Quality Control	
	Pipelaying	
	Cablelaying	
	Building Construction	
	Deployment	
	Testing/Start-up	
Transportation (other than soil)	Bulk transportation	•tankage (fluids) •palletizing •bin/hopper •also see Soil Transportation
	Component transportation	•machinery •facility elements •modules
Foundation and Stabilization	Surface conditioning	•roads •landing pads •facility foundations

TABLE 3.5.1-I.-TASKS MATCHED TO EQUIPMENT (cont'd)

Task	Sub-task	Equipment
Foundation and Stabilization	Deep foundation	•pile driving •pier setting •anchoring
Equipment Maintenance, Service, and Repair	Diagnostic	
	Servicing	•replacement of wear elements (consumable) •fueling •lubrication •cleaning (dust)
	Replacement of major components	
	Storing and Inventorying	•parts •materials •consumables
	Development of repair equipment and tools	
	Wrecking	
	Building of service facility	
Building Element Manufacturing	Stone Cutting	
	Brick making (ceramic, cement, binder)	
	Sand-bagging	
	Sintering	
	Rubble masonry	
	Concrete elements (cement, aggregate)	
	Glass production (fiberglass)	
	Metallic elements	
Miscellaneous	Tunnelling	

and insurmountable issue impacting the design of this equipment. They felt that the one MWe power supply cited in the strawman scenario is too low for construction operations when compared to similar terrestrial operations. Realistic construction equipment power requirements must be established.

That power supply was, however, one of the basic assumptions for the workshop. Therefore, participants used it to define a set of required equipment performance parameters:

- •Minimum on-board power system mass
- •10 to 100 kWe for each machine for a to-be-determined (TBD) duty cycle
- •Establishment of techniques for refueling mobile equipment with cryogens in vacuum.

Adaptable terrestrial power technologies include:

- •fuel cells
- •rechargeable batteries
- •nuclear
- •beamed
- •wires•umbilical cord
- •heat sources (RTG and mass storage)
- •solar
- •mechanical (flywheel and springs).

At present, this technology area's readiness is Level 3. To achieve Level 6 will require at least six years of

vibration and environmental tests at estimated duty cycles in vacuum. Participants felt fuel cells, in particular, would require lengthy development to achieve operability in a dirty, noisy environment. Today, they can operate only in clean, vibrationless environments.

3.5.3 Worksite Illumination

For direct or indirect (electronic) viewing of a lunar worksite, participants felt that ideal, or at least acceptable, ambient lighting requirements must be established. In addition to human needs, supervisory remote control of construction machines will require appropriate illumination for viewing. This requirement could extend to pattern recognition systems as well.

Five performance parameters requiring definition were identified. They were:

- range of intensity
- diffusion
- contrast
- source angles
- color spectrum mix.

An adaptable terrestrial body of experience exists in the form of solar and electrical illumination. For solar, these are:

- lenses
- polished or deposition-formed bright reflectors
- diffusing reflectors
- diffusing filters.

From electrical illumination, these are:

- incandescent spotlights
- incandescent floodlights
- fluorescent bulbs.

Electrical systems are well developed except for bare elements that might be used in the lunar environment.

The technology readiness of lunar worksite illumination was believed to be a combination of Level 3 and Level 6. Participants estimated a period of two years to reach Level 6. A testbed should be established to simulate solar illumination at various angles for representative worksites. Experimental devices would be used to modify conditions for evaluation by human and electronic means.

3.6 LUNAR DESIGN MANUALS

Due to the lack of codified engineering information about the Moon, participants considered a series of engineering handbooks or design manuals necessary for developing lunar mining and construction technology. They considered equipment and building construction part of that technology. They felt that proceeding from first principles or raw data was inappropriate for engineering design tasks, and that handbooks or design manuals would provide better guidelines. Three types of design manuals were identified as requirements for developing lunar mining and construction technology. They were:

- Equipment and Machine Design Standards and Guidelines
- Lunar Design Code
- Regolith Behavior.

3.6.1 Equipment and Machine Design Standards and Guidelines

To design equipment and machinery for lunar mining and construction, participants considered it essential to establish a knowledge base of design practices, constraints, performance objectives, guidelines, and data. Regarding lunar practice, the current level of understanding for these engineering guidelines and standards is at Level 1. Participants made no estimate for time required to develop such guidelines. Terrestrial experience, though broad, is believed to have limited adaptability to lunar practice.

3.6.2 Lunar Design Code

The establishment of a lunar design or building code was considered essential for minimum standards of safety and mission success. This code would require information similar to that in the manual for equipment and machines. However, this material would be specific to building structures on the Moon. It would focus on civil structural design criteria, life support criteria, foundations and so forth. Participants wanted to be able to look up lunar structural footing requirements in such a code, just as they can on Earth.

Technology readiness level for building structures on the Moon is at Level 2. Further experimentation is required using lunar soils and simulants and new test methods. Paper simulations of lunar mining and construction project scenarios using the code are needed to test its completeness, accuracy and sensitivity.

Participants felt that NASA should aid development of the code along with universities, private contractors, Corps of Engineers, industry, etc. Terrestrial experience was believed to be appropriate only as a model.

3.6.3 Regolith Behavior

Establishing a generic descriptive model of the lunar regolith under both disturbed and undisturbed conditions was considered very critical by participants. This is because regolith behavior is at the root of the major issues of mining and construction.

Three classes of performance parameters are required for complete description of regolith behavior. These are:

- Parameters of in situ and disturbed behavior, such as strength, compressibility modulus, damping, etc
- Parameters describing soil interactions with machinery
- Parameters codifying acceptable environmental risks, such as slope stability, seismic conditions, dynamic loadings, etc.

The current level of technology readiness is Level 2. Participants felt that three years are needed to reach Level 6, if that is defined as the release of the regolith behavior model. Terrestrial building codes can serve as models. Lunar soils and simulants can be used to develop and test the model. Continued revisions will be needed.

Testing the regolith behavior model must occur simultaneously with tests of the lunar building code, since the building code depends upon the model. Paper/numerical predictive simulations of various lunar base scenarios are required for testing the soil model and building code.

3.7 CONSTRUCTION SITE DEVELOPMENT SESSION SUMMARY

Participants focused on the topic of construction site development challenges and problems. Thirteen challenges and problems were identified. The 13 were re-combined into five categories for discussion:

- general design principles
- lunar base development philosophy
- lunar base construction site requirements
- construction site operations
- lunar design manuals.

Throughout the discussion of construction site development challenges and problems, participants repeatedly stressed the

critical need for an earth-based simulator of the lunar environment. Such a simulator would be used to test engineering concepts, construction concepts and practices, models, performance, operations, and equipment.

Two general design principles for lunar construction site development were identified:
- •simplicity
- •learning from experience

Participants feared that cutting-edge technological solutions could be inappropriately favored over simple solutions. Participants felt that organizational, management, and personal experiences were as important to capture as technical knowledge.

Two lunar base development philosophies were discussed. Incremental, gradual development was selected in favor of discrete change.

Four requirements were identified for developing the lunar base construction site. These were:
- •logistics
- •site layout
- •foundation design and/or construction
- •road serviceability and surface transportation system.

Logistics was considered critical to base success. Foundation design and/or construction is important for safety and cost-effective design. Road serviceability and the selection of a surface transport system are important to lunar base success, but not critical to safety.

Lunar base construction operations included the topics of equipment matched to tasks, mobile power, and worksite illumination. Before equipment can be matched to tasks, a design philosophy must be selected. Seven task types and many sub-tasks were identified which must be matched to equipment. Power was considered the most critical and insurmountable issue impacting the design of construction equipment.

Three types of design manuals were identified as requirements for developing lunar mining and construction technology. These were:
- •Equipment and Machine Design Standards and Guidelines
- •Lunar Design Code
- •Regolith Behavior.

All three are required to establish guidelines for actual engineering practice on the moon.

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SECTION 4

Facilities Construction on the Moon: Issues and Problems

4.1 INTRODUCTION

The Facilities Construction Session (FCS) participants addressed the issues and problems associated with the central lunar base structures:

- the habitat
- movement and operations with large structures
- transport facilities such as roads and tunnels
- transport facility building equipment
- base power supplies
- weight constraints for all facilities and equipment.

Base power supplies and weight constraints are discussed in the context of the other items.

Though there was some overlap in topics with the Construction Site Development Session (CSDS), the emphases of the two groups were different. CSDS primarily addressed the precursor activities which will lead to the establishment of the facilities discussed in the FCS.

4.2 HABITATS

4.2.1 Above Ground Habitats

Space station modular habitats and inflatable habitats were the two types of living facilities participants discussed. Modules would be used in the early base development phase with large inflatable structures coming on line at an undetermined later time. Once the facilities have been landed on the Moon, primary concerns are placement relative to other base facilities and protection from radiation.

The space station habitat modules would be used initially because they will be available "off-the-shelf." Participants considered many aspects of space station technology to be adaptable to the lunar base. Some participants thought it would be advantageous to adapt space station technology in the early phases of the base rather than building new facilities. Crew time, a precious resource, should not be used to build and outfit new facilities. New facility construction is expected later in base development, when the crew is larger.

Participants considered oxygen extraction to be a primary siting consideration for the lunar base, so habitat modules should be placed close to any existing ore deposits. Power sources will then be situated between the two locations. However, there will be a trade-off between the richness of the ore deposit and the local topography. Participants felt that establishing the base in a fairly flat area was desirable so that minimum regolith need be moved.

Prior to the placement of base facilities, participants considered a precursor mission desirable. This mission could characterize all the local topographic features. It should also be capable of characterizing local subsurface regions so the existence of any hard deposits, boulders, or voids could be determined.

To place the module, it can be placed directly on the regolith, with struts to support it and keep it from rolling over. It was felt that if the site were smooth and level, the bearing pressure of the module on the regolith would be very small, "not even as great as an astronaut's boot." Another method would be to dig an excavation 25-30% of the module's diameter, lower the module into it,

and then fill around and over the module with the excavated regolith to cover it.

Participants identified several problems with the excavation method:

- an excavation will require considerable power, equipment, instrumentation, delivery weight to surface, and crew EVA time;
- soil may sag or shift, unbalancing and rotating the module so the floors are no longer level;
- hydrogen embrittlement of the aluminum in the skin from the hydrogen in the regolith;
- elimination of waste heat, because regolith is an excellent thermal insulator.

To counteract the sagging problem, it was suggested that the excavation be cut well below the desired level and then backfilled so that the module's regolith foundation would be well known. The problem with hydrogen embrittlement from the regolith was considered surmountable.

Once the module is in place, it will need protection from radiation. Participants identified two methods:

- covering the module skin directly with regolith;
- constructing a shelter around the module and covering it with regolith, either as loose deposits or as sandbags.

Figure 4.2.1-1a illustrates covering the module skin directly with regolith. One problem with this method is that it prevents inspection of the skin and any systems features which reside there. A second problem is that the lunar crew could have to displace a large amount of regolith.

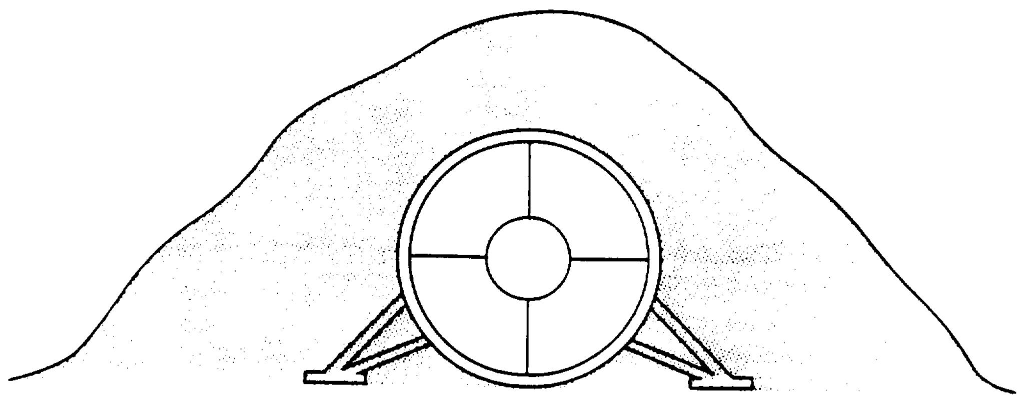
An alternative is to build an offset frame structure around the module and load regolith onto that, as shown in Figure 4.2.1-1b. This would create a vault external to the module. Regolith could then be used to cover the offset structure and, because the regolith cover would reside at its natural angle of repose on the offset structure, the cover would be stable.

A second alternative involves the use of an offset structure with nearly vertical shelves. This alternative is illustrated in Figure 4.2.1-1c. Regolith could be piled between the shelves, which would be high enough so that the regolith would remain at its natural angle of repose. The virtue of this type of offset structure is that the regolith mass which must be displaced is less than that for either the plain offset structure or no offset structure.

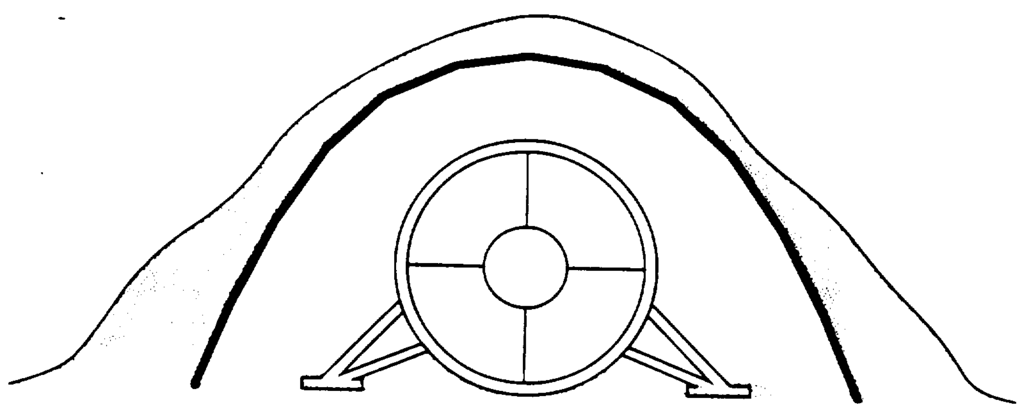
A third alternative is a two-ply type of offset structure in which the two skins form concentric hemi-cylindrical shells around the habitat module. The space between the shells is filled with regolith. This type of structure requires the least regolith displacement. However, participants estimated that all the offset structures would have approximately equal masses.

In building a habitat shield structure which is not a continuous surface, great care must be taken to prevent pathways through which radiation can leak. Even very tiny leaks can reduce shielding efficiency considerably.

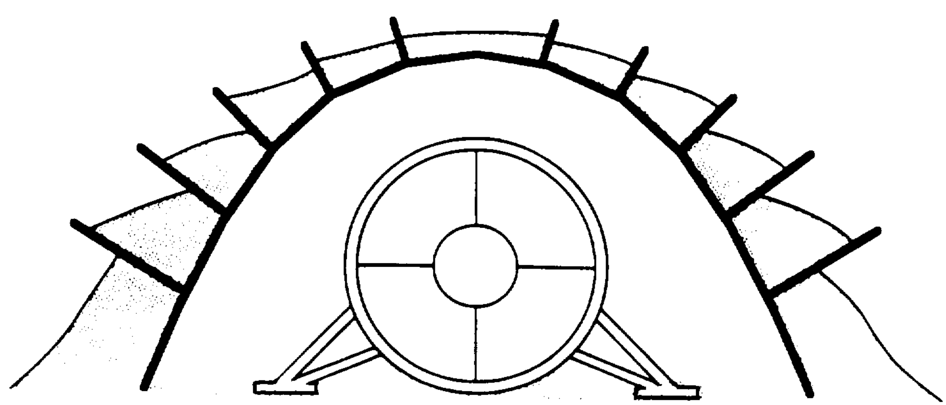
A second consideration in radiation protection is the depth of the regolith coverage. One participant noted that two meters of regolith has been estimated as necessary to bring radiation exposure to the level considered acceptable for the American population. Participants felt that level of safety to be excessive for the lunar crew. They estimated that one half meter of regolith shielding provides a shielding



a-Regolith Blanket Around Hab Module



b-Offset Structure With Regolith Blanket



c-Offset Shelved Structure With Regolith At Natural Angle of Repose

Figure 4.2.1-1.-Module Coverage Methods

equivalent of 70 g/cm^2. For comparison, the Mars exploration vehicle radiation shielding level was estimated at 20 to 30 g/cm^2. Therefore, participants thoughts one half meter of regolith would provide four times the shielding available to the Mars exploration crew for twice the shield thickness. The lunar crew is also expected to reside in the lunar module for six months, compared to 15 months on Mars. Participants felt that one-half meter of regolith coverage represented significantly less construction work than two meters. They identified two indicators of acceptable coverage:

- •crew effort needed to displace the necessary regolith;
- •level of effort relative to other duties.

Participants noted that although blankets of regolith could solve the radiation exposure problem, they create a new one: connecting new modules to the covered ones. The space station module docking mechanism is designed as a high pressure bellows, but it only moves a few centimeters. Therefore, the modules themselves must be moved together; the docking mechanism is inflexible. If the module is covered with loose regolith, the regolith must be removed from the docking connector, leaving it covered with dust. Sandbags might relieve the dust problem, but they still require effort to move. If the module is seated in an excavation, then in addition to removing the regolith cover, the module must also be re-excavated.

Two means of handling this concern were identified:

- •place the module in a partial excavation, but surround it with an offset structure laden with regolith;
- •cover the connectors with dust caps and only excavate a portion of the regolith which covers them.

In the second case, a retaining wall normal to the long axis of the module would hold the remaining regolith in place. The connectors themselves would also need radiation protection.

Inflatable membrane structures offer alternatives to pre-fabricated modules. They entail the same concerns with placement and shielding as prefabricated modules do. However, they have other advantages and disadvantages which must be considered.

Their primary advantage is their excellent weight to volume ratio. However, they are difficult to build, and participants considered them to be new, untested technology.

Five steps were identified for inflatables construction:

- •unstowing and deploying;
- •seating in an excavated area;
- •inflating;
- •outfitting with multiple floors and other internal structures, life support systems, and other systems;
- •covering with regolith for radiation protection (may occur before internal outfitting to protect workers).

The representative NASA inflatable design, shown in Figure 4.2.1-2, uses a central column for support and access to multiple levels.

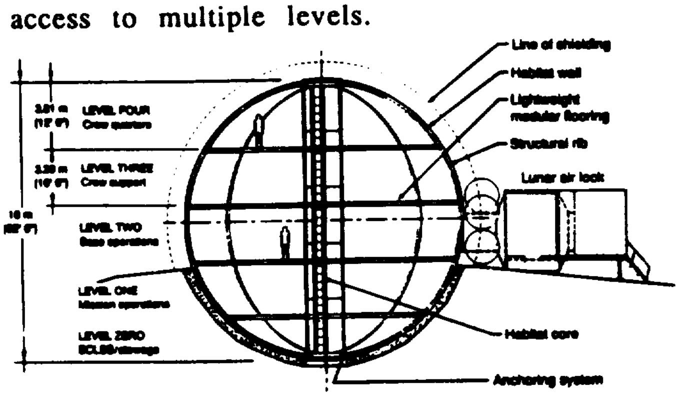


Figure 4.2.1-2.-Inflatable Habitat Structu:

Air pressure was considered by participants to be a central problem for the use of inflatables on the Moon. Pressure differentials across a membrane of one psi are unachievable with today's inflatable structures. Participants therefore felt that an internal operating pressure of 14.7 psi for lunar inflatables is unrealistic. Air pressures such as 8 psi (Tibet or Chile) or 10 psi (Denver) were considered more technologically achievable and essential for building optimum structures.

Participants considered leakage of gasses through an inflatable membrane to be of concern, although they felt that losses through airlocks would significantly outweigh leakage. They identified several factors affecting leakage:

- material
- thickness
- surface area
- temperature
- folding of material
- seam soundness.

The larger the inflatable's surface area, the greater the leakage. Although participants felt that material thickness could be improved, they considered seams to be most important. They were also concerned about embrittlement of the material through outgassing to the vacuum.

FCS participants developed several approaches to controlling leakage across a membrane. These were:

- coating the membrane with a plastic material;
- covering the inflatable with regolith;
- using inflatables to form cast structures.

Coating the membrane with plastic would render it impervious to leakage and outgassing. Covering the inflatable with regolith would help solve the internal-external force balance problem. To cast structures, two membranes would be inflated with a space between them. A foam or to-be-determined "low temperature material" would be injected into the space for molding into the desired shape. The membranes would be peeled off for re-use.

Two primary disadvantages to inflatables were identified:

- complexity of construction
- lack of knowledge.

Participants felt that inflatables construction was too complex for early base phases and possibly for later phases. Therefore, many participants felt that importing pre-fabricated modules from Earth was the best solution for habitats. Participants also felt that insufficient knowledge exists for inflatables to be exclusively relied upon; backups must be provided.

Participants suggested two alternative uses for inflatables. They were:

- unpressurized inflatable tents which would shield pressurized modules or shelter equipment
- intermediate facilities stage between the earliest space station-type modules placed on the surface and later, possible below-ground structures.

The intermediate phase is seen as the time to experiment. Terrestrial testing beforehand is considered essential.

Participants concluded that above-ground construction was too dangerous and required too much EVA for humans to do it Therefore, robots would probably do most of above-ground construction, with humans present to troubleshoot. A semi-autonomous construction system is necessary for the base to be built by 2010 rather than after 2025. The semi-autonomous system must be very simple to preclude extensive

human supervision. The system design concept should be based upon all-human construction and then adapted to automation.

4.2.2 Below-Ground Habitats

Underground habitation on the Moon falls into two categories. They are:

- •adapting natural phenomena
- •constructing artificial tunnels.

Lava tubes, if they exist on the Moon, may form natural cavities which could be adaptable to human requirements. Artificial construction consists of tunneling and lining or sealing. Participants considered underground habitation superior to above-ground for reasons of safety, and they felt the crew should move underground as soon as possible.

If lava tubes exist on the Moon, the roofs could be several meters thick, providing natural radiation protection. These tubes could be lined and then pressurized to form living space. Figure 4.2.2-1 shows a lava tube being prepared for habitation with an inflatable structure.

Because lava tubes are generated by lava flowing downhill very quickly, participants thought they might have very steep slopes of 13-15% and thus be unsuitable for habitation. However, participants felt that even with such steep slopes, the tubes could still be useful for storage of waste products or for growing food. In that case, the slope would not be as critical. Another method of handling the slope problem would be to construct different levels, like canal locks. There would still be many steps with multiple levels. Participants feared that lava tubes may have collapsed roofs which have left floors so littered with debris that the tubes are essentially unusable.

To build large, habitable tunnels under the lunar surface, participants considered a large-scale tunnel-boring machine necessary. The machine would begin its drilling about midway up the wall of a crater, using the crater as a repository for the debris generated from the mining. The tunnels would then be filled with inflatable liners or sealed by other methods. They identified four additional ways to seal the tunnels:

- •glazing
- •plasma deposition
- •thin metal foil
- •conventional materials.

Airlocks would be positioned throughout the tunnels to prevent single points of failure.

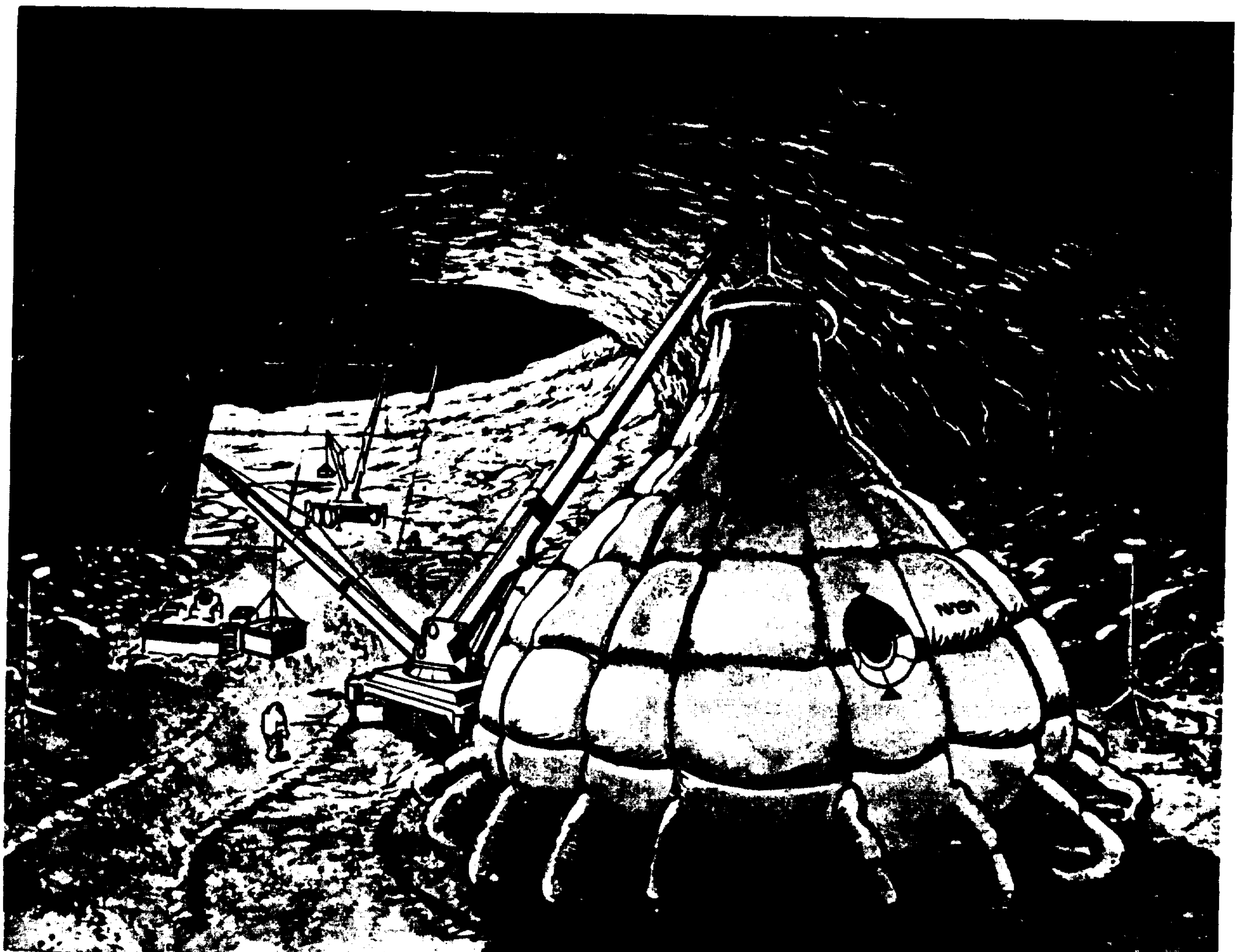
Participants envisioned two tunnel configurations:

- •large circles concentric to the crater with connecting "spokes"
- •one large downward spiral.

The connecting tunnels need not be as large as the living areas, so they could be drilled by boring machines with smaller drillheads.

Another method of building tunnels was though the use of an experimental vehicle called the "subselene." This machine, introduced in a plenary session talk by Joseph Neudecker of Los Alamos National Laboratory (LANL), is a nuclear-powered rock-melting device. It was developed by LANL and has been demonstrated on a small scale. According to Neudecker, this device melts rock at the rate of about one-half meter per hour, extruding the rock as a glassy melt which would be used to line the tunnels.

The subselene has several advantages for building tunnels. It is indifferent as to material, melting through basalt or regolith with equal ease. It can achieve directional melting either



through unequal heating or by making the gripper pads which aim the beam directional. It has very attractive ratios of material melted per hour of work. Its operations are completely remote and automatic. The shape of the penetrating face determines the shape of the tunnel, so that flat floors would be simple to achieve with a horseshoe-shaped face. The byproducts need not all be used to line the tunnels; they can be fabricated into different forms for many uses. Gases will also be released which can be captured and put to use.

However, the subselene also has drawbacks. The most serious drawback is its tremendous power requirements. A five-meter diameter lunar tunneler would require 134 individual rock melting heaters, each of which in turn requires three MW of thermal power. This power would be provided by a liquid metal heat pipe which would be connected to a nuclear fission reactor situated immediately behind the melting head of the tunneler. Total thermal power requirements were estimated at approximately 400 MW for a five-meter-diameter lunar tunneler and 150 MW for a three-meter-diameter tunneler. Those estimates were based on a very fast advance rate of 80 meters per day.

A second drawback to the subselene is thermal balance. So much waste heat would need elimination that participants considered this to be the limiting feature on the size and advance rate of the machine.

Finally, the subselene is both heavy and costly. Each subselene device was estimated to weigh 320 t, and the unit development cost was estimated at $50 million. If weight-to-orbit costs of $3,000 per pound are used to factor the expense of lofting the subselene into space, then it could cost over two billion dollars to get one machine to the Moon. A summary of Joseph Neudecker's plenary talk on the subselene can be found in Appendix D.

Participants felt that tunneling was very desirable for later stages of base development. Besides creating living space, tunnelers and rock melters can also mine. They are highly compatible with automation and robotics. If a subselene is used, needed tunnel shapes could be melted as desired.

4.3 EQUIPMENT FOR MOVING

Unloading, moving, and emplacing habitat modules and other large structures will require special equipment. Participants discussed four types:

- cranes
- gin poles
- elevator platforms
- Boeing"straddler."

As discussed by Brent Sherwood of Boeing in the FCS, the Boeing straddler design has telescoping legs. Therefore, it can move off the lander's cargo platform slowly, while the platform remains level. It can then walk over the lander and remove payloads, even picking up the lander itself in the event the lander became disabled. Participants considered cranes and other devices requiring ramps to be unsuitable unloaders, because the lander's cargo bay is about nine meters from the surface. They also felt that elevating cargo platforms would be unsuitable because the platform would then require a large hole for clearance for the rocket engines. Participants felt that moving cargo around rocket engines would be unsafe in any event. Finally, a gin pole arrangement could well have problems with anchorage.

Once the habitat is successfully unloaded, it must be properly situated. As discussed in Section 4.2.1,

the habitat can either be placed directly on the regolith or in an excavation. Excavations can be performed by human crewmembers or by a machine. Size, depth, and difficulty of excavation will determine which method is used. If an excavating machine is required, it must be available either before the habitat is delivered or on the same cargo flight.

Participants felt that the unloading device will probably also move the habitat module to its resting place. For the initial module, this poses no problem. However, when additional modules are added, they must be lined up precisely by the machine to permit connection with established modules. According to participants, this will require a machine capable of precision manipulation as well as brute force. The same machine will also move and place other large structures such as:

- power storage devices
- units of the nuclear power plant
- scaffolding for photovoltaic array placement.

Other types of equipment will be required for facilities construction. Participants felt that early equipment should be general purpose, and later equipment could be more specialized. They decided that a limited crew with specific construction objectives will need equipment for lifting, moving, and digging. Those tools would probably be electric, assuming the power supply was available. Participants envisioned the necessary equipment to include:

- snow-blower
- snap-together gin pole
- block and tackle
- ladders
- mobility vehicles
- hand tools.

4.4 TRANSPORTATION FACILITIES

Participants noted that there are several requirements for roads. Roads are necessary because they assist movement from point to point. They provide a level, smooth surface for machinery to use, and they also inhibit the nuisance of dust. Dust will be even more of a problem on the Moon than it is on Earth because of its electrostatically "sticky" tendencies. Participants felt that every step should be taken to prevent dust from getting onto viewing surfaces, into moving parts, or on mating surfaces.

Although some concepts of lunar roads show them to be elevated above the surface, FCS participants felt that scraping off the top few centimeters of fluffy regolith to expose the hard, compacted regolith would make more sense. This would produce depressed roads rather than elevated ones.

Participants identified three problems for road-building. They were:

- boulders
- surface decompaction
- dust.

Participants considered explosives the simplest solution to boulder removal. Regarding surface decompaction, some participants felt that driving over the compacted regolith would further compact it. Other participants felt the reverse was true. To solve the dust and decompaction problem, participants suggested covering the roads with gravel in the earliest phases of base development.

The source of the gravel was expected to be tailings from the oxygen mine beneficiation process. For a 100-ton per year oxygen-production rate, participants estimated that 140 tons of gravel would result each year. Participants thought that

consolidated paving materials and paving machines would be available to stabilize the road surface in later stages of base development.

Participants proposed a rail system as another means of transportation, where very heavy traffic exists. They felt this would avoid excavation requirements and the problems of degradation and dust. A problem with rail is anchoring it to the regolith, since terrestrial ties would be too difficult to transport to the Moon. Because terrestrial rails could not be transported either, participants suggested manufacturing them from basaltic glass by a rock melting mechanism. A mechanism which melted regolith into glass could be used on the surface. A problem which participants raised but did not resolve was equipping machinery to ride rails and roads.

Tunnels were the third transportation proposal. Because they would be built by the same methods described above in Section 4.2.2, paticipants felt they would appear only in the very mature stages of base development.

4.5 POWER

Initial base power sources would be regenerable fuel cells and photovoltaic arrays (PVA's) from space station technology. As the base grows, nuclear power would replace PVA's, although they would continue to be used for habitat emergency backup.

Four problems with PVA's were identified. They were:

- brittleness
- large, massive support structure
- low efficiencies
- dust adhesion.

New technology may resolve the problem of brittleness. Current PVA's were described as very brittle, single-layer gallium arsenide crystals, with nominal end-of-life efficiencies of about 15% at approximately 100°C. A new type of PVA material could serve as an alternative to the current versions. It was described as so flexible that it could be rolled up like a sleeping bag. It is a non-crystalline photovoltaic amorphous silicon which is deposited on a stainless steel substrate. Its advantages are low weight and easy stowage. However, it is only 5% efficient.

Three solutions to the dust adhesion problem were proposed:

- "record-cleaner"
- squeegee
- towers.

A robot would move an electrostatic device similar to that used for cleaning records over the array surface and pull the dust off. Squeegees could work, but were considered unsuitable because:

- expendable materials from Earth would be required
- it could scratch the arrays.

Mounting the arrays on towers could raise them above the dust. However, participants were uncertain about the height lunar dust will travel when disturbed. Therefore, they did not propose specific tower heights.

Power for the oxygen plant must be provided by the nuclear generator. Participants anticipated that the mine will require about 90% of the base reactor's generated power. The remaining power will be distributed to the habitat, laboratories, and equipment. If a rock-melting or tunneling device is eventually added to the base, another reactor will be required.

Participants expected power to be distributed by surface or buried cable. Possible power distribution problems were identified as follows:

- requirement for invertors and conversion to AC to transmit
- temperature problems with cable material if they are laid on the lunar surface.

4.6 FACILITIES CONSTRUCTION SESSION SUMMARY

Six topics concerned the FCS. They were:

- habitats
- movement and operations with large structures
- transportation facilities
- transportation facility construction equipment
- base power supplies
- weight constraints.

Power supplies and weight constraints were discussed primarily in the context of the other items.

Participants identified pre-fabricated modules and inflatable structures as the two above-ground habitat alternatives. Pre-fabricated modules were considered superior to inflatable structures for the initial base. Participants felt that inflatables should be used only in later base stages.

Participants felt that above-ground construction is too dangerous to be done by humans. It should be done by robots, via a simple, semi-autonomous construction system.

Below-ground habitats were considered much safer for crew habitation than above-ground. However, the complexity of their construction renders early occupancy unlikely.

Equipment which unloads large structures must also be able to move and emplace them. Such equipment must combine the characteristics of brute force and delicate precision manipulation. Other types of construction equipment needed in early base phases should be non-specialized.

Roads, rails, and tunnels were proposed transportation system alternatives. Road-building could be accomplished most simply be scraping away the top few centimeters of regolith, exposing the hard layer beneath. Rails are an alternative to roads where very heavy traffic exists. Tunnels require large machines to build. Because they are difficult and expensive to construct, they are expected to be used only in later base phases.

PVA's are expected to be the initial base power source. Nuclear power will be required for oxygen mining and processing operations. Power will be distributed by cable.

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SECTION 5

Conclusions and Recommendations

5.1 MINING SESSION

Mining session conclusions were:

- mining requirements for 150 t/yr oxygen production are achievable and comparable to a small terrestrial sand and gravel operation
- hydrogen extraction information is insufficient to determine its feasibility
- one MWe nuclear power is sufficient for oxygen extraction and equipment operations.

Recommendations were:

- operate during daytime to avoid equipment metal embrittlement
- use explosives to displace compacted regolith
- increase research in automation and robotics
- use conventional power distribution systems.

Further research in the following technologies was recommended:

- processes for oxygen production with related products
- automation in mining systems
- materials with high strength to weight ratios
- operations research:
    - mining/processing integration
    - modeling/simulation of mine operations (to allow trade studies)
    - scheduling
- remote sensing for site selection, characterization, navigation
- energy transmission and distribution to remote vehicles
- large scale Earth-based lunar simulator (especially for processing)
- bootstrapping and field modification (initial development and deployment), i.e. development of the very first stage equipment, parallel systems, and handling of equipment to the point where equipment will be dedicated
- EVA and space transportation
- other space manufacturing engineering pertinent to production of resources on the moon for use in space, such as large pressure vessels, heat shields, silicon chips, aerobrakes
- thermal control system for dusty environments, i.e., in vehicles
- thermal management in feedstock handling, including avoidance of waste heat and preprocessing of feedstock so that it is at the right temperature when it gets to the processing plant
- telesciences and computer-mediated control systems
- space safety: protection of crew from unusual hazards when they go EVA to repair equipment.

5.2 CONSTRUCTION SITE DEVELOPMENT

Construction Site Development session conclusions were:

- logistics is critical to base success
- foundation design and/or construction is important for safety and cost-effectiveness
- road serviceability and surface transport system selection are important for base success, but not critical to safety
- before equipment can be matched to tasks, a design philosophy must be selected
- power is the most critical issue affecting construction equipment design.

Recommendations were:

- use simple solutions to problems
- use and learn from experience: technicla, organizational, management, and personal
- make changes gradually, not suddenly
- develop design manuals to establish guidelines for lunar engineering practice
- establish a large-scale terrestrial simulator of the lunar environment to test engineering concepts, construction concepts and practices, models, performance, operations, and equipment
- develop realistic power requirements base on terrestrial experience
- make lunar samples available for engineering tests
- create a lunar sample engineering experiment analysis team
- form a consortium to support NASA, consisting of industry, universities, Bureau of Mines, Corps of Engineers, private contractors, etc.
- use existing technology bases to choose and develop equiment for the lunar mission, including civil engineering, construction, hazardous waste, mining, nuclear, automation and robotics
- provide timely notice of NASA documents concerning exploration missions, and establish and circulate a distribution matrix so that involved individuals can be aware of available documents
- achieve wide and easy access to NASA databases
- use resources and linkages which the workshop participants identified

5.3 FACILITIES CONSTRUCTION SESSION

The Facilities Construction Session conclusions were:

- pre-fabricated modules are superior to inflatables for early lunar base stages
- above-ground construction is too dangerous for humans and should be performed by semi-autonomous robotic systems
- below-ground habitats are superior to above-ground habitats, but are unlikely to be developed early due to the complexity of construction
- equipment which unloads large structures must combine brute force and precision manipulation to emplace the structures
- roads can be built most easily by scraping off the top few centimeters to expose the underlying hard regolith
- rails are alternatives to roads in cases of heavy traffic
- tunnels are difficult and expensive to construct and should be used only in later base stages.

Recommendations were:

- inflatables should be used only in later base stages
- early base phase equipment should be non-specific
- distribute power to equipment by cable.

APPENDIX A

LINKAGES AND RESOURCES

One of the tasks assigned to the workshop was identification of sources of expertise for handling the problems brought up in the sessions. Those linkages and resources are listed below by session.

MINING SESSION

Discussion of interdisciplinary functions and technologies pertinent to lunar mining produced the following table. It is based on an earlier list developed by Carlos Moreno.

TABLE A-I.-INTERDISCIPLINARY MINING FUNCTIONS AND TECHNOLOGIES

DISCIPLINES	APPLICATIONS	ISSUES
Structures Maintenance	Construction architectures habitats, labs	Strength, Mass reliability
Materials sciences	Use of lunar rocks to provide improvements at lower Mars to lunar delivery, or Earth materials required to solve lunar problems	Strength/weight thermal expansion properties, radiation shielding and protection, dust adherence (electrostatic charge), lubrication
Robotics/software	Rovers, less EVA	Position/Force Control, AI, expert systems
Telecom/navigation and information management	Other operations monitoring	Sensing, site selection, data interpretation & processing, perception, path planning
Thermal management	Other operations equipment (power, processing, underground habitats)	Environmental Extremes, thermal control system

TABLE A-I.-CONT'D

DISCIPLINES	APPLICATIONS	ISSUES
Energy Management	Other operations, labs, habitats	Power requirements, transmission & storage
Human Factors	Other operations & activities	Command interface, productivity, and psychology
Life Support	EVA support, maintenance	Command interface (video/ alarm system), repair & maintenance
Systems Integration & operations research	Interfaces with community	Vehicle mobility, ISRU plant interface, power plant interface, base construction requirements, navigation aids and beacons, radiation hardening, total base efficiency & productivity, total lunar base reliability, self-sufficiency, resistance to hard environment
Sensors	Other base facilities	Conflict in information management, broad disciplinary applications
Manufacturing Energy	Other Lunar or space applications	Metals processes in lunar environment, also selection of mining systems, synergistic processes

TABLE A-I.-CONT'D

DISCIPLINES	APPLICATIONS	ISSUES
Safety in Operations	Integrated Base Operations	Traffic, radiation shelter, power system operations, maintenance
Transportation Off Surface	Integrated into other areas	

TABLE A-II.-MINING SESSION LINKAGES

DISCIPLINES	SOURCES	LINKAGE
Systems Integration	California State Institute, UC San Diego	
	Rockwell	small scale rocket components
	Defense contractors	
	Large Scale Program Institute	system integration work for JSC
	SBIRs	NSF/NASA/DOE/DOD programs
	JPL	Pathfinder
	NASA research & academic centers	Current NASA support
Automation	Canadians	
	ODETICS (So. Cal)	SBIR (nuclear power plant)
	JPL/ARC	Pathfinder
	SAIC	NASA support
	DARPA	Martin Marietta
	Caterpillar, other mining manufacturers	
	CMU, MIT, Ohio State, WA State	university programs
	US Navy	
Telecommunications	UC Berkeley (computing)	
	NASA labs	Pathfinder/space station
	CMU Mining robotics	
	AT&T Labs	
	SDI contractors	
	DARPA Martin Marietta	
	Technoledge (Palo Alto)	

TABLE A-II.-CONT'D

DISCIPLINES	SOURCES	LINKAGE
Manufacturing	U of AZ (Center for Excellence) Bureau of Mines Carbotek, Coors Ceramics MIT, Battelle, LANL, JPL, A.D. Little	
Materials	U of AZ (ceramics & copper) U of TX SDI, LANL, LeRC Defense & Aerospace contractors U of ILL & Cal. Space Institute	
Thermal Management	JPL, LeRC, DOE labs, Hughes, Martin Marietta, Bechtel, Lockheed, Rockwell, SDI contractors U of WA	
Human Factors	Bureau of Mines, ARC, NRC, DOE contractors, MIT, USAF School of Medicine, Antarctica Research (NSF), submarine research	
Structures	Aerospace companies, mining equipment manufacturers	
Energy Management	LeRC, Nuclear power industry, Hughes, Bechtel, SDI, Sandia	
Life Support	JSC/other NASA centers, Biosphere Project	

CONSTRUCTION SITE DEVELOPMENT SESSION

TABLE A-III.-CONSTRUCTION SITE DEVELOPMENT LINKAGES

TECHNOLOGY AREA	LINKAGE
lunar worksite illumination	General Electric Co.
mobile power	NASA Lewis Research Center, Karl Faymon
construction logistics	NASA, contractors with equipment & construction experience, schedule consultants
construction site development power requirements	Solar Engineering Research Initiative, industrial associations
behavior of regolith	WES/COE, ASTM, ASCE, NASA, NAFAC (DM-7), model code agencies
foundation design & construction	same as for regolith behavior
road serviceability & selection of surface transport system	same as for regolith behavior plus AASHTO in Los Alamos (microwave sintering)

FACILITIES CONSTRUCTION SESSION

TABLE A-IV.-FACILITIES CONSTRUCTION SESSION LINKAGES

TECHNOLOGY AREA	LINKAGE
Deliverable modules	Boeing, prime aerospace contractor community, Marshall Space Flight Center
Inflatable structures	Mike Roberts, NASA Jack Boyt, Precision Air Structures Geiger Associates, New York Bird Air, Dr. Dante Bini, Dr. Milburne (Boulder)
Fabric manufacture & membranes	Seaman Corporation Many German, Japanese producers and one in Finland, Alan Hirasuna, Los Angeles for high strength fabrics: Burlington & Dupont
Construction equipment	Eagle Engineering Caterpillar, John Deere
Automation & robotics	Red Zone Robotics Jim Brazell, Georgia Tech DOE, Oak Ridge Laboratories Goddard Spaceflight Center Marshall Spaceflight Center Langley Research Center JPL
Sub-sea drilling & remote undersea vehicles	Bechtel, Carnegie-Mellon Univ., Stanford, MIT
Mining & tunneling equipment	Robbins Co., Java, Wirth Co. in Germany, Los Alamos National Labs on rock melter

TABLE A-IV.-CONT'D

TECHNOLOGY AREA	LINKAGE
Power-Nuclear Reactors	Bechtel advanced power division, Lewis Research Center, Westinghouse
Materials Consolidation	Construction Services Lab in Chicago Battelle Group (Lisa McCauley)
Concretes	Arnold Wilson, BYU T.D. Lin, Portland Cement
Lunar Environment	Dave Carrier, Bromwell & Carrier Al Binder, Lockheed Harrison Schmitt, Camus NASA online data base
Production Plant	Space 88

APPENDIX B

GOALS FOR THE NEXT WORKSHOP

Another task the workshop attendees were asked to perform was an identification of goals for the next workshop. Each session generated such a series of goals, which are listed below.

MINING SESSION

Topics for Other Workshops

1. Heat Effects on Equipment
2. Workshop on Mining Mars and the Asteroids
3. Far Planet Mining
4. Underground Mining
5. Power and Distribution

Workshop Goals

1. Bring practical mining experience to bear on concept development.
2. Provide the mining community with insight on what NASA is doing and plans to do.
3. Transfer knowledge and contacts.
4. Assist in program formulation.

CONSTRUCTION SITE DEVELOPMENT SESSION

Future Workshop Goals

1. Develop a simulation and gaming model for interaction and feedback at the workshop.
2. Provide participants with significant resource materials in time for review prior to the next workshop(s). The next workshop(s) will serve as kickoffs for task force-type efforts.
3. Address the construction effort to support the staged development of a lunar base.
4. Identify specific construction technologies necessary for staged development of a lunar base.
5. There is a need for a series of workshops focused on developing specific technologies such as soil design manual, building code, etc.

FACILITIES CONSTRUCTION SESSION

Future Workshop Goals

1. Keep the same people as the core of the next workshop, but consider inviting new people who have areas of specific expertise.
2. Increase opportunities for informal exchange of material among workshop sessions, both in terms of physical location of sessions and time made available for conversation and discussion. Also, perhaps session members could be mixed each day to achieve even more diversity.
3. Increase formal information exchange between groups through additional, short plenary meetings in which session status reports are given.
4. Time next workshop to be held approximately nine months to one year from the first workshop. Coordinate with 1990 fiscal year planning by Mission Analysis and Systems Engineering (MASE) and Planet Surface Systems.
5. Institute a newsletter or round-robin mailing arrangement for workshop attendees to keep them informed of each other's relevant activities.
6. Focus more on specifics such as base development, site layout, power systems, etc. This workshop was felt to be general, and the next one needs to be more specific.
7. Group size should be no larger than 12 and no smaller than six for good dynamics and diversity.

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APPENDIX C

WORKSHOP STRAWMAN SCENARIOS

FROM SECTION 5 OF

PLANETARY SURFACE SYSTEMS

REQUIREMENTS DOCUMENT (PSSRD)

5.4.5. Lunar Resource Base

The Lunar Resource Base Architecture is designed to focus on issues involved in safely and reliably operating an advanced lunar base that makes substantial use of in situ resources. The goal is to identify and to characterize those elements and architectures that enable and evolve long-term, high-leverage, permanent facilities on the moon. The baseline for FY89 represents a substantial infrastructure for construction and ISRU production.

5.4.5.1. Overview Summary

Mission Objective: Develop a substantial safe and reliable infrastructure by leveraging lunar resources for propellant, base supplies, and construction.

Surface System Mission: Construct, operate, maintain, and promote base infrastructure.

Top-level Functional Requirements

A Construct and operate facilities that utilize in situ resources to produce propellants, base supplies, and construction materials.

B. Service and maintain surface based landers.

C. Provide housing and support for crew.

D. Expand base infrastructure using in situ resources.

Assumptions and Guidelines

A. Assure crew health and safety and reliable mission operations.

B. Nearside site at 0° latitude, 24° east longitude (in southern Mare Tranquillitatis, just north of the crater Molke).

C. Ignore user operations and accommodations during initial studies.

D. All facilities and equipment are considered permanent with 30 year life, unless further analysis indicates otherwise.

E. Optimize use of in situ resources with constraints for construction and operations

F. Space Transportation can deliver 20 t of cargo to surface. Latitude effects are not considered in initial analysis.

G. Flight rate is once a year for cargo and twice a year for crew.

H. Except for data and programmatic concerns, precursor exploration missions are ignored in initial analyses.

I. Build up approaches need only consider mass and logical sequencing during initial analyses. More detailed concerns such as man-power, volume, etc, may be postponed to subsequent analyses.

J. Vehicles land within 25 m foot print with enough accuracy for a 50 m pad. Blast effects extend 1 km for rugged equipment and 10 km for delicate equipment.

5.4.5.2. Description

Figure 5.4.5-1 depicts the initial baseline for the key elements and their layout. Activity is long-term and at a relatively high level; facilities are permanent. Two SSF derived modules provide a pressurized volume for housing eight permanent crew with one year tours of duty. Advanced lunar EMUs and EVA systems permit daily, full duration extravehicular activities. The modules include a workshop capable of minor repairs. A

larger pressurized habitat structure is under construction using ISRU construction materials. Piloted surface transportation includes a midrange pressurized rover and a short range unpressurized rover. The pressurized rover serves as a mobile work station and includes power and tools. Regolith mining operations provide feedstock for O_2 and volatile extraction plants that produce ample LOX, H_2, and other volatiles for lander propellants and base supplies of process reagents, life support, and other expendables. The mining equipment has attachments that reduce the need for special construction equipment. A ceramics plant and metals plant also use regolith to produce construction materials. Other facilities provide for product storage and distribution. A 1 MWe class nuclear reactor provides the power for base operations. Advanced landing and space vehicle servicing facilities are located near permanent pads that provide a base for landers that ferry crew and cargo from space-based transfer vehicles. Surface improvements include roads and parking areas for specialized surface vehicles such as propellant carriers.

The base is substantially closed with resupply from Earth of low intrinsic value items such as life support gasses, propellants, and bulk construction materials. Plant, mining, vehicle servicing, and construction operations are substantially automated. Base operations occur throughout the lunar day and night. The plants, energy system, and other base facilities form an integrated, synergistic complex that optimizes energy and raw material usage.

5.4.5.2.1. Human Systems

A permanent crew of eight with one year tours of duty resides at the base. (The crew size and tour will be iterated.)

5.4.5.2.1.1. Life Support Systems

The base provides a regenerable life support system for lunar habitat, rovers, and other applicable surface elements that is substantially closed with resupply of fluids from Earth (less than 1%). The life support system can make use of gases, water, etc. produced from local resources. The budget excludes initial amounts needed to start up the system, and contingency supplies but includes emergency supplies. Food and crew personal supplies come from Earth, however, there is substantial fresh food produced in CELSS.

5.4.5.2.1.2. Shelter

Two SSF derived modules provide shelter and work areas for a permanent crew of eight with one year tours of duty.

- The habitat/workshop will require 100 kWe.
- The workshop is in a volume of 1/3 SSF common module. It includes equipment such as a programmable multi-machine, handtools, small power tools, test equipment, and an interface for mounting external equipment or LRUs.

  Initial habitat is largely self-contained and erected at base. Final habitat will use ISRU construction materials. A product of the study is to define these ISRU construction materials.

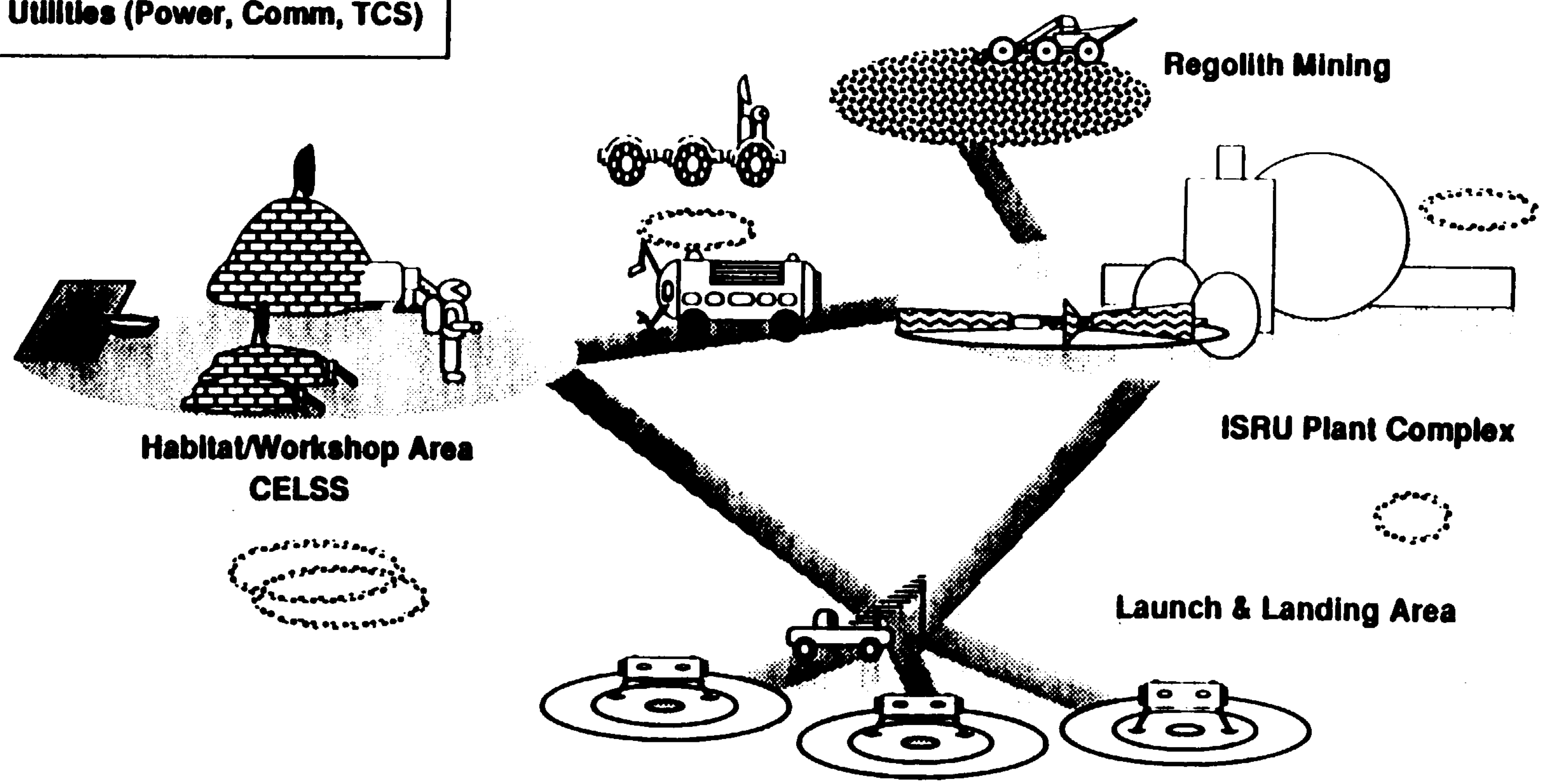
Table 5.4.5.2-1 Lunar Resource Base Baseline Elements Summary

Functional Area	Element	SSEC	EDF
Comm/Nav/Data	Surface TNIM		103-5
Construction	Crane	12	061
	(mining equipment with attachment)	10	040
Energy	50 kWe PVA/RFC (for Hab backup)	35	003
	1 MWe Nuclear Reactor	32	001
	Energy Distribution System		051
Human Systems	Lunar EMU	19	010
	Habitat Modules (8 crew)		027
	Regenerative Life Support	23	022
	TCS	24	032
	Workshop		053
	Large ISRU Pressurized Volum		054e
ISRU	Digger	10	75
	Feedstock Carrier		76
	Beneficiation Facilities		77
	O_2 Plant	27	45
	Volatiles Plant		78
	Sintered Regolith Plant		77
	Metals Plant		80
	Construction Materials		81
	Storage and Distribution Facilitie		82s
Launch & Landing	Navigation Aids		036
	Pilot Aids	26	096
	Propellant Carrier	14	097
	Cord Cart	15	098
	Landing Pad		099
	Thermal & µ-meteoroid Blanket		100
	Auxiliary Power Cart	16	101
	Thermal Control Cart		102
Manufacturing	NA		
Surface Transportation	10 km Unpressurized Rover	6	16
	100 km Pressurized Rover	7	60
	Roads		43
User Accommodations	NA		

Figure 5.4.5-1 Lunar Resource Base

◊ Learn to life and work on extraterrestrial surface
◊ Capture resource opportunities for self-sufficiency

Infrastructure
- Habitats
- Launch/Landing Facilities
- Surface Transportation
- Propellant Plants
- Utilities (Power, Comm, TCS)



5.4.5.2.1.3. EVA

Systems support four EVAs daily. Each EVA provides 8 hours of work on application. Frequent, full duration EVAs occur.

- Operations will minimize EVAs.
- EVA will involve at most 4 crew for 8 hours. (Except for crew transfer).
  - Walk, carry light objects, manipulate large objects.
  - 115 kg suits, with Apollo like flexure.
  - EVA crew can connect / disconnect umbilicals, replace LRUs.
  - EVA crew cannot do fine work.

5.4.5.2.1.4. IVA

The base systems provide for supervisory control of ISRU plant and mining operations and remote control of construction.

5.4.5.2.2. Construction

- Mining equipment with attachments provides the baseline for construction equipment.
- Civil engineering projects makes maximum use of ISRU materials.

5.4.5.2.3. Surface Transportation

Provide for transportation of crew and materials in the vicinity of landers and the base. Study shall analyze the needs for local transportation at the base and define a suitable set of performance requirements.

- Pressurized 100 km vehicle that holds crew of 4. Provides limited tools and power for base operations. For initial designs assume required performance is 100 km range and a total payload of four crew with EVA suits, 72 hours nominal operations, 100 hours of life support supplies per person, and payload of 1000 kg. Use is daily.
- Unpressurized, 2 crew rover. For initial designs assume required performance is 10 km range and a total payload of 2 crew in EVA suits, 48 hours of life support supplies per person, and payload of 1000 kg. Use is daily.

5.4.5.2.4. In Situ Resource Utilization

Provide capability to produce the following from lunar material.

Product	Amount (t/yr)
O_2	150
H_2	15
C	5
Other volatiles	Capture all available
Metals	50
Ceramics	100

- ISRU products have suitable purity (e.g., rocket propellant, fuel cell, or, life support grade).

- Production plants form an integrated complex that takes advantage of process synergies (e.g., waste heat, by-products,...). The intent is to investigate leverage of integration.
- Mining, beneficiation, production, and storage need not be collocated.
- Sintered Regolith Ceramics Plant is based on Battelle study.
- Initial estimate of mining is 45,000 t/yr.
- Assume mining equipment except in ISRU includes excavator with smoother attachment, hauler.
- The ISRU complex provides for storage of products and their distribution.

5.4.5.2.5. Energy

- A base power supply of 1 MWe from a nuclear reactor is assumed as an initial estimate.
- A 50 kWe PVA/RFC power supply is assumed as an emergency backup for the habitat modules.

5.4.5.2.6. Manufacturing

NA.

5.4.5.2.7. Launch/Landing Operations

- IA & TIA interface agreement defines basic interface.
- Traffic assumptions: 3 flights/yr.
- Piloted: 2/yr, 4 crew and 6 t cargo.
- Cargo: 1/yr, 20 t.
- Landers use cryogenic H_2/O_2.
- Equipment includes crane, propellant cart, power cart, radiator cart, auxiliary power cart.

5.4.5.2.8. User Accommodations

User accommodations are not considered in this baseline.

5.4.5.2.9. Comm/Nav/Data

No special provisions are identified.

5.4.2. Mars Outpost

The Mars Outpost Reference Architecture considers the surface systems for establishment of a small, man-tended outpost on the Mars surface for the early human exploration of Mars. The baseline assumes that the mission objectives are to establish an initial human presence on Mars and to undertake the scientific and human exploration of Mars and its moons. This architecture is designed to focus on the major enabling elements, their performance envelopes, and their construction, operations and potential evolution.

5.4.2.1. Overview Summary

Mission Objective: Establish and safely and reliably operate a human tended exploration outpost on the Martian surface.

Surface System Mission: Construct, operate, and promote outpost.

Top-level Functional Requirements

A. Provide safe housing and reliable support for base crew.
B. Provide outpost facilities.
   1. Energy
   2. Comm/Nav/Data
   3. Maintenance & resupply
   4. Pressurized workspace (laboratories & workshops).
   5. Lander Accommodations
C. Provide surface transportation.
   1. Exploration.
   2. Base operations.
D. Construct or deploy user elements.

Key Assumptions and Guidelines

A. Assure crew health and safety and reliable mission operations.
B. The site is the Chryse Basin complex, at 0° latitude, 33.5° west longitude.
C. First manned landing at site deploys operational habitat module and stays for a total of 30 days.
D. The surface infrastructure is minimal for accomplishing the basic exploration goals defined in the SRD Man Tended Phase of the Mars Evolution Case. The early flights (especially the first one) should strive to be minimal in the sense that they use the least equipment and operations needed to fulfill mission goals.
E. All facilities and equipment are considered permanent with 30 year life, unless further analysis indicates otherwise.
F. Construction and operations maximize use of in situ resources that do not require substantial infrastructure. Substantial ISRU (e.g., propellant plants) are beyond the scope of this baseline.
G. One cargo and one piloted vehicle are at the base site. Initial analysis need not consider latitude effects on vehicle performance.
H. Crew, resupply, and cargo flights occur every mission opportunity, i.e. about every 26 Earth months.

I. Vehicles land within 25 m foot print with enough accuracy for a 50 m pad. Blast effects extend 1 km for rugged equipment and 10 km for delicate equipment.

J. Support of user operations and accommodations is limited to initial construction or deployment and periodic maintenance and resupply unless specified otherwise. User accommodations will not be addressed in the initial FY89 baseline.

K. Only those activities on the Martian moons that are unique to developing the surface outpost will be considered in this subcase. General exploration is the subject of the μg Body Exploration Architecture. Resource development is the subject of the Mars Resource Base Architecture. The initial FY89 baseline will address only Martian surface activities.

L. Except for engineering data and programmatic concerns and for final outpost site selection, precursor exploration missions should be ignored.

M. The space transportation vehicle supplies crew habitat facilities for Phobos operations.

5.4.2.2. Description

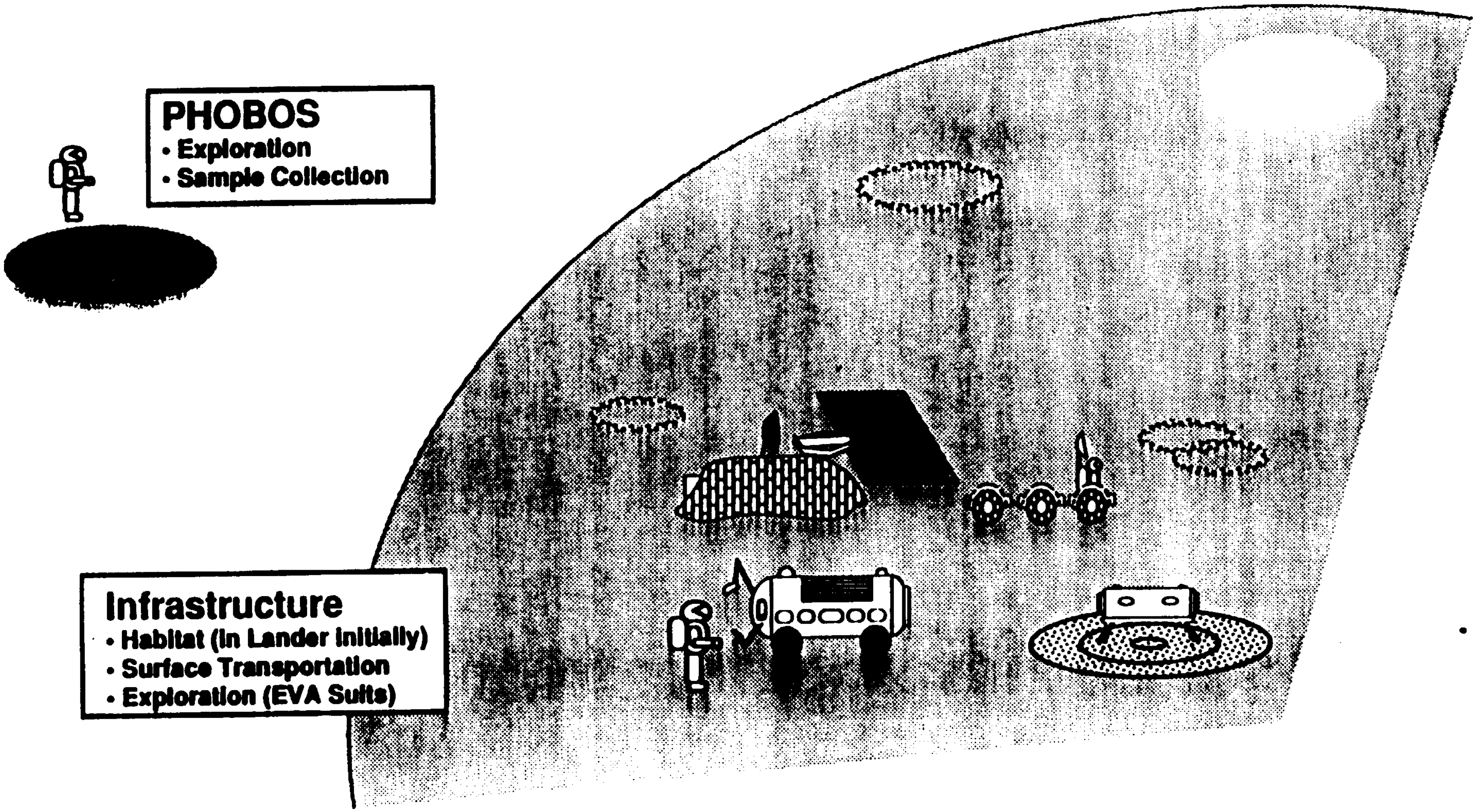
Figure 5.4.2-1 depicts surface elements. The outpost is man-tended with tours of duty ranging from 1 to 2 years. Infrastructure consists of habitats, surface transportation, and EMUs and EVA support. Table 5.4.2-1 summarizes the surface elements. These include construction equipment, rovers for local and remote surface transportation, power sources, and a pressurized habitat for crew and laboratories. User accommodations are for science and exploration. The baseline user set is taken from the SRD (§2.3.4.5.8 of the Mars Evolution Case). The current baseline limits its scope to early exploration missions and to establishing an initial outpost. Precursor missions are assumed to include an Mars Orbiter and a sample return mission that map Mars and gather engineering information. The first piloted flight emplaces a habitat, utilities (power and thermal), and improves landing facilities. Subsequent flights expand the initial minimal surface infrastructure. This baseline does not consider substantial development of ISRU.

Table 5.5.2-1 Mars Outpost Baseline Elements Summary

Functional Area	Element	SSEC	EDF
Comm/Nav/Data	Surface TNIM System		
Construction	Crane Digger (with smoother attachment		
Energy	50 kWe PVA/RFC for Habitat Module		
Human Systems	Mars EMU Habitat Module Regenerative Life Support TCS		
ISRU	NA		
Launch & Landing	Navigation Aids Pilot Aids		
Manufacturing	NA		
Surface Transportation	100 km Pressurized Rover 10 km Unpressurized Rover Truck		
User Accommodations	NA		

Figure 5.4.2-1 Mars Outpost

◊ Scientific and human exploration of Mars and Phobos.
◊ Initial human presence on Mars.



5.4.2.2.1. Human Systems

5.4.2.2.1.1. Life Support Systems

The outpost has a partially closed regenerative life support system for Martian habitat, rovers, and other applicable surface elements. Here closed means closed with respect to resupply from Earth. Partial closure for gases and water is assumed (resupply of less than 10% from Earth). The resupply budget does not include initial amounts needed to start up system, nor contingency nor emergency supplies.

5.4.2.2.1.2. Shelter

The outpost provides housing and laboratory facilities for a crew of four with tours of duty from one to two years.

- Habitat is 4 x 20 m SSF derived module.
- A 1/2 m regolith covering provides radiation protection.
- Four crew are available in a man-tended operation.
- The module require 25 kWe for crew support and housekeeping. This sum does not include science equipment in the lab.

5.4.2.2.1.3. EVA

The outpost provides for four safe and reliable EVAs daily. Each EVA provides 8 hours of work on application. Frequent, full duration EVAs occur on the surface.

5.4.2.2.2. Assembly and Construction

The outpost provides basic construction capabilities to deploy modular user elements and to deploy the habitat. Elements deployed/constructed during the first mission include:

  - Habitat Module
  - PVA/RFC
  - TCS
  - Regenerative Life Support
  - Launch & Landing Navigation & Pilot Aids
- Construction operations are all weather, daytime.

5.4.2.2.3. Surface Transportation

The base provides for transportation of crew and materials in the vicinity of landers and the base. Surface transportation includes the following.

- Pressurized rover: Provides limited tools and power for base operations. Can achieve a 100 km range from base with a total payload of four crew with EVA suits, 100 hours of life support (air, H_2O, food, etc.) supplies per person, and a user payload of 1000 kg. Nominal maximum sortie lasts 72 hours. Can be used daily. May serve as a contingency habitat for four at the base.
- Unpressurized rover: Provides portable life support system (PLSS) for crew. Can achieve a 10 km range from base with a total payload of two crew in EVA suits, 48

hours of life support (air, H_2O, food, etc.) supplies per person, and a user payload of 1000 kg. Nominal maximum sortie lasts 10 hours. Can be used daily.

No long range rover is provided.

5.4.2.2.4. In Situ Resource Utilization

No specific requirements for ISRU are identified besides covering modules with regolith.

5.4.2.2.5. Launch and Landing

The outpost provides minimal launch/landing facilities. These consist of the following

- Navigation and pilot aids (deployed in the early flights.)
- No or minimal pad

The following characteristics are baselined.

- Traffic: 1 cargo and 1 piloted flight per opportunity (~26 mo).
- Piloted capacity: 5 crew and 10 t cargo
- Cargo capacity: 50 t cargo.
- Landers use cryogenic H_2/O_2.

5.4.2.2.6. User Accommodations

The SRD specifies the baseline science elements for the outpost in §2.3.4.5.8 of the Mars Evolution Case.

- User accommodations will not be considered in the initial studies.

5.4.2.2.7. Energy

The outpost provides energy for

- habitat
- auxiliary power for landers
- rovers

A 50 kWe PVA/RFC supply is provided during the first mission. User power accommodations are as specified in the MSDB.

5.4.2.2.8. Comm/Nav/Data

Communication satellites are available for global and Earth access.

APPENDIX D

PLENARY PAPER SUMMARIES

Speaker: David Carrier
Topic: Geotechnical Engineering on the Moon

From a geotechnical point of view, the Moon is a relatively simple place. The soil extends to a depth of at least 2 m, and perhaps to 20 m or more, virtually everywhere on the lunar surface. There are occasional big boulders, but bedrock is not at the surface.

The three main constituents most likely to be responsible for problem soils on Earth are absent on the Moon: there is no water, no clay minerals, and no organics. Furthermore, the mineralogy and particle size distribution of lunar soils are limited to a fairly narrow range. As a result, the single most important factor controlling lunar soil behavior is relative density. While the top few centimeters are very loose, eons of meteorite impacts have produced a very dense sub-surface. A "lunar building code" is now being written which will cover such topics as allowable bearing capacity, slope stability, and trafficability. This information will be used for the design of lunar bases, observatories, and mines in the 21st century. (Conclusion from Carrier and Mitchell, 1989)

Speaker: Joseph Neudecker
Topic: Tunneling and Glazing in Place

High-speed lunar surface transportation between manned scientific, commercial, or logistical facilities will require subsurface tunnels because humans must be shielded from Galactic Cosmic Rays and Solar Proton Event irradiations. We present a concept called SUBSELENE in which heat from a nuclear reactor is used to melt rock and form a self-supporting, glass-lined tunnel suitable for Maglev or other high-speed transport modes. We argue that SUBSELENE is an optimal approach to forming transportation tunnels on the Moon because: (1) it uses a high-energy-density, high-efficiency, nuclear power supply; (2) it does not require water or other rare volatiles for open system muck handling or cooling; (3) it can penetrate through a mechanically varied sequence of rock types without complicated configurational changes; (4) it forms its own support structure as it

goes; and (5) it is highly amenable to unmanned, automated operation. We outline the R&D needed to develop a SUBSELENE device with small-scale, field-tested, rock-melting penetrators. (Summary from Neudecker, et al, Los Alamos National Laboratory, no date given)

Speaker: Robert Waldron
Topic: Oxygen Production Methods

A set of applications were described for lunar resources in the categories of raw soil or rock, minimally processed soil or rock, and refined products. Applications included among them radiation shielding, energy storage, reaction mass, propellants, heat exchange, thermal insulation, manufactured products and export products. Lunar resource accessibility for known lunar elements was described, ranging from easy to difficult. Basics of integrated materials processing were shown, from mining to final product. Lunar resource utilization constraints were described. A set of potential materials derivable solely or predominantly from lunar sources was identified. Propellant process and operations performance parameters were described. Also outlined were candidate propellant systems for in-situ propellant production and lunar propellant selection options, manufactured vs. import . The in situ propellant production candidate systems were classified according to use or non-use of reagents, terrestrial or lunar. Other processes were classified according to the first major step in the process. Lunar LOX preliminary process options were identified. Advantages of separate H2 and other volatile-separation processes were outlined. Common features of LOX and H2 extraction were shown. Steps for in situ propellant productions were shown, as were infrastructure, mass and power requirements, and generic options. (adapted from plenary session viewgraphs from R. Waldron, Rockwell Lunar and Planetary Systems, May 1989)

Speaker: Willy Z. Sadeh
Topic: Alternate Construction Methods

Basics of lunar environment were described. Four stages of lunar development were stipulated: exploration, pioneering, outpost, and base. Engineering tasks specific to construction on the Moon were outlined, as were a set of general criteria for a lunar base. Activities specific to each development phase were described, including the types of structures considered appropriate to that phase. Initial structures were pre-fabricated, deployable, terrestrial imports. "Base" phase structures were large, using both Earth-imported and lunar-produced (in-situ) components and lunar fabrication. Lunar construction conditions were described, including general environmental conditions, regolith, and foundation loads. Structural characteristics were described. Construction system options were listed, including a variety of prefabricated/preassembled modules and inflatable modules. A Colorado State University option using inflatable fabric modules "tufted" by tensile columns was described in detail. A set of comparative considerations for different types of structures was listed. Advantages and disadvantages of prefabricated/preassembled modules, rigid modules, inflatable modules, erector-type modules, post-tensioned concrete modules and underground modules were discussed. (adapted from plenary session viewgraphs, Willy Z. Sadeh, Marvin E. Criswell, Paul S. Nowak, Colorado State University, May 1989)

Speaker: Dave McKay
Topic: Extraterrestrial Resources

It was explained that using in situ resources is desirable because the cost of transportation prohibits importing all required materials from Earth. Office of Exploration (OEXP), Pathfinder, and the Office of Space Science and Applications (OSSA) were identified as three NASA entities which are looking at lunar based resources. Four topics were identified as central to in situ resource utilization (ISRU): oxygen production (for use primarily as propellant); recovery of volatiles such as hydrogen; production of construction materials; metals recovery. Oxygen production and volatiles recovery involve roughly similar kinds of activities: mining, beneficiation, processing, capture, storage, recycling of reagents, and waste disposal. Metals are a

byproduct of oxygen production. Construction materials production can range from simple sandbagging of regolith or sintered bricks to the manufacture of concrete, ceramics, or glass. Further discussion detailed significant aspects of recovery of those resources. Present resource recovery priorities are on oxygen production as propellant and lo-tech construction materials because they offer the greatest economic leveraging.

Presenter: Ray Leonard
Topic: Third Strawman Scenario for Workshop: "Material Handling Systems for Lunar Mining, A System Design for 100 Metric Ton Annual Oxygen Production"

This study, conducted at the request of the Mining and Construction Workshop Steering Committee, stems from a brief system description presented by the author to the steering committee in St. Louis, February 21, 1989. The system described consisted of a hydrogen/oxygen fuel cell powered front end loader and similarly powered hauler or dump truck. The system was picked for its simplicity and versatility. The two units together with a series of attachments, such as a backhoe for the front end loader and a dozer blade for the truck, would be able to handle most if not all of the early material handling requirements. This more formal study provides the documentation and justification of the assertion made at that planning meeting.

The study also includes a set of recommendations for research and development. Some of the testing requirements will require a simulation facility sized to accommodate a front end loader and which will have lunar soil modeling, thermal cycling, and vacuum capabilities. (Executive Summary of strawman, Ray Leonard, Ad Astra, Ltd., Santa Fe, New Mexico, 1989)

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APPENDIX E

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FIGURES

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APPENDIX F

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APPENDIX G

WORKSHOP SCHEDULE

	May 2	May 3	May 4
Morning	Opening Plenary Session	Identify Linkages (3 concurrent sessions)	Working Session Reports Chairmen Reports
Afternoon	Define Problems (3 concurrent sessions)	Define Next Workshop Goals (3 concurrent sessions)	Steering Committee Meeting

APPENDIX H

LUNAR CONSTRUCTION SITE DEVELOPMENT TOPICS

Prime Topic	Secondary Topic	Tertiary Topic
Define "construction sites"		
List required exploration documents	•ERD-Exploration Requirements Document	
	•SRD-Study Requirements Document	
	•PSSRD-Planet Surface Systems Requirement Document	
	•Elements Catalog	
	•Study Data Handbook	
	•Special Assessment Studies	
Define site development and interface boundaries	•Surface Stabilization	
	•Roads/Transportation System	•Power Distribution •Communication •Water/fluids •Waste recycling •Heating, ventilation, and air conditioning (HVAC) •Waste heat •Thermal control and management
	•Launch/Landing Facility	
	•Paving and Lighting	
	•Grading (Cut/fill)	
	•Shielding (regolith)	
	•Foundations	
	•Site Exploration	•Surveying/topography •Subsurface exploration •Siting criteria
	•Site planning and arrangement	•Plant/animal module •Habitats •Tankage/pipelines •Warehouse •Service facility ••mobilization ••operational ••propellant plant ••power facility

Prime Topic	Secondary Topic	Tertiary Topic
Define assumptions	•Latitude and longitude given	
	•Process-independent	
	•Resource recovery-main focus	
	•Specific amount of material to be used: 10, 45, 100,000 t/yr;interface at mine excavator/hauler	
	•Mining regolith	
Identify Challenges/Problems	•Realistic determination of power requirements for construction operations, processing, and manufacturing (current NASA estimates seem low)	
	•Road Serviceability (design approach)	
	•Selection of surface transportation system (materials/moving equipment)	
	•Siting layout considerations: dust, vibration, hauling distance, hazardous materials and processes	
	•Foundation design and construction	
	•Construction mobilization: laydown area, equipment	
	•Contractual relationships	
	•Equipment matched to tasks	
	•Simplicity	
	•Staged development of the lunar base (interim process and facilities)	

Prime Topic	Secondary Topic	Tertiary Topic
Identify Challenges/Problems	•Development of Lunar Building Code Design Manuals (include soil structure interaction guidelines and standards)	
	•Development of design and standards for machinery and soil interaction	
	•Learn from experience	
	•Behavior of regolith - excavation, pouring, strength, compressibility	

Participants chose to address identification of challenges and problems because this was the focus of the workshop. The topic of contractual relationships was deleted.

1.1.3

PROCEEDINGS OF THE SECOND WORKSHOP ON IN SITU RESOURCE UTILIZATION

Orlando, Florida

June 6-9, 1989

Sponsored by NASA, Office of Exploration

Directed by David S. McKay

Compiled and Edited by

Bridget Mintz Register

CONTENTS

PAGE ii INTENTIONALLY BLANK

Section	CONTENTS	Page

Section	CONTENTS	Page

APPENDICES

TABLES

FIGURES

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SECTION 1

Introduction

1.1 BACKGROUND AND GOALS

The second In Situ Resource Utilization Workshop was held in Orlando, Florida from June 6 through June 9, 1989. It built upon the results and organization of the first workshop, which was held in 1987.

The goal of the first workshop was to investigate potential joint public/private development of key technologies and mechanisms required to enable the permanent habitation of space. The primary goal of the second workshop was the identification of extraction methods for obtaining useful products from the regolith of the Moon. Secondary goals focused on extraction of useful products from Mars and the asteroids.

1.2 PARTICIPANTS

Approximately 60 participants attended the second workshop. They were from diverse backgrounds, including academia, NASA, contractors, and industry. Many of the participants had attended the first workshop. A complete listing of participants is given in Appendix B.

1.3 WORKSHOP STRUCTURE

The workshop organization is illustrated in Table 1.3-I.

Five working groups met concurrently. Their subject areas were:

- •Feedstock Definition and Precursor Science and Exploration
- •Chemical and Industrial Oxygen Production Processes
- •Plant Engineering
- •Volatile Extraction
- •Metals, Ceramics, Semiconductors, Arable Soils, and Other Byproducts.

TABLE 1.3-I.-WORKSHOP STRUCTURE

Day	Morning	Afternoon
6/6/89	Plenary session	Working group meetings
6/7/89	8:30-9:30: Plenary reports Working group meetings	Working group meetings
6/8/89	8:30-9:30: Plenary reports Working group meetings	Final working group meetings
6/9/89	Group leaders' final reports	

The initial plenary session consisted of a group of papers covering background technical information for the availability and extraction of products from the lunar regolith or other extraterrestrial sources. Plenary papers are summarized in Appendix A.

1.4 PROCEEDINGS ORGANIZATION

The organization of the proceedings document mirrors the structure of the workshop. Section 1, the Introduction, provides background and organizational information. Sections 2 through 6 report the findings and conclusions of the working groups. Section 7 lists conclusions.

Appendix A summarizes some of the plenary papers. Appendix B lists participants, and Appendix C provides references.

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SECTION 2

Feedstock Definition and Precursor Science and Exploration

2.1 OVERVIEW

The Feedstock Definition and Precursor Science Session was charged to look at feedstock definition and at the needs for additional scientific exploration of the Moon, Mars, Phobos, and Deimos to utilize their in situ resources. The session members chose to emphasize the Moon because the information available on it is much more extensive than for the other planetary bodies. (Two attachments are included; the first examines proposed lunar polar orbiters, and the second provides an overview of current knowledge of Phobos and Deimos.)

The group was given four explicit tasks. The first two tasks were to describe feedstocks in terms of certain attributes and in terms of the current state of knowledge. Those attributes are mineralogy, geochemistry, geological setting, physical state, physical properties, size, surface distribution, depth distribution, variability, homogeneity, and accessibility. An additional attribute which the working group considered vital was the nature of so-called "ore" reserves. The identification and verification of such ore reserves was thought to be of the highest importance. At the same time, this information is almost completely unknown, so the group felt that obtaining that knowledge was critical to in situ resource utilization (ISRU).

The third task was to identify the critical unknowns and to indicate how exploration/development plans could change if more information became available. In particular, the group was asked to address concerns of cost and development time for the technologies involved. The fourth and final task was to indicate the type of resources data which are needed for extraction process and plant design selection.

The session looked at feedstocks in a five-step hierarchical manner as the following list indicates:

- •Type of resource: gas, metal, or material
  - •Specific gas, metal or material
    - •Process
      - •Specific feedstock for the process
        - •Knowns, unknowns, and required precursor studies concerning the feedstock

Specific gases include oxygen, solar wind implanted volatiles (e.g., H2 and He), and magmatic volatiles (such as S). Metals which were considered include iron, nickel, cobalt, silicon, aluminum, and titanium. Specific materials include glasses, ceramics, cement/concrete, cosmic-ray shielding, and agricultural materials. Many processes were considered for gas and metal extraction, such as hydrogen reduction of ilmenite for oxygen and magma electrolysis for silicon. Materials recovery processes were undefined. (Details of extraction processes are discussed later in the proceedings. Please refer to the discussion of Section 3, Group 2.)

The feedstocks were evaluated in terms of the data which are currently known and the data which need to be determined. For some of the feedstocks, precursor studies were identified which will help fill the data gaps. Each feedstock was considered in the context of a particular product and/or process.

Three guidelines served as the basis

for the evaluations. First, only naturally occurring feedstocks were considered. Second, beneficiation was treated as part of the feedstock preparation. Therefore, byproducts from beneficiation were included under the definition of naturally occurring feedstocks, but byproducts from extraction processes were not. The final guideline was the decision to emphasize versatility of feedstock acceptance in the manufacturing processes. That decision was made because it is unlikely that the site selection of a lunar installation will be dictated by feedstock considerations. Therefore, the manufacturing processes which will be used must be versatile enough to handle local, less than ideal, resources. Alternatives to the ideal are also described and evaluated where possible.

2.2 GAS RESOURCES

2.2.1 Oxygen

Hydrogen reduction of ilmenite was the first process considered for oxygen extraction. Both high-titanium mare soils and high-titanium mare basalts were discussed. For the mare soils, it is known that the concentration of ilmenite is reduced in the production of agglutinates. In these particles, ilmenite is fine-grained, and it is commonly locked in glass and mineral fragments. Reflectance spectra of the lunar surface provide data on TiO2 concentrations rather than ilmenite. One unknown is the set of parameters for beneficiation for ilmenite. A second unknown is the level of impurities which can be tolerated in oxygen production processes. *One necessary precursor study is mapping for titanium from both Earth and the lunar geoscience orbiter (LGO). An evaluation and development of beneficiation methods is a second desirable precursor study.*

For the high-titanium mare basalts, it is known that the Apollo 11 and 17 sites are relatively good ore sites. Ilmenite in these basalts is coarser-grained than it is in the soils. However, mining and beneficiation of these basalts will almost certainly involve rock crushing. Consequently, utilizing the basalts will probably be more difficult than utilizing the soils, which can essentially be scraped up. However, the exact mining and beneficiation requirements are not presently known. Depth to bedrock is unknown. The actual locations of appropriate lava flows are unknown, as is the stratigraphy of the flows. In particular, it is unknown which flows will have high concentrations of ilmenite at depth. All of these unknowns need to be filled in by site-specific studies and evaluations.

The hydrogen fluoride process will take any silicate feedstock, and vapor pyrolysis will use any feedstock. Electrolysis with fluoride flux can use mare soil and low-iron highland soil. It is known that the mare soils will require beneficiation, but the optimum ores and sites are unknown. *Precursor studies required for using the mare soils include remote sensing for both the ores and sites, verification of any discovered ore bodies, and evaluation of the material for ease of beneficiation.* The group felt that the term "remote sensing" needed better definition.

Three different types of feedstocks were considered for the magma electrolysis method of extracting oxygen. They were highlands anorthositic gabbro soil, all highlands soils, and mare soils which are low in iron content. Highlands anorthositic gabbro soil is known to be a common highlands soil type. Optimum ore bodies and their locations are unknown. Detection, verification, and analysis of ease of

beneficiation of these ore bodies are needed. Apollo 16 and Luna 20 sites are good locations for highlands soil. Low-iron mare soils were found at the Luna 16 site, but their extent there is unknown.

2.2.2 Solar Wind Implanted Volatiles

The single process examined for extracting these volatiles is thermal release -- simple heating of any lunar soil. Mature, high-ilmenite mare basalt soils are considered to be the best feedstock for these volatiles. The proportions and absolute amounts of the gases, including helium isotopes and total helium, vary in different soils. It is known that the volatiles are concentrated in the outer 200-400 angstroms of individual lunar soil particles. In terms of grain size, the volatiles are most heavily concentrated in that fraction of soil particles which are smaller than about 50 micrometers.

Specific details on the volatile concentrations in the different soils, soil depths, and particle size groups are needed. Further study of the extant lunar collection is needed to establish the volatile proportions and absolute quantities in different soils and to establish their concentration by depth. The data base on solar wind implanted gases needs to be evaluated in terms of potential resource utilization, as does existing knowledge of thermal release methods. Additionally, volatile sources other than lunar soils need evaluation. An example of an alternate source is polar ice, where LGO data are required.

2.2.3 Magmatic Volatiles

This group of volatiles includes sulfur (S), chlorine (Cl), and other magmatic volatiles in coatings on fire-fountain-produced volcanic glass beads (e.g., Apollo 17 Shorty Crater "orange soil") and in certain soils (e.g., mare vs. highlands). As with solar wind implanted volatiles, thermal release was the only process which was discussed. Volcanic glass beads and mare basalt soil were the two feedstocks under consideration.

Volcanic glass beads are known to occur in Shorty Crater, but the extent of the deposit there is unknown. Details of thermal release of the volatiles in these beads are also unknown. Exploratory remote sensing of pyroclastic deposits would eliminate some of the unknowns concerning this resource. *More study on particle coatings is needed, both to quantify their composition and to understand their origins.*

The mare basalt soils are known to be as good a source of sulfur as the volcanic beads. Sulfur is present in the mare soils in the form of the mineral troilite, FeS, which is relatively abundant in many mare basalts. *Required precursor studies include the evaluation of sulfur content in existing samples and the evaluation of sulfur production from products of solar-wind thermal release.*

2.3 METALS

For all metals discussed below, no specific ore bodies have been identified. Any number of mare basalts or highland anorthosites, or their soils, can be utilized as source materials. It remains for precursor studies of the extraction techniques to define the exact functional needs before the details of which materials can be used are identified. The "ore bodies" for such processes are numerous.

2.3.1 Iron, Nickel, Cobalt (Fe, Ni, Co)

Two extraction processes for these metals were discussed. The first is the carbonyl process, which would use mature mare basalt soils. The second is magma electrolysis, which for metal extraction would use a high Fe mare basalt soil that would yield minimal oxygen.

For the mature mare basalt soils, it is known that all agglutinates have native Fe, but how much of that is accessible by the carbonyl process is unknown. It is also unclear as to whether those regions of the Moon with high Fe content, as identified by spectral data, correspond to areas with high native Fe (i.e., elemental iron) concentrations. Ground truth data are lacking. *To fill in these gaps, Fe occurrences should be determined from lunar sample studies.*

2.3.2 Silicon, Aluminum, and Titanium (Si, Al, Ti)

Electrolysis by fluoride (F) flux and magma electrolysis were the two techniques discussed for extracting Si. Both F flux and magma electrolysis will use anorthositic gabbro highlands soil and low Fe mare soils. As discussed above under oxygen extraction by magma electrolysis, it is known that the sites of Apollo 16 and Luna 20 are good locations for highlands soils, that anorthositic gabbro soil is a common highlands soil type, and that low Fe mare soils were found at the Luna 16 site.

Electrolysis and hydrogen fluoride (HF) dissolution were the two Al-extracting processes which were briefly considered. Any feedstock was felt to be acceptable for HF dissolution. Al is present at the five percent level in maria soils and in 15% of highlands soils. It is probable that a rather pure anorthositic Apollo 16 soil would be the more desirable feedstock.

Over 10 percent of many maria soils are comprised of TiO_2. However, it comprises less than one percent of highlands soils. The Apollo 11 and 17 soils and rocks contain the highest TiO_2 contents of all lunar samples yet examined. Such materials could be suitable for Ti extraction. No specific Ti-extraction processes were identified.

2.4 MATERIALS

A necessary precursor study for all the materials in this section is a systematic evaluation of existing ideas on products which can be made from lunar soils.

2.4.1 Glass

Glass is thought to be important as a construction material in the evolution of a lunar base. The absence of water on the Moon means that glass will have a much higher tensile strength than it does on Earth, due to the weakening effects of water. However, glass will still be very brittle, so it should be used in fiber form where great tensile stresses need to be applied. This will distribute the tensile load over many elements. To protect the glass fibers from contamination with water, they should be coated with metals such as iron, magnesium, or aluminum. (Blacic, 1985)

The members of the session listed three types of feedstocks. For optical glasses, they cited anorthite. For glass foams, they named pyroclastics. Rock wool was cited as a generic glass feedstock. Specifics of use of these feedstocks depend upon their characteristics. *To alert the community to the potential uses of glasses, it was felt that the data*

available on the uses of lunar glasses should be evaluated.

2.4.2 Ceramics, Cement, Concrete, Shielding, and Agricultural Materials

Ceramics are created as byproducts of a variety of manufacturing processes. Highlands soils may be a feedstock for ceramics, but more information must be available about processing techniques before particulars can be identified.

The use of cement/concrete depends on the evolving technology and on a variety of byproducts from other recovery, beneficiation, and manufacturing processes.

Any regolith will suffice for shielding, and it is recommended that the topography be used to advantage. Any lunar regolith is also suitable for agriculture, but some sources may be superior. Examples would be pyroclastics, and KREEP-rich rocks with enrichments of potassium (K), sulfur (S), zinc (Zn), phosphorus (P) and other elements needed in trace amounts for plant growth. Data on the use of lunar soils are evolving. Among the unknowns are the relative merits of hydroponics, rock wool, and regolith. The need and nature of any additives to lunar soil for agriculture are also unknown.

2.5 OVERVIEW OF LUNAR POLAR ORBITERS

James Burke
June 7, 1989

An overview of current proposed lunar polar orbiters, their objectives, and instruments was provided by James Burke. That overview is summarized in Tables 2.5-I and 2.5-II.

TABLE 2.5-I.-PROPOSED LUNAR POLAR ORBITERS

Sponsoring Entity	Research Objective
NASA	Lunar Geophysical Orbiter (LGO): lunar science (solar system) resource knowledge
Soviet and Japanese	Same as GLO (ESA "POLO" inactive)
Non-NASA USA	Lunar Polar Orbiter (LPO): polar ice, other resources, science

TABLE 2.5-II.-COMMON LUNAR POLAR ORBITER MEASUREMENTS

Objective	Method
Distribution of K, U, Th, Fe, D, etc.	Gamma-ray spectrography (NaI, Ge (cold))
Distribution of mid-Z elements	Alpha backscatter
Distribution of minerals (Ol, Px)	Visible/Near infrared spectrometer
Polar ices	Neutron spectrometer
Distribution of remnant magnetism	Electron reflection
Magnetic field at s/c	Magnetometer (s)
Layering	Electromagnetic sounding
Heat flow	Passive microwave radiometry
Topography and Selenodesy	Radar altimetry
Location mapping	Camera
Chemical bonds, light elements	Laser Raman spectrography

2.6 IN SITU RESOURCE UTILIZATION OF PHOBOS, DEIMOS, AND NEAR-EARTH ASTEROIDS

Lucy McFadden
June 8, 1989

2.6.1 Phobos and Deimos

Knowledge of the materials available for in situ resource utilization of Phobos and Deimos is constrained by our limited knowledge of these bodies. What follows is a review of our current knowledge of Phobos and Deimos, the primary questions which result from existing information, and what information is needed to answer these questions.

The shape and volume of Phobos and Deimos are derived from stereographic images from the Viking orbiter. The shape of Phobos is described by sixth order spherical harmonics (Duxbury, 1989), while the shape of Deimos (which shows no evidence of tidal deformation) is described by a triaxial ellipsoid (Duxbury and Callahan, 1989). For Phobos and Deimos, the mean radius is 11.0 and 6.2 km, respectively. The mass of Phobos is 1.27 +/- .11 X 10^{19} g, and the mass of Deimos is 1.8 +/- .15 X 10^{18} g (Duxbury). From Viking spacecraft tracking data, their masses are determined. Densities of 2.2 +/- 0.5 and 1.7 +/- 0.5 are derived for Phobos and Deimos (Duxbury, 1989, and Duxbury and Callahan, 1989).

Evidence for the presence of a regolith on both Phobos and Deimos comes from three independent sources: thermal inertia measurements, surface morphology and photogeology, and ground-based radar measurements.

Mariner 9 and Viking thermal inertia measurements are consistent with a lunar-like regolith. Both Phobos and Deimos have surfaces covered with craters that are debris-filled to tens of meters. Phobos has grooves across part of its surface that are approximately 100 m deep and debris-filled to tens of meters. Two craters on Phobos show layering on the crater walls at 40 m depth.

Deimos shows evidence of regolith movement, and it has no grooves. There are craters that are probably tens of meters deep that are filled to the brim with regolith. Albedo patterns indicate downslope motion of the regolith. These morphological observations indicate the presence of loosely consolidated material that has moved around on the surface since the cratering episodes.

Radar reflections from Phobos published by Ostro (1989) indicate the presence of loosely consolidated material to depths of at least 3.5 cm, the wavelength of the radar beam. The radar reflections at this scale are similar in magnitude to lunar reflections, indicating a similar degree of surface roughness. At larger human scales of meters to decameters, Phobos is smoother than the Moon. A surface density of 2 g/cm^3 +/- 20% is derived from these radar data. The radar albedo, polarization ratio and the spectral shape of the radar echo are within the range of these parameters as measured for C-type asteroids with flat, featureless visible spectra that are probably most similar to CM-type carbonaceous chondrite meteorites.

Given that three independent sources of information indicate the presence of a lunar-like regolith that is tens of meters deep in some places, we are reasonably certain of its existence. Its distribution has not been mapped thoroughly.

Mineralogical information is derived from ground-based photometry, spectroscopy, and from color ratios

derived from Mariner 9 ultraviolet spectroscopy and Viking lander images of Phobos measured through the Martian atmosphere (calibrated). The Soviet spacecraft PHOBOS acquired approximately 40 visible and near-infrared images, a few spectra covering the 0.3-1.0 micron spectral region, and several 100-image cube spectra in the 0.7-3.5 micron spectral range. When analyzed, these images and spectra will provide information on the distribution of water across the surface of Phobos.

The early ground-based data (UBV photometry by Zellner) and older spacecraft data have composite spectral shapes similar to the spectrum of asteroid 1 Ceres and a low albedo on the order of two to five percent. The spectra have a UV absorption band and are flat and featureless in the visible. These spectral features are consistent with reflectance spectra of CI and CM carbonaceous chondrite meteorites as measured in the laboratory. It is from these broad-band measurements and the density determinations that the assumption of the presence of carbonaceous material is based. No diagnostic spectral features of carbonaceous material have been observed because there are none in this spectral region.

Ground-based measurements during the 1988 opposition of Mars and infrared spectra from the French infrared spectrometer (ISM) on the PHOBOS spacecraft provide additional compositional information.

Broad-band JHK measurements were made of Deimos (Phobos could not be detected). The color of Deimos at wavelengths of 1.25, 1.5, and 2.2 microns are in the field of the redder D- and P-type asteroids which preferentially populate the outer regions of the main asteroid belt and the Trojans (in orbit in front of and behind Jupiter). Narrow-band photometry across the 3.0 micron water band were also measured (Bell et al., 1989). There is no water absorption band in Deimos spectra at 3.2 microns to the 5% detection limit of the measurements.

ISM spectra are available for Phobos. The analysis of these and all three-micron data are dependent on thermal modeling and removal of the thermal component from the spectra. This introduces a large uncertainty in the data. Reliable, verified results from the ISM spectra of Phobos are not available at this writing.

The absence of water on Deimos is not consistent with a CI or CM carbonaceous chondrite composition. The low albedo and red IR colors of Deimos suggest there are probably organics on Deimos. Data exist from the PHOBOS mission to address the question of the presence of water on Phobos. Those data are currently under analysis.

The statement that Phobos and Deimos are composed of carbonaceous material is currently a best guess. This must be kept in mind in basing planning of in situ resource utilization development on knowledge of carbonaceous chondrite meteorites. The association of the composition of Phobos and Deimos with any meteorite type must be demonstrated. This would be done with a mission capable of obtaining a chemical inventory of these satellites.

Knowing that these two satellites have a regolith and no atmosphere, we know that solar wind components H and He are implanted in the regolith. The calculations estimating how much have not been done.

2.6.2 Near-Earth Asteroids

The mandate of this workshop was to consider the Moon, Mars, Phobos, and Deimos. Resource utilization of asteroids was assumed to be covered by considering the case of Phobos and Deimos. While our knowledge of the asteroids more limited than that of Phobos and Deimos, we know that there is more compositional diversity among the near-earth asteroids than between Phobos and Deimos (McFadden et al, 1989). Continuing and expanding search programs will increase the number of asteroids available for use in the manned exploration program. We must be able to characterize the asteroids in terms of the physical and chemical parameters used to described the Moon. Ground-based and in situ studies are sorely needed to evaluate their potential contribution to NASA's future exploration program. Finally, their accessibility in terms of energy, time, and cost will determine whether or not they can be usefully incorporated into the planned expansion into the inner solar system by the human race.

SECTION 3

Chemical and Industrial Oxygen Production Processes

3.1 OVERVIEW

Group 2 of the ISRU Workshop was directed to examine, in detail, process candidates for the production of oxygen. They were asked to examine a number of process characteristics. Two such characteristics were the type of feedstock the process required and the degree of feedstock flexibility the process could accept. Another set of process characteristics included product stream, composition, efficiency conversion, and energy requirements. Feedstock sizing and unit size were additional attributes. This group was also asked to fill out Element Definition Forms (EDFs) which specified these process and plant characteristics. This section was written by Eric Christiansen, NASA/SN3.

3.2 OXYGEN PROCESSES

Fifteen lunar oxygen production processes were described (see Section 3.5). These varied from processes with similarities to existing terrestrial industrial practices, such as ilmenite reduction, carbothermal reduction, and fluxed electrolysis, to more exotic systems such as ion separation and vapor phase reduction.

The 15 lunar oxygen processes were compared on a relative basis using eight different criteria, as given in Table 3.2-I. Four criteria were considered more important and were therefore weighted at twice the value of the others. Thus the unweighted value for each of the four criteria was multiplied by two before it was entered in the table. The four criteria which received the double weightings were:

1.) Process simplicity. This criterion reflects on relative cost of development, ease of operation, process reliability, and confidence in process viability under lunar environmental conditions. This comparison was made on the basis of the number of major process steps (elements).

2. Maintainability. This criterion is an indication of process duty cycle (i.e., how often the plant is down for maintenance) which drives plant size for a given production rate, operations expense (in terms of the necessary level of manpower, resupply, telerobotics), and overall system lifetime. The process maintainability comparison was made on the basis of the severity of process operational conditions (i.e., corrosive and/or high temperature environment).

3. Yield. Process yield is defined as the amount of oxygen recovered from a given quantity of raw lunar material. It is an indication of the relative amount of mining and solids material handling necessary for the various processes.

4. Resupply requirements. This comparison was based on the relative efficiency of reactant recovery.

As given in Table 3.2-I, additional comparison criteria are also important. These include performance numbers such as ratios of plant mass and power to oxygen production rate. However, these were not included in the comparison because they were not known to the same level of detail in all cases. The purpose of the comparison was to use available knowledge to make an initial cut at selecting the most suitable lunar oxygen production processes. These processes were then

TABLE 3.2-I.-RELATIVE COMPARISON OF LUNAR OXYGEN PRODUCTION PROCESSES

Criteria

	Maturity	Feed-stock Req.	Yield	Reactant Resupply	By-products	Complexity	Reliability	Energy	Wted Sum	Cut-off= 37	Non-wted sum
Wting Factor:	1	1	2	2	1	2	2	1			
Process											
Ilmenite Redn by H2O	4	1	4	8	2	10	10	2	41	41	25
Carbotherm Redn	4	5	6	6	3	8	6	2	40	40	27
Vol Xtrac	3	3	2	10	1	8	10	4	41	41	26
H Sulfide Redn	3	3	6	6	2	6	2	2	30		20
Carbochlorination	2	5	10	2	5	2	2	2	30		22
Fluorine Xchng	2	5	8	2	4	2	4	2	29		21
hydrofluoric acid leach	3	5	8	4	5	2	4	2	33		24
magma electrolysis	3	3	6	10	2	10	2	1	37	37	23
caustic electrolysis	2	5	6	6	3	2	4	2	30		21
Li Redn	2	5	6	6	2	8	4	2	35		23

TABLE 3.2-I.-RELATIVE COMPARISON OF LUNAR OXYGEN PRODUCTION PROCESSES (cont'd)

	CRITERIA										
	Maturity	Feedstock Req.	Yield	Reactant Resupply	Byproducts	Complexity	Reliability	Energy	Wted Sum	Cutoff= 37	Nonwted sum
Wting Factor:	1	1	2	2	1	2	2	1			
Process											
Flxed Electrolysis	3	5	10	8	3	10	6	1	46	46	29
Vapor phase Redn	2	5	4	10	2	10	6	3	42	42	27
Ion Sep	2	5	8	6	4	10	4	1	40	40	26
magma partial oxidation	1	3	6	8	3	6	8	2	37	37	23
methane/ H2O ilm. Redn.	2	1	4	6	1	10	8	2	34		20

studies in more detail to generate additional data for comparison, and to define the next development steps necessary for determining all applicable comparison data.

Eight lunar oxygen production processes were selected for further analyses.

System definition forms were filled out for seven of the eight selected lunar oxygen production schemes:

- •Hydrogen reduction of ilmenite.
- •Carbothermal reduction of silicates and iron oxides
- •Fluxed electrolysis
- •Molten silicate (magma) electrolysis.
- •Magma partial oxidation.
- •Ion (plasma) separation.
- •Vapor phase reduction.
- •Volatile extraction (H2 and H2O) was also selected as apromising lunar oxygen production candidate. Since Group 4 was working this process, it was not considered further by Group 2.

TABLE 3.2-I.-RELATIVE COMPARISON OF LUNAR OXYGEN PRODUCTION PROCESSES (Cont'd)

Criterion	Description	Scoring (unweighted)
1. Maturity	Technology readiness level	1=basic principles &/or theory exists 2=process conceptual design exists 3=experiments & lab-scale demonstrations 4=component validation in laboratory 5=component validation in relevant environment
2. Feedstock Requirements	Ability of process to use varied feedstock materials	1=low feedstock flexibility 1 mineral 5=all minerals - capability to go anywhere
3. Yield	Oxygen extracted per feedstock mined	1=low yield, 3=medium, 5=high yield
4.Reactant Resupply	Reactant recovery potential & resupply requirements	1=difficult, multiple step ops, dense reactants 3=medium 5=easy, low density reactants
5. Byproducts	Types & quantities of useful byproducts. Consider simplest O2 production scheme & resulting unprocessed byproducts	1=useful byproduct, 5=five or more
6. Complexity	Simplicity of O2 process scheme, in terms of the number of major process steps/units (development and operations costs, process viability, reliability)	1=9+major steps, 2=7 or 8 steps 3=5 or 6 steps 4=3 or 4 5=2 or fewer steps
7. Reliability	Relative maintainability of process compared in terms of the severity of process conditions (lifetime, maintenance requirements)	1=high temperature/ highly corrosive 3=non-corrosive/ moderate temps 5=low temp/ non-corrosive
8. Energy Requirements	Required state of power needs; i.e., thermal vs. electric (large thermal requirements indicates leverage potential from heat recovery and energy management techniques, nuclear-power waste-heat recovery).	1=100% electric power 2= >50% electric 3= primarily hi-temp thermal 4= >50% hi-temp 5= 100% lo-temp thermal

Criteria not considered: product purity, autonomous ops capability (continuous vs. batch processes). Important criteria to be evaluated after more detailed study: mass/production and power production ratios, stowed volume, operations and setup manpower requirements, costs and savings. Other criteria to consider: 1/6 gravity effects, process dependent differences in program to full-scale O2 production, need for lunar pilot plant vs. first lunar demonstration plant.

3.3 ELEMENT DEFINITION FORMS (EDFS)

Element definition forms (EDF's) were filled for a 150 t/yr lunar oxygen production plant:

Hydrogen reduction of ilmenite process systems:
- ◊ 3-stage fluidized bed reactor.
- ◊ Vapor electrolysis order.
- ◊ Reactor auxiliary systems.
- ◊ Oxygen liquefication.

Carbothermal reduction system including a silicate reduction reactor, carbon monoxide reduction reactor, water electrolysis unit, oxygen liquefication.

Fluxed electrolysis systems:
- ◊ Electrolysis cells (including containment structure, bus bars).
- ◊ Metal separation.
- ◊ Oxygen purification unit.

Magma electrolysis systems:
- ◊ Electrolysis cell.
- ◊ Containment structure and feed system.

Partial magma oxidation.
- ◊ Melting, oxidation, cooling units.
- ◊ Grinding and magnetic separation units.
- ◊ Acid dissolution and electrolysis units.

The EDF's are summarized in Table 3.3-I.

Note: Mining, beneficiation, power, thermal control radiators, and oxygen storage/distribution systems were assumed covered by Group 3, and by the 1989 Mining and Construction Workshop. In general, Group 2 defined the requirements for sizing these more generic systems (i.e., feedstock type, mining rate, power, and heat rejection loads).

Development plans and schedules were completed for the 7 lunar oxygen processes considered by Group 2. More detailed planning was possible for the better studied processes (ilmenite reduction by hydrogen, fluxed and magma electrolysis, carbothermal reduction). Specific high-leverage engineering and trade studies for the lunar oxygen processes were described in 10 issues to be considered (ITBC) forms. (Summary Table 3.3-II).

TABLE 3.3-I.-LUNAR OXYGEN PROCESS ELEMENT SUMMARY

	Feed stock Type	Mass (t)	Avg. Power (kw)	Peak Power (kw)	Stand-by Power (kw)	Vol (m^3)	Est. Op Life (yr)	Re-sup-ply (t/yr)
1. Il-menite Redn by H2	Il-menite							
a. reac-tor		23	150	450	15	43		2
b. H2O elec trolysis cell		1.3	210	210	0	2.5	5	
c. Reac-tor auxil iaries		6	40	40	2			0.6
d. O2 lique faction		12	30	30	3			
TOTAL		42.3	430	730	20	45.5		2.6
2. Carbo thermal Redn	No restrictions (Silicates, FeO reduced)							
(Estimates include reactors, H2O electrolysis, O2 liquefaction)								
TOTAL		13.3	386	463			10	
	Feed stock Type	Mass (t)	Avg. Power (kw)	Peak Power (kw)	Stand-by Power (kw)	Vol (m^3)	Est. Op Life (yr)	Re-sup-ply (t/yr)
3. Fluxed Elec-trolysis	No restrictions (all oxides reduced)							
a. elec-trolysis cells		93	252	252	210	15	6	
b. metal separa-tion		10	10	10	10	2	6	
c. O2 purif-ication unit		2	1	1	1		6	
TOTAL		105	263	263	221	17		

TABLE 3.3-I.-LUNAR OXYGEN PROCESS ELEMENT SUMMARY (cont'd)

Magma Electrolysis	Lo-Ti Mare Basalts, Most highlands (anorthositic soils, A16 not acceptable)							
	Feed: 4.8 t per t O2 produced.							
a. Electrolysis cell		8	300			1	10	
b. Containment & feed		2						
TOTAL		10	300			1		
	Feed stock Type	Mass (t)	Avg. Power (kw)	Peak Power (kw)	Stand-by Power (kw)	Vol (m^3)	Est. Op Life (yr)	Re-supply (t/yr)
Magma Partial Oxidation	No Restrictions (high FeO preferred)							
	Feed: 34 t per t O2 produced.							
a. melting, oxidation, cooling		0.5	65	65	0	1.72		
b. grinding, magnetic separation		1.6	16	16	0	0.83		
c. acid dissolution, electrolysis		2.05	312	312	0	2.02		
TOTAL		4.15	393	393	0	4.57	20	◁

Reference: Group 2, 1989 ISRU Workshop
Basis: 150 t O2/yr

TABLE 3.3-II.-LUNAR OXYGEN PROCESS ISSUES/TRADE STUDIES

Hydrogen Reduction of Ilmenite:

1. Preoxidation of Ilmenite prior to processing with H2.
2. H2S impurities removal before high-temperature electrolysis.
3. Moving bed design vs. fluidized bed vs. fixed bed.
4. High-pressure GO2(?) storage during day, power generation w/expansion/liquefaction at night.
5. Conventional liquid water electrolysis vs. vapor phase electrolysis.

Fluxed Electrolysis:

6. Power conditioning (needs low voltage, high current, high reliability).
7. Flexibility of process-minimize process complexity according to products required.

Magma Electrolysis:

8. Electrolysis cell needs engineering work.
9. Funding required for adequate progress.

Vapor Phase Reduction:

10. Beneficiation needs.
11. Rapid quenching techniques of hot gases(trade conventional condensers vs. nozzle expansion vs. others).

Plasma Decomposition, Ion Separation

12. Phase I Issues:
    Particle size, separation efficiency, side reactions, dissociation kinetics, extent of elemental ionization, contamination of gas stream, particle heating rate.
13. Phase II Issues:
    Extent of element ionization, contamination of gas stream, particle heating rates, ability to automate, utility of byproducts, compositional difference between simulated and real lunar materials, istrumentation and controls, collection and removal of by roducts
14. Phase III Issues:
    Extent of elemental ionization, contamination of gas stream, particle heating rates, heat loss in system, utility of byproducts, compositional differences between simulated and real lunar materials, instrumentation and controls, collection and removal of byproducts, weight of pilot plant, materials of construction, use of composites.
15. Phase IV Issues.
    Life time of lunar plant.

Reference: Group 2, 1989 ISRU Workshop, ITBC Forms

3.4 LEVERAGE VALUE OF OXYGEN PRODUCTION ON THE MOON, MARS, PHOBOS, AND DEIMOS.

Generalized statements presenting the advantages of oxygen production on the Moon, Mars, and Phobos/Deimos were given in 5 Justification Forms. (Summary Table 3.4-I)

TABLE 3.4-I.-JUSTIFICATIONS FOR EXTRATERRESTRIAL PROPELLANT PRODUCTION

MOON

Lunar Oxygen

1. Economics: Total program cost will be less with oxygen production.
   a. Economics improve as flight rate (number of landings/yr. increase.
   b. Economics improve as lunar program time period (or plant lifetime) lengthens.
   c. Economics improve as number of people on lunar surface increases.
   d. Economics improve if reusable (vs. expendable) landers are available.
   e. economics improve given the following process characteristics:
      i) Low mass, power, and manpower requirements.
      ii) High reliability, low maintenance, long plant life.
      iii) Simplicity.
      iv) Ability to bring the plant on-line at earliest possible date with minimum lunar pilot plant and testing requirements.
      v) High yield, simple mining requirements.
      vi) Minimum resupply requirements, such as for reactants and spares.
2. Safety, Flexibility, Reliability: Lunar oxygen increases safety, flexibility, and reliability of a lunar base program, irregardless of propellant economics.
3. Demonstration of extraterrestrial resource utilization-necessary for any long-term use of space.
4. Economic justification for lunar landers (ascent and descent) relatively easy. Use of lunar O2 in LEO dependent on more efficient lunar mass to LEO transportation system (such as mass drivers, lunar aerobrakes, lunar fuels, etc.).
5. Cost analysis for using lunar O2 instead of Earth O2 in Lunar Landers:
   Basis: 150 t O2/yr.
   One year cost saving status.
   Cost of delivering LO2 from Earth-Moon: $1650M (150t/yr@$11k/kg).
   Cost of plant delivery: $ 167 M (15.1t@ $11k/kg).
   Cost of labor: $ 170M (1700 manhour/yr @ $1000k/man-hr.).
   Annual Cost Savings (Approximate) $ 1,300 M

TABLE 3.4-I.-JUSTIFICATIONS FOR EXTRATERRESTRIAL PROPELLANT PRODUCTION (Cont'd)

6. Life-Support Synergism: All processes can provide O2, and some processes can recycle solid wastes, water, and nitrogen.
7. Useful byproducts: metals, ceramics, (sintered and cast),etc.

MARS

Martian oxygen, and perhaps CO as fuel:

1. Payload increases: O2 on Martian surface increases payload without drastic program changes/costs.
2. Enables single stage, reusable lander/ launcher. Reusable spacecraft necessary for efficient long-term martian base servicing.

Martian O2, CO, N2, and H2O:

3. Safety reliability and flexibility of transportation and martian based enhanced with even small O2 plant. ECLSS closure parameters become less important. Recovery from some failures possible.

PHOBOS/DEIMOS

Phobos/Deimos propellant production:

1. Decrease LEO mass for piloted Mars missions. 30% savings over enough time (20yrs?).
2. Propellant in Mars orbit increases safety, reliability, and flexibility of regularly run missions to Mars surface.
3. May enable a single stage, reusable Earth-Mars trans. system.

Reference: Group 2, 1989 ISRU Workshop, Justification Forms

System and element definition forms were submitted for a Mars CO2 Atmospheric processor (producing 100t LO2/yr), and a Phobos/ Deimos water extractor (producing 100t LO2/yr and 12.6 t LH2/ yr).

◊ Mars CO2 processor requires verification of system ruggedness, autonomy, and dust removal efficiency (dust particles > 0.1 nm must be removed prior to zirconia cell) before being ready for flight hardware design. Precursor data is needed on dust size and chemistry.

◊ Phobos/ Deimos propellant production plant design work suffers from little hard data on actual compositions and surface mechanical characteristics of the moons. Some general system level and transportation trades are possible.

3.5 LUNAR OXYGEN PROCESSES

Fifteen lunar oxygen processes were identified, described, and evaluated during Group2 activities at the 1989 ISRU Workshop. A very brief discussion of these 15 processes follows. More detailed descriptions, process schematic, reaction chemistry, advantages, disadvantages, and references for the first 13 of these 15 candidates are given in the reference: Christiansen, E. L. : "Conceptual Design of a Lunar Oxygen Pilot Plant, " Eagle Engineering Report 88-182, NASA, Contract NAS9-17878, July 1, 1988. The last two were initially discussed during the workshop sessions.

3.5.1. Hydrogen Reduction of Ilmenite

Process chemistry is:

- $FeTiO_3 + H_2 = Fe + TiO_2 + H_2O$
  Redn @ 900-1000 C
- $H_2O = H_2 + 1/2\ O_2$
  Electrolysis

Figure 3.5-1 shows a simplified process schematic.

Advantages:

◊Uncomplicated process chemistry. Verified in laboratory.
◊Reactant recovery is accomplished in one step by simple water electrolysis.
◊Low density of hydrogen translates into low reactant mass makeup (low resupply).
◊Direct terrestrial counterparts exist for major process equipment (fluidized - bed reactor). Industrial operating experience can be drawn on to assist development.
◊Process temperatures below feedstock melting point. Reduces corrosion concerns.
◊Iron production possible.

Disadvantages:

◊Ilmenite concentrates required for process efficiency.
◊More development work needed for high - temperature electrolysis cells. Sulfur impurities and dust are potential problems.
◊Kinetics relatively slow for divalent-containing ilmenite.
◊Thermodynamic conversions relatively low at reaction temperatures requiring high gas flow rates. Fluidized bed reactor possible option, but 1/6-gravity effects must be determined.

3.5.2 Carbothermal Reduction

One possible reductant is methane(CH_4)

- $(MO)(SiO_2) + 2\ CH_4 = 2\ CO + 4\ H_2 + Si + MO$ where MO is any metal oxide.
  Redn @ 1625 C
- $2\ CO + 6\ H_2 = 2\ CH_4 + 2\ H_2O$
  Catalytic React @ 250 C
- $H_2O = H_2 + 1/2\ O_2$
  Electrolysis

Process schematic in Figure 3.5-2.

Advantages:

◊Less mining required since silicates reduced. No beneficiation required.
◊Process studied in laboratory and works with magnesium-silicate lunar simulates.
◊Terrestrial counterparts exist.
◊Silicon and iron possible byproducts.

Disadvantages:

◊Molten silicates will be corrosive. Refractory line vessels required.
◊Carbon reactant recovery more difficult than hydrogen. Carbon losses in metal and slag streams more likely.
◊Need catalysts; notoriously prone to poisoning and have limited lifetimes.

3.5.3 Volatile Extraction.

Solar wind hydrogen deposited in the lunar soil can be evolved by thermal processing, 80% by 600 C. This hydrogen reacts with iron-oxides to form water. If the hydrogen is recycled through the bed to provide heat transfer, a substantial amount of oxygen can be recovered. A process schematic is given in Figure 3.5-3.

Advantages:

◊Both oxygen and hydrogen can be produced.

◊Vacuum pyrolysis data on lunar samples exists and can be used for design.

Disadvantages:

◊Large amounts of solids handling and thermal energy transfer must be accomplished with a minimum of lost gases.

3.5.4 Hydrogen Sulfide Reduction

General reaction chemistry (where M= Fe, Ca, Mg):

- $MO + H_2S = MS + H_2O$
  Reduction
- $MS + Heat = M + S$
  Therm. Decomp.
- $H_2O = H_2 + 1/2\ O_2$
  Electrolysis
- $H_2 + S = H_2S$
  Regeneration

Advantages:

◊Higher yield than hydrogen reduction of ilmenite. Less mining and solids handling. Beneficiation may not be required.

Disadvantages:

◊Thermal decomposition yield and conditions uncertain.

◊Sulfur is corrosive.

3.5.5 Carbochlorination

See EEI Report 88-182 for process chemistry. Schematic given in Figure 3.5-4.

Advantages:

◊Higher yield process because both FeO and Al_2O_3 can be reduced.

◊Production of aluminum and low carbon steel is a necessary byproduct.

Disadvantages:

◊Reactant recovery very difficult, equipment and mass intensive, and questionable efficiency.

3.5.6. Fluorine Exchange

Simplified reaction chemistry (where M = Ca, Al, Fe, Si, Mg, Ti) :

- M oxides + F_2 = M fluorides + O_2
  500 C
- M fluorides + K = Metals + KF
  Redn w/K vapor
- $KF = K + 1/2\ F_2$
  electrolysis @ 846 C

Advantages:

◊Reacts with all oxides.

◊Oxygen liberated directly.

◊Metal byproducts (Al, Si, etc.).

Disadvantages:

◊Considerably more complicated than simplified chemistry suggests. Fluorine recovery extremely complex & mass/energy intensive.

◊All steps have not been proven in laboratory.

◊Oxygen must be purified of fluorides.

3.5.7 Hydrofluoric Acid Leach

Process schematic and chemistry in Figure 3.5-5.

Advantages:

◊Reacts with all oxides.

Disadvantages:
◊Complicated HF recovery.
◊Chemistry not completely verified in laboratory.

3.5.8 Magma Electrolysis

Direct electrolysis or molten silicate electrolysis are other names. Figure 3.5-6 shows schematic.

Advantages:
◊Require no flux.
◊Beneficiation not required.
◊Laboratory studies are on-going.

Disadvantages:
◊Molten silicates extremely corrosive.
◊Electrode consumption problems can occur, even if platinum electrodes used (low melting point platinum-iron and Pt-Si alloys can form).
◊Dendritic growth of metals across electrodes which short cell can be a problem.

3.5.9 Caustic Solution Electrolysis

Process diagrams given in Figures 3.5-7a and 3.5-7b

Advantages:
◊Na in lunar materials could possibly be used to make up process losses.

Disadvantages:
◊Some authors claim high yield process with reduction of Fe, Al, Ti, and Si oxides possible. Others say only iron-oxides reduced by sodium. Uncertainties large.
◊Inert electrodes needed.
◊Caustic recovery difficult.
◊Dendritic growth possible problem.

3.5.10 Reduction by Lithium

Chemistry:

•$2\ Li + MO = Li_2O + M$
(where MO = oxides of Si, Fe, Ti)
•$Li_2O = 2\ Li + 1/2\ O_2$
(electrolysis of lithium oxide)

Figure 3.5-8 gives a process schematic.

Advantages:
◊High yield process.
◊Metals production possible.

Disadvantages:
◊Lithium oxide recovery from lithium reduction reactors solid product will be difficult.
◊Lithium oxide electrolysis needs experimental work.

3.5.11 Fluxed Electrolysis

Also goes by electrolysis of a molten salt. Figure 3.5-9 shows process.

Advantages:
◊High yield process. All oxides reduced.
◊Currently being studied.
◊Some terrestrial industrial experience applies (aluminum industry). However, several unconventional processing requirements exist.

Disadvantages:
◊Inert electrodes require further development.
◊Components of flux require recovery steps.
◊Lifetime of equipment may be limited by process conditions.
◊Fluorides must be stripped from oxygen product.

3.5.12 Vapor Phase Reduction

Process schematic in Figure 3.5-10.

Advantages:
◊Bulk lunar soil serves as feedstock. Beneficiation probably not needed.
◊Reagents not required.

◊Process currently being studied.

Disadvantages:
- ◊Energy intensive. 3000 K temperatures required.
- ◊Containment materials problems will be severe.
- ◊Removal of materials from condensors may be severe problem.
- ◊Low process pressures will require large volume equipment for given product rate.

3.5.13 Ion Separation

Figure 3.5-11 shows schematic.

Advantages:
- ◊Metals byproducts.
- ◊Any lunar soil feedstock.
- ◊Currently being studied.

Disadvantages:
- ◊Requires helium plasma carrying gas.
- ◊Extremely high temperature (8000-10,000 K), energy intensive process.
- ◊Small particle feed size required.

3.5.14 Magma Partial Oxidation

Process given in Figure 3.5-12.

Advantages:
- ◊Works on any iron-oxide containing soil. Feedstock independent.

Disadvantages:
- ◊Fairly complex for oxygen recovery. After oxidation and magnetite recovery, other oxygen production possibilities exist (iron-oxide reduction by hydrogen or CO).
- ◊Process not examined/ verified in laboratory.

3.5.15 Methane-Water Ilmenite Process

Previously known as super-critical water oxidation. Reference: Van Buskirk, P. D. : "CO/H2 Reduction of Ilmenite Process Using Water/ Methane as Feed," LESC Memorandum, June 21, 1989. Process scheme shown in Figure 3.5-13.

Advantages:
- ◊Plug flow reactor possible. Gravity independent.

Disadvantages:
- ◊Requires beneficiated ilmenite feed.
- ◊Currently just paper study. No laboratory data available. Need laboratory verification of reactor parameters/conditions as next step.

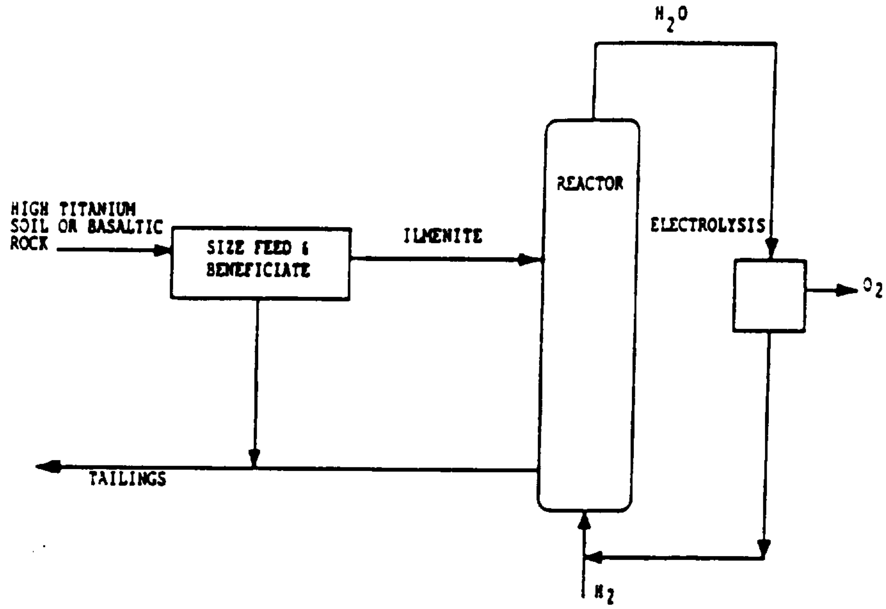


Figure 3.5-1A.-Simplified Schematic of Hydrogen Reduction of Ilmenite Process

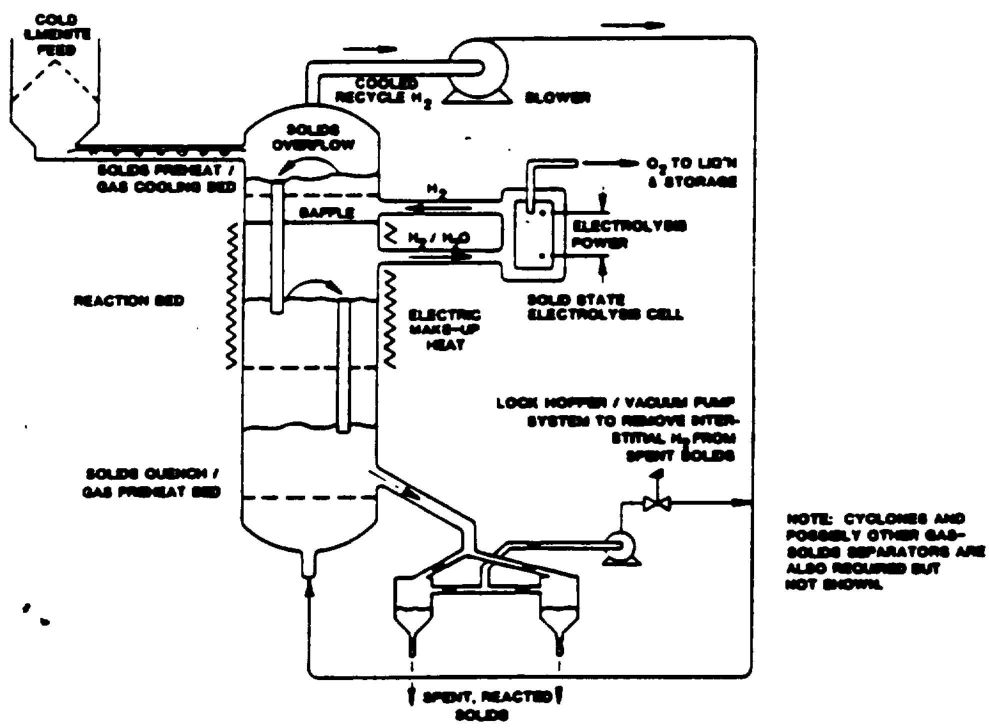


Figure 3.5-1B.-Three-Stage Fluidized Bed Reactor Concept for Ilmenite Reduction

Figure 3.5-1.-H2 Reduction of Ilmenite

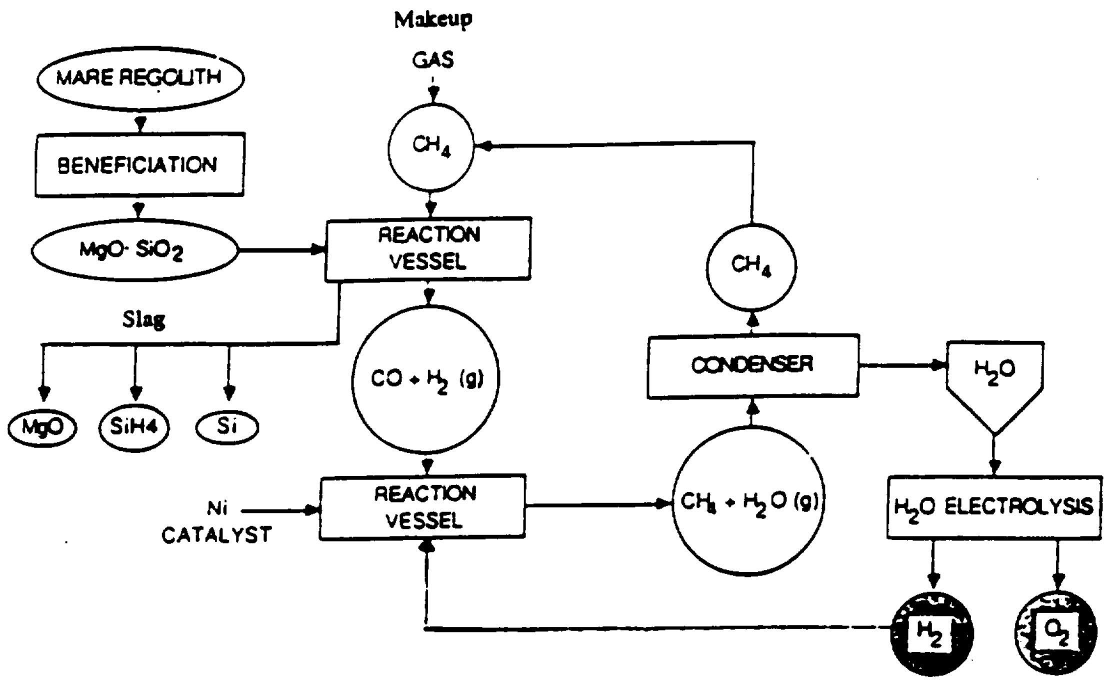


Figure 3.5-2A.-Carbothermal Process with Methane Reductant

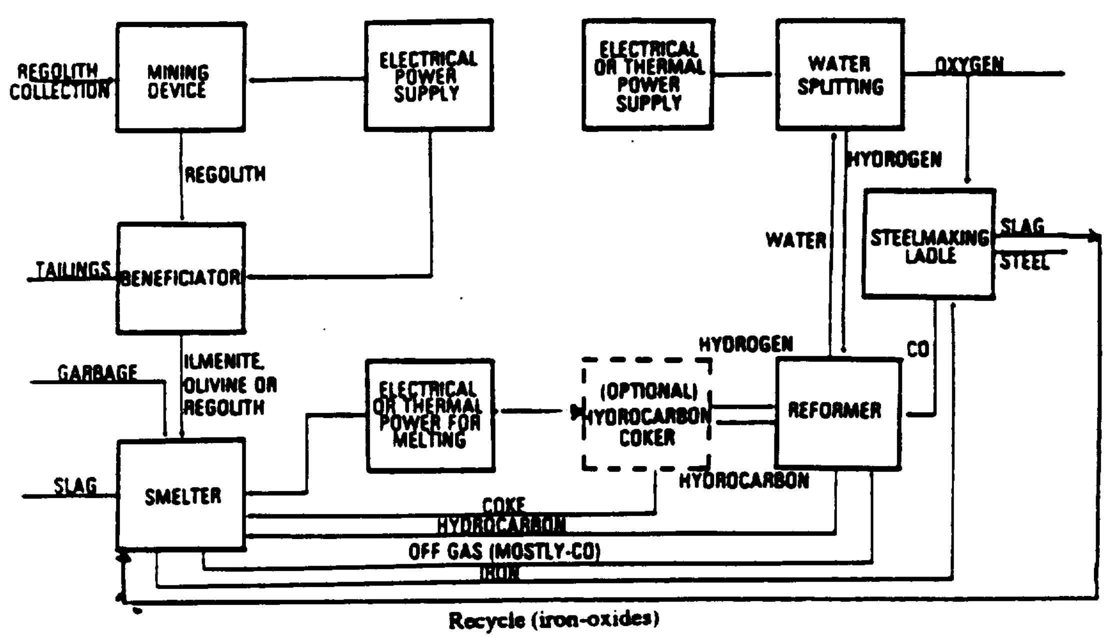
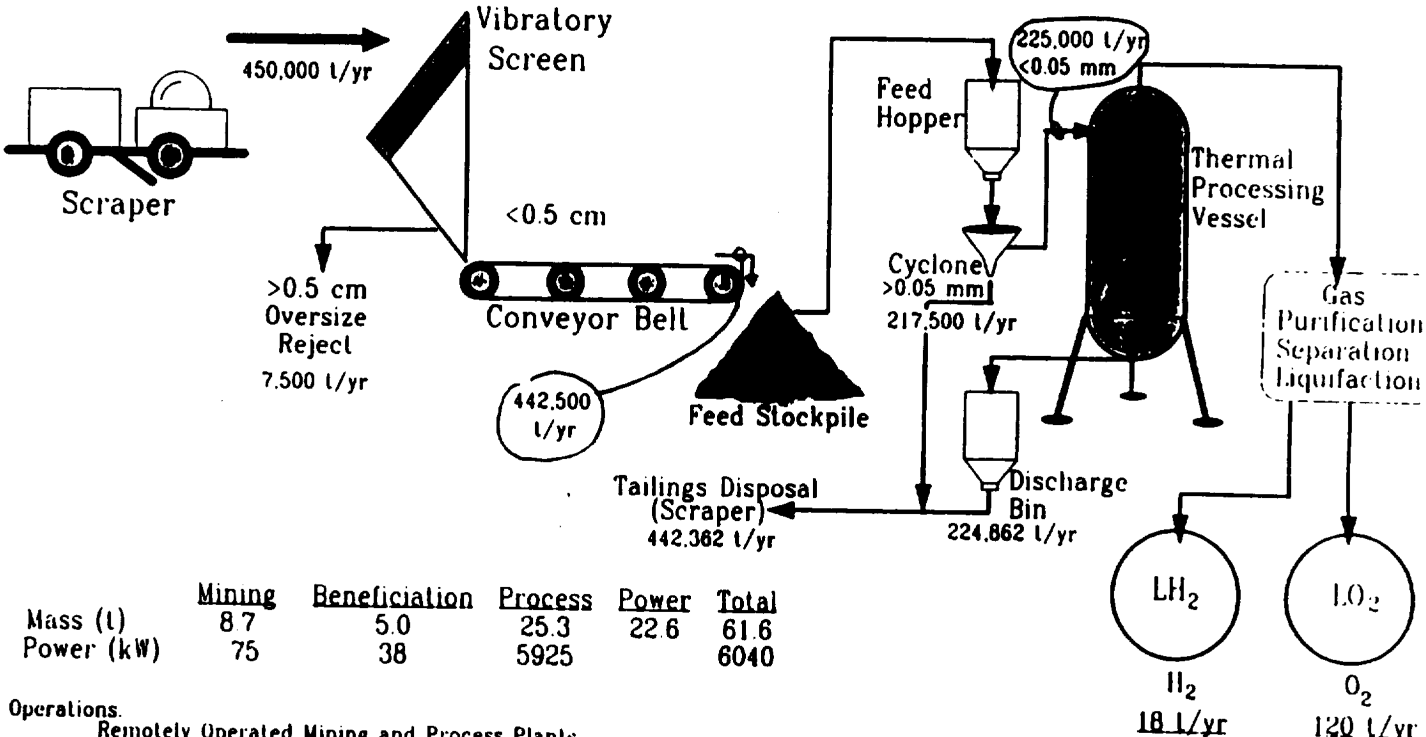


Figure 3.5-2B.-Carbothermal Process with Carbon Reductant

Figure 3.5-2.-Carbothermal Reduction



	Mining	Beneficiation	Process	Power	Total
Mass (t)	8.7	5.0	25.3	22.6	61.6
Power (kW)	75	38	5925		6040

Operations.
- Remotely Operated Mining and Process Plants
- Continuous Processing, mining/tailings disposal day only

Nuclear Power:
- Assume no nuclear waste heat used in process.
- Substantial power mass savings possible if use hi-grade (600°C) nuclear waste heat

References: Mining and Construction Workshop (1989)
Eagle Report 88-182 "Lunar Oxygen Pilot Plant Conceptual Design" (1988)

Figure 3.5-3.-Lunar Hydrogen Recovery Process

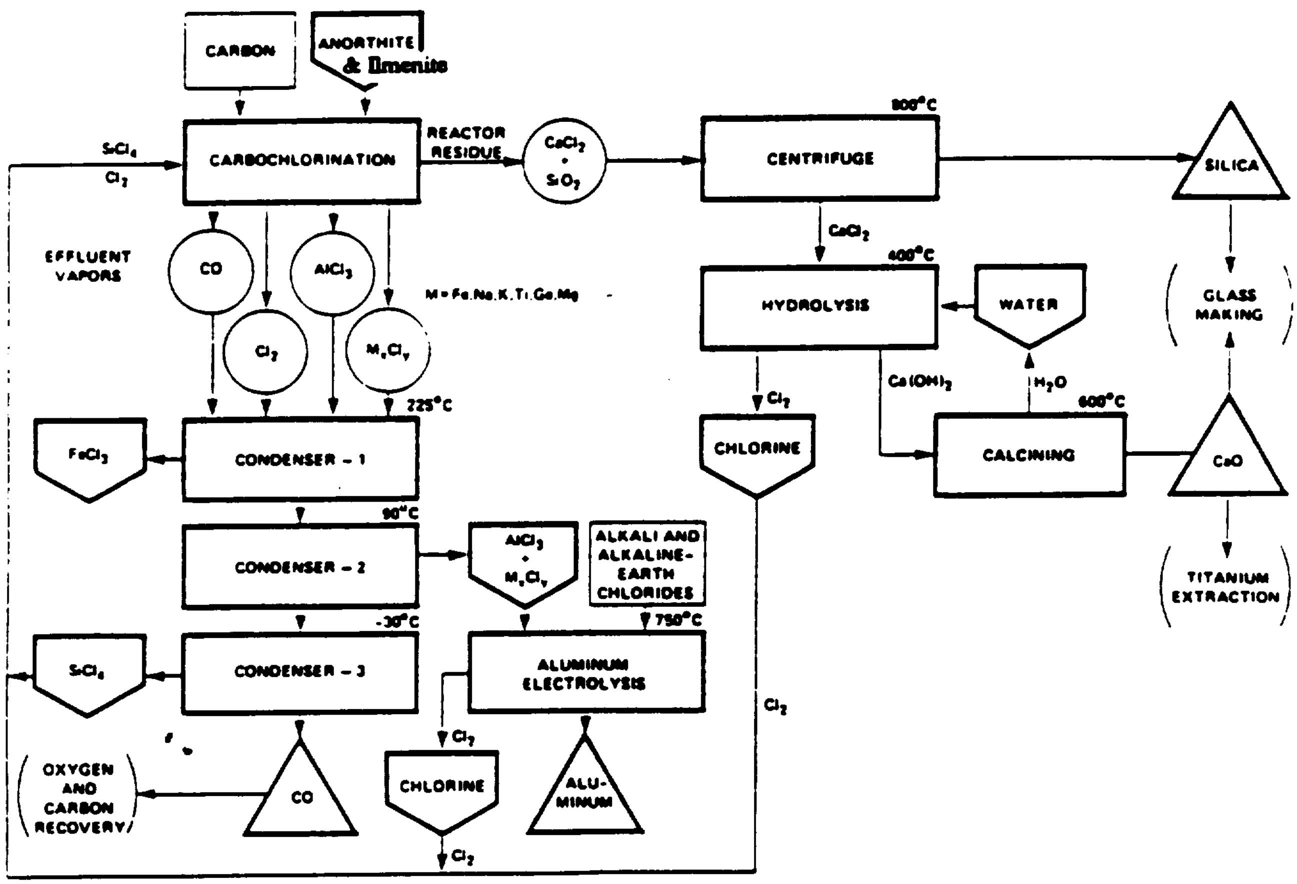
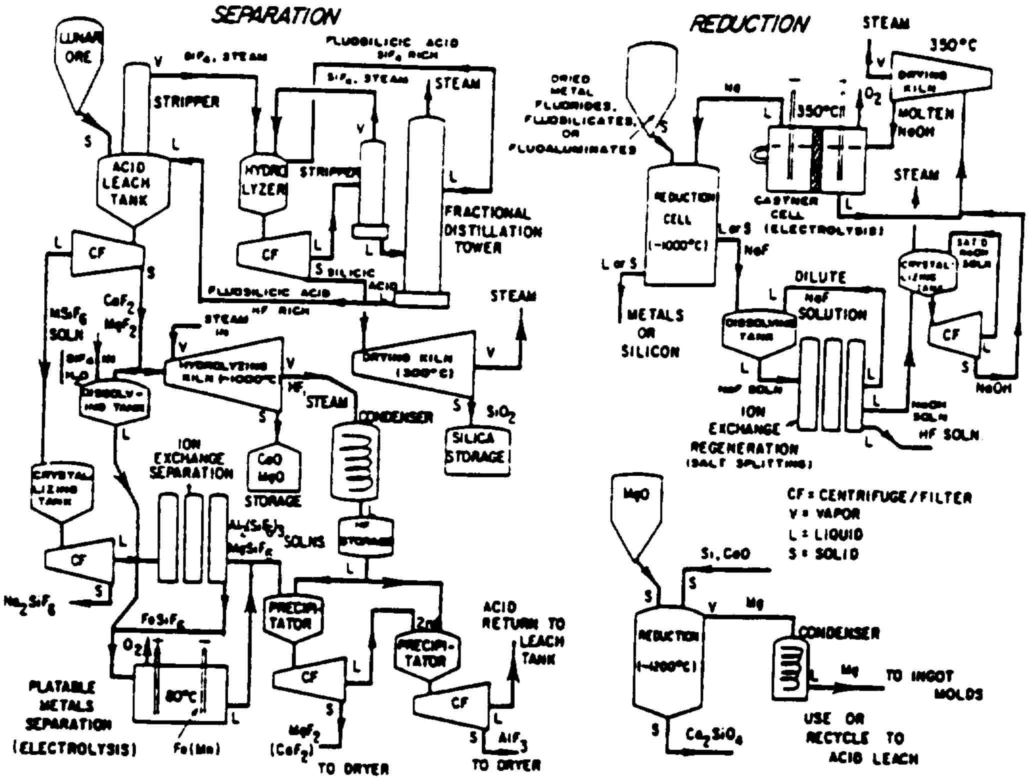


Figure 3.5-4.-Carbochlorination Process Flowsheet



HF ACID LEACH PROCESS EQUATIONS

(1) $xMO \cdot SiO_2 + (4 + 2x)\ HF = xMF_2 + SiF_4\ (aq) + (2 + x)\ H_2O$
(1a) $xMO \cdot SiO_2 + (5 + 2x)\ HF = xMF_2 + HSiF_5\ (aq) + (2 + x)\ H_2O$
(2) $SiF_4\ (aq) - NH_2O = SiF_4\ (v) + nH_2O(v)$
(2a) $HSiF_5\ (aq) + nH_2O = SiF_4\ (v) + HF\ (aq) + nH_2O(v)$
(3) $(1-y)\ [SiF_4\ (v) + 4H_2O = Si\ (OH)_4 + 4\ HF]$
(3a) $(1-y)\ [SiF_4\ (v) + 2H_2O = SiO_2 + 4HF]$
(4) $(1-y'z)\ [xMF_2 + H_2O = xMO + 2xHF]$
(5) $y\ [SiF_4 + 4Na = Si + 4NaF]$
(6) $y'[xMF_2 + 2xNa = xM + 2xNaF]$
(7) $z[xMF_2 + xSiF_4\ (aq) = xMSiF_6\ (aq))]$
(8) $z[xMSiF_6\ (aq) + xH_2O + \text{electrical energy} = (x/2)O_2 + xM + xH_2SiF_6]$
(8a) $z[xMSiF_6\ (aq) + M'SO_3R^* = xM'SiF_6(aq) + xMSO_3R^*]$
(9) $mNaF + mR^*OH = mNaOH + mR^*F$
(9a) $mNaF + (m/2)\ Ca\ (OH)_2 = mNaOH +)(m/w)\ CaF_2$
(10) $mNaOH + \text{electrical energy} = mNa + (m/4)O_2 + (m/2)H_2O$
(11) $(1-y)\ [Si\ (OH)_4 = SiO_2 + 2H_2O]$

Note: R^* = ion-exchange; $m = 4y + 2xy'$

Figure 3.5-5.-HF Acid Leach Process Schematic

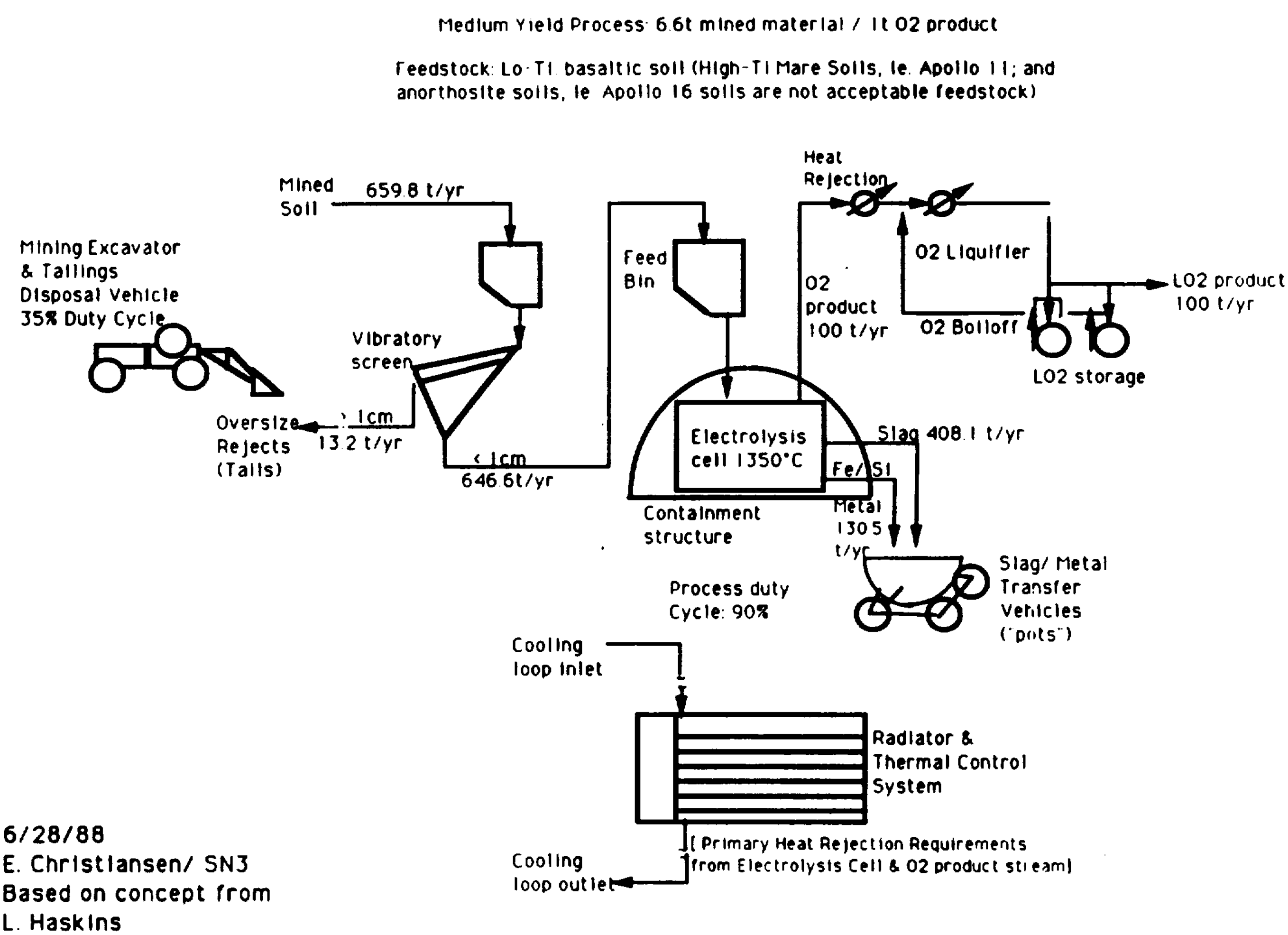


Figure 3.5-6.-Molten Silicate (Magma) Electrolysis Schematic

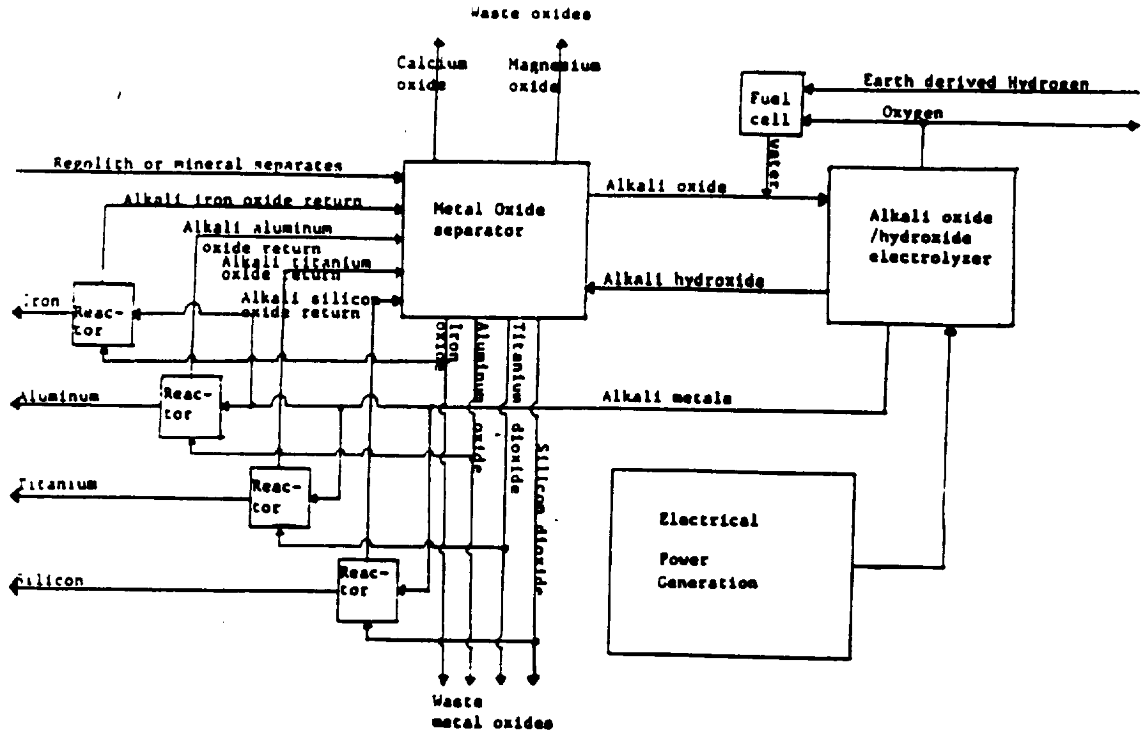


Figure 3.5-7A.-NaOH (Caustic) Electrolysis

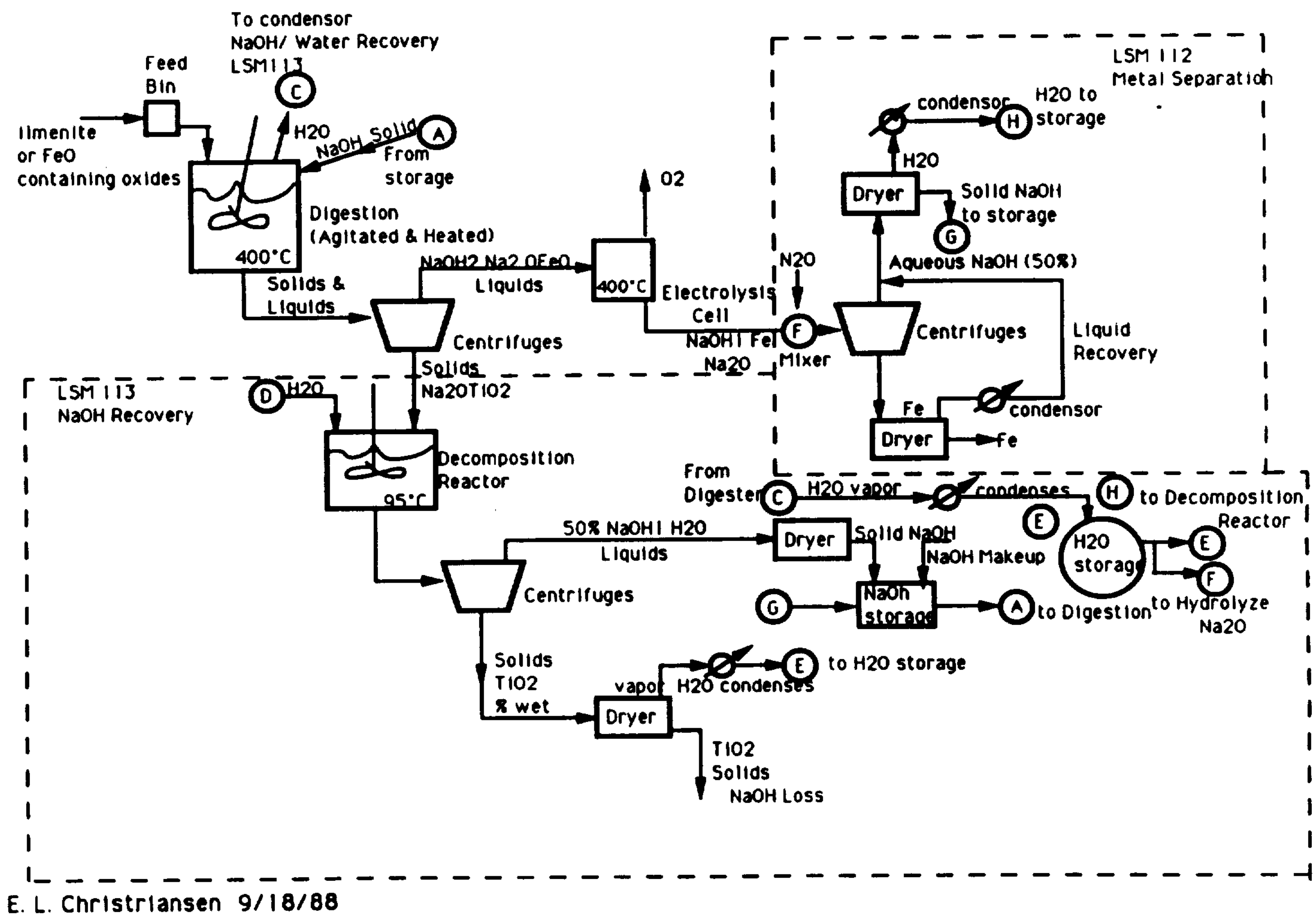


Figure 3.5-7B.-Lunar Oxygen From NaOH Electrolysis

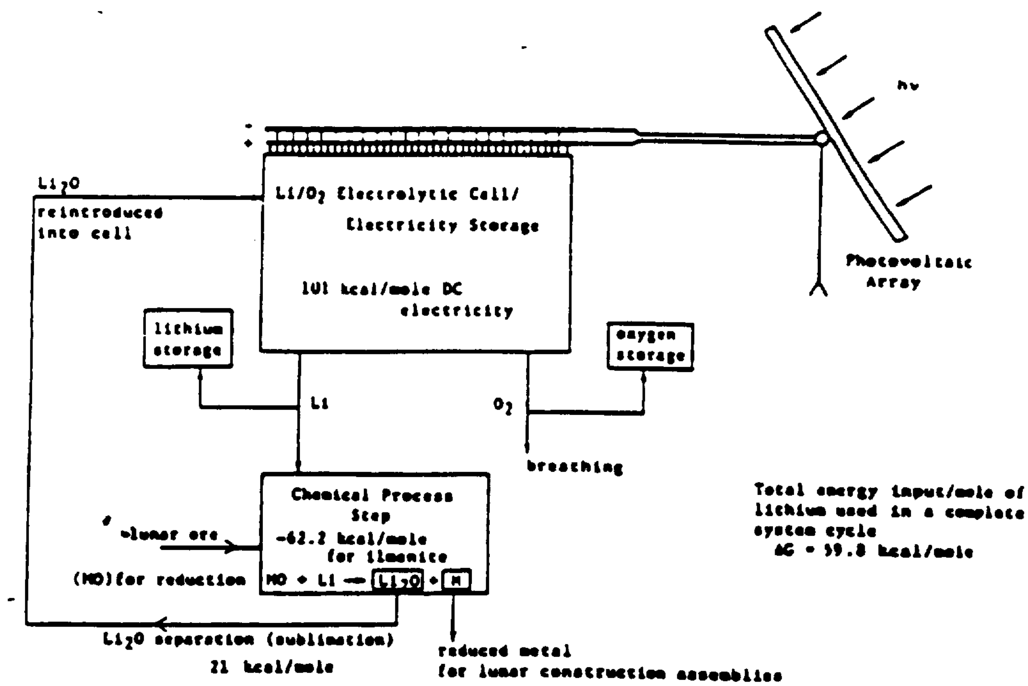


Figure 3.5-8. - Indirect Electrochemical Reduction with Lithium

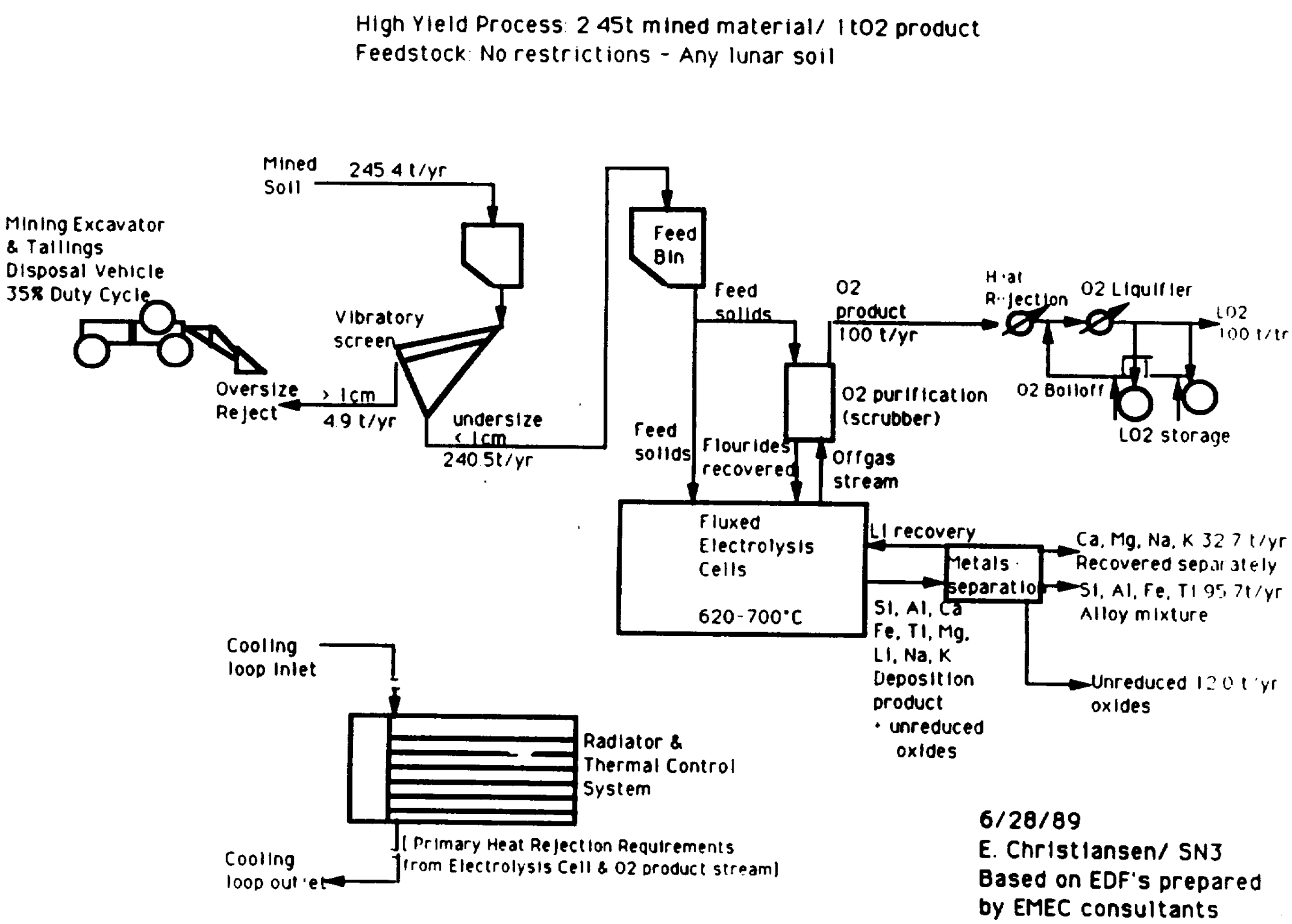


Figure 3.5-9.-Electrolysis of a Molten Salt (Fluxed Electrolysis) Schematic

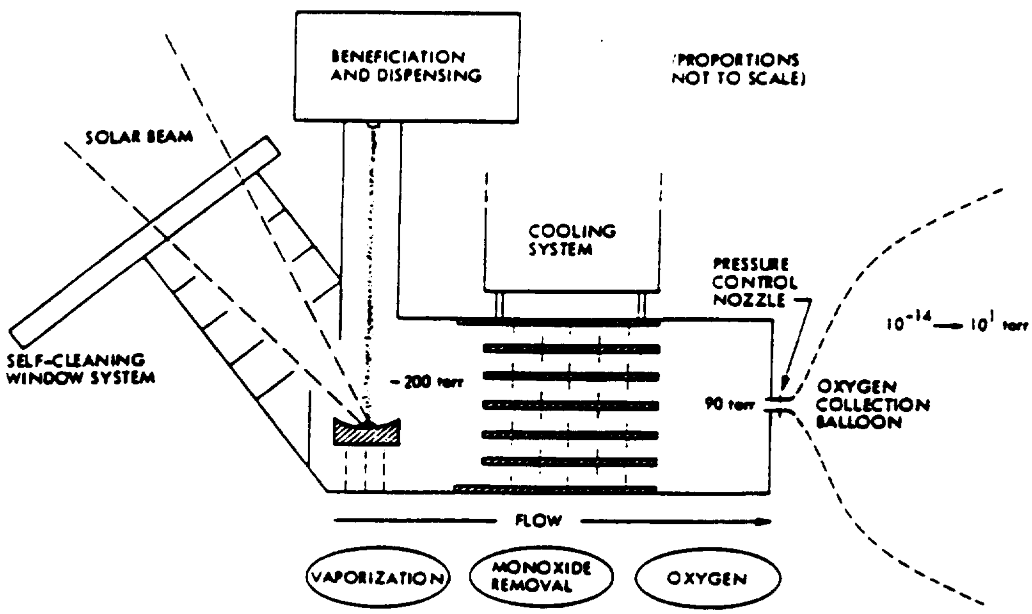


Figure 3.5-10.-Vapor-Phase Reduction Process Schematic

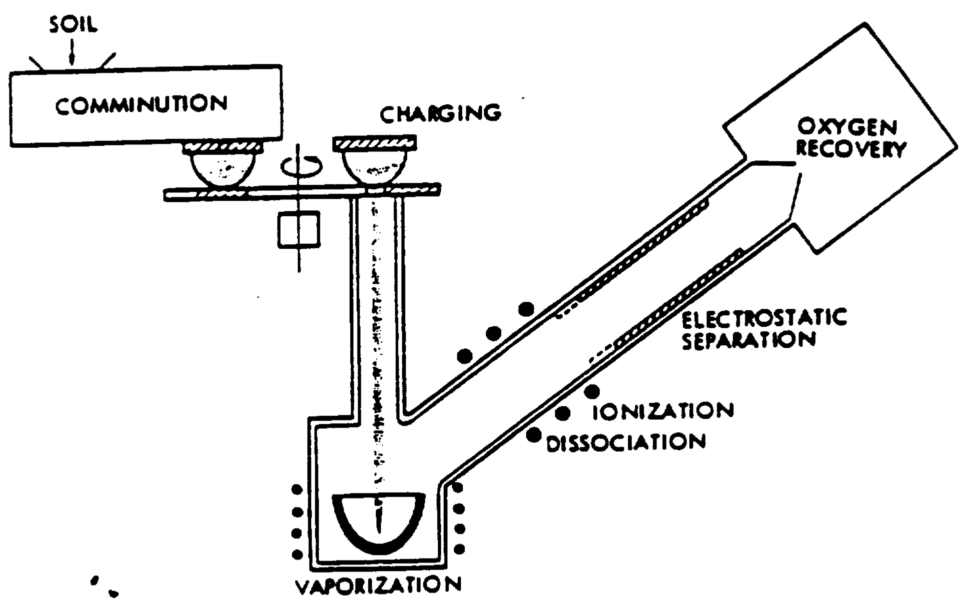
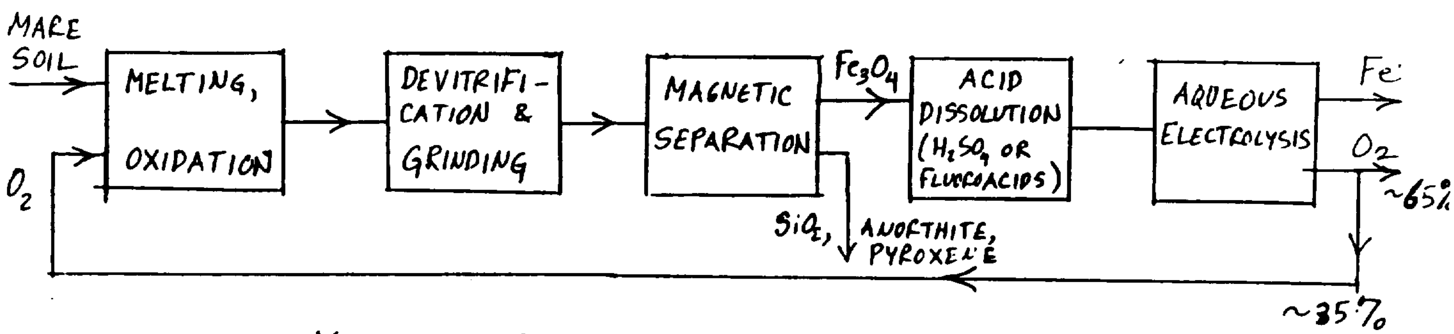


Figure 3.5-11.-Ion Separation Process Concept



MAGMA PARTIAL OXIDATION

MELTING LUNAR SOIL IN PRESENCE OF AIR OR OXYGEN CONVERTS CRYSTALLINE OR GLASSY IRON SILICATES TO SILICA-FREE MAGNETIC IRON OXIDE (Fe_3O_4) IN 90% OR BETTER YIELD.

THIS FRACTION CAN BE MAGNETICALLY SEPARATED FROM REMAINING CRYSTALLINE PHASES, DISSOLVED IN ORDINARY MINERAL ACIDS, AND ELECTROLYZED TO YIELD IRON AND OXYGEN.

Figure 3.5-12.-Magma Partial Oxidation

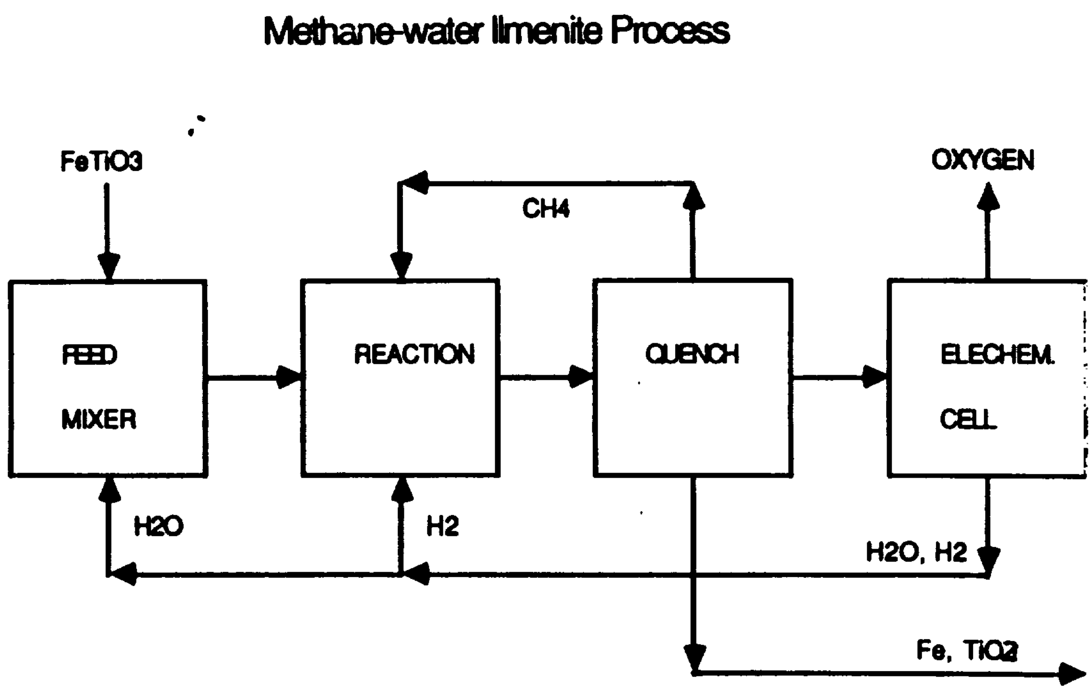


Figure 3.5-13.-Methane-Water Ilmenite Process

SECTION 4

Plant Engineering

4.1 PURPOSE

The purpose of the Plant Engineering Group was to define as well as possible a typical processing plant for In Situ Resource Utilization (ISRU) and to identify the issues that need resolution for the complete development of such a plant. Conceptual designs were sought rather than details of the oxygen production process, except where details such as reagents and imported materials were important for design. Emphasis was placed on lunar processing because another group was assigned to consider Martian applications. Also, the Moon is the obvious first step for ISRU.

4.2 APPROACH

The approach used was to first lay out a basic block diagram for a generic ISRU plant with all of the necessary components. Then the issues associated with each block were discussed. Candidate systems, such as the hydrogen reduction of ilmenite, were not considered. It was thought that a generic approach would be better. It was recognized, however, that the detailed considerations of each of these systems would require some input from the candidate systems.

After each of the blocks were identified for the generic plant, a block by block needs assessment was performed to determine issues associated with such matters as instrumentation and control, tribology, thermal management, and mechanical systems. Though time constraints prevented the identification of similar terrestrial systems and equipment which could be suitably modified, Element Definition Forms (EDFs) were filled out.

4.3 GENERIC BLOCK DIAGRAM FOR A LUNAR ISRU PLANT

The basic plant block diagram has three components, Beneficiation, Processing, and Oxygen Separation and Storage. In the Ore Input and Beneficiation Block, received ore is processed to meet all other plant requirements. Outputs from this block include possible volatiles from heating the ore, acceptable ore for the oxygen plant process, and scrap not suitable for use by the rest of the plant. The Beneficiation Block output is sent to the Process Block. Oxygen is produced there, as well as other process materials such as refractories and metals. Oxygen is separated from the process product stream and liquefied for storage. Other gases which are removed may either be recycled back to the Process Block, as in the case of hydrogen reduction of ilmenite, or they may be sent off to other storage units. In all cases, electrical and heat energy production and removal will be required.

4.4 BENEFICIATION BLOCK

The components of the Beneficiation Block depend strongly on the type of process used. The baseline case which utilizes all possible elements starts with a rock feeder system which uses a vibratory or screw feeder with klinker removal. The roughly sized rock would then be ground and sized according to process requirements. The sized feed would then pass through a separator which separates according to the composition demanded by the process. For example, in the ilmenite reduction case, the separation would be magnetic. Following separation, the materials would most likely be

stored in a holding area. This is because the beneficiation process is expected to feed the oxygen production unit in batches rather than in a direct, continuous feed.

Departures from this complete system include the following:

- Regolith feed which would go directly to the sizing system with fine regolith; alternatively, a thermal aggregation process to coarsen ultrafine regolith could even be used.
- Melt type processing such as magma electrolysis would probably eliminate all steps of grinding, sizing, and composition determination since the magma electrolysis will use all materials.

Instrumentation and control issues include the following:

- Particle size determination using laser based optical systems or vibratory screens.
- Mass flow systems for solid flow.
- Chemical analysis which will be affected by the process tolerance and requirements.
- Temperature.
- Mass transport methods and thermal management of the solids.
- Dust control for the thermal surfaces and possibly the use of inflatable collars and coverings that protect critical components and joints.
- Mechanical drivers used for transporting materials and other types of actuation.

The principal issue which developed from the discussion about the Beneficiation Block was the need to remove sulfur from the feed ore. It was suggested that this requirement might result in a pre-process plant as large as the main process plant.

4.5 OXYGEN SEPARATION AND STORAGE

The Oxygen Separation Block requires that particulates be removed by upstream filtration (electrostatic filtration) if the hydrogen reduction of ilmenite is used. Depending on the type of separation unit used, the gas may have to be cooled. The separation processes which were considered include electrochemical (high and low temperature), Thermal Swing Absorption (TSA) using lunar zeolites, diffusional membranes, cyclone separation (no moving parts separation relaying on weight differences of the gases), and fractional distillation. Further heat exchange will have to be done on the discharge gas (that gas which does not contain oxygen) and possibly on the oxygen, depending on the liquefaction process used.

Instrumentation and control requirements are fairly conventional, including pressure, temperature, differential pressure, chemical composition, thermal management, and flow control. An expert system will be required for operation of the block. Valves will need to be configured for robotic removal (robotic handshaking) as well as for the special design requirements demanded by the lunar environment.

Issues include materials corrosion and environmental heat rejection. There are also issues of erosion in the incoming heat exchanger at the beginning of the Oxygen Separation Block.

4.6 OXYGEN PROCESS

The actual oxygen generation process received the least attention due to the great variety of possible processes. Five categories of oxygen processes were identified:

- Thermochemical Reduction - hydrogen reduction of ilmenite, carbothermal reduction, volatile recovery, and hydrogen sulfide reduction.
- Thermochemical redox - carbochlorination
- Thermochemical oxidation - fluorine exchange
- Reactive solvent - HF leaching
- Electrochemical - magma electrolysis
- A separate group (see Section 3) dealt with oxygen processes in much greater detail.

4.7 DETAILED REPORTS ON EACH BLOCK

The group decided that too many process possibilities existed for them each to be considered in detail within the time constraints of the workshop. Therefore, only one or two key process possibilities were considered by the small groups. A concern expressed was that there would be too much quantitative emphasis placed on the deliberations of the groups when there was insufficient time and data to provide the detailed information required by such a quantitative analysis. This point is stressed here to prevent the overuse of the results of the groups. The groups were told to pinpoint problem areas for the processes considered and to put as much "intelligence" as possible into the numbers. Issues covering lack of knowledge were stressed in the discussions. EDF's and ITBC's were developed as a part of this process and are found in the appendices.

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SECTION 5

Volatile Extraction

5.1 OVERVIEW

The task of Group 4 was to examine processes for the extraction of volatiles. As part of that, they discussed the kinetics and chemistry of various processes. They looked at the techniques and hardware which would be required for extracting volatiles. Various techniques were assessed, including thermal, microwave, and gas separation. Part of the assessment of techniques was to look at the advantages and disadvantages of the various processes. Comparison of proposed techniques with state-of-the-art terrestrial solutions constituted another part of the assessment process. Plant needs for thermal management, control, and so forth were discussed, along with recommendations for development. Finally, various elements of the extraction systems were reviewed.

The approach of the group was to first identify the volatiles which could be extracted from the Moon, Mars, Phobos, and Deimos. Next, the group defined an extraction system's primary and secondary stages or elements. The primary stages consist of acquisition, beneficiation, processing, species separation, and storage. Secondary elements include logistics, maintenance, transportation, power, control, and mechanical conveyances.

Third, volatile extraction for each planetary body was considered in the context of the various phases of the extraction system. For each phase, both a baseline and an alternate process or mechanism were selected. For example, the baseline choice for Martian volatile storage was liquid tanks, and the option was solid containers. Where appropriate, recommendations concerning the process or mechanism were made. Each planetary body was treated in this manner.

Following the discussion of specific extraction systems, candidates for research and additional scientific exploration missions were identified. Technical spinoffs which could emerge from this research and exploration were also identified and discussed.

5.2 AVAILABLE PLANETARY VOLATILES

Lunar volatiles which the group considered accessible were H2, H2O, He, CO2, CO, CH4, and N2. Volatiles which were ruled out were Ar and other rare gases, and halogens. The only volatiles which were considered available and useful on Mars were H2O and CO2, primarily from the atmosphere. Excluded Martian volatiles include H2, He, CO, SO2, O2, N2, CH4, halogens, Ar, and other rare gases. Potential alternate volatile sources on Mars are the polar ice caps and the soil at the equator. The latter can be drilled. Available volatiles on Phobos and Deimos include H2O, CO2, CO, H2, He, CH4 and HC. Halogens, SO2, O2, N2, NH3, Ar, and other rare gases were ruled out. The availability of H2O enables other processes, such as aqueous extraction.

5.3 THE VOLATILE EXTRACTION SYSTEM

As stated above in the Introduction, five basic steps comprise the extraction system: acquisition, beneficiation, processing, species separation, and storage. The sub-elements of the system include logistics, maintenance, transportation, power, control, and mechanical conveyance.

Acquisition can be by excavation or

by passive or active in-situ methods. Excavation methods include both the use of machinery and the use of physical devices such as explosives. Machinery considered includes bucketwheel/trencher mobile miner, dragline/rail, scraper/paddleloader, front-end loader, slinger/snowblower/ballistic sweeper, clamshell, backhoe, and auger. Physical devices other than explosives consist of electromagnetic, vacuum/screen, adsorption, and electrostatic. In-situ passive methods include heating or using selective microwaves in a dome. Active in-situ methods considered include the hot rototiller, the mole, the thermal or microwave hot clamshell, and the drill/heat/pump.

Several beneficiation methods exist. Those identified include screens, electrostatic, pneumatic, electromagnetic, cyclonic, grinding, ballistic, ultrasonic, density (for liquids), Wilfrey Table, and robotic handpicking.

Six processing/species separation methods were considered. These were thermal, microwave radiation, O2 combustion and vibration, crushing, ultrasonics, and chemical. Six thermal processing techniques were explored. One was a solid/solid heat exchanger using heat pipes. A second type of heat exchanger was a solid/gas type using a carrier gas. A third type of thermal process was direct solar. The other three types of thermal processes were microwave, RF heating, and laser. Simple cooling was one type of species separation technique. More complex techniques included diffusing membranes, adsorption, laser isotope, chemical, ionization/plasma, molecular sieves, and pressure swing adsorption.

Storage was examined in the context of physical states (solid, liquid, gas). Aboveground storage options include radiation shielding, tanks, containers. Underground storage is an alternative.

5.4 ACQUISITION OF VOLATILES

For the Moon, the primary or baseline acquisition method selected was the bucketwheel/mobile miner. The alternate method was the dragline.

For extraction of volatiles from Martian polar icecaps, a number of systems were considered. Machines discussed included a drill, heat, and pump system, a dragline/rail system, a bucketwheel/trencher, an auger, a scraper, and a front-end loader. Other types of systems which were considered included blasting and microwaves in a dome.

The major Martian volatile extraction concern was felt to be the need to acquire water. Polar caps consist mostly of ice and dust. The water/ice below that is permafrost. The project team would be aiming for the vein of water/ice. A precursor mission to establish the presence of water ice in various locations on Mars was considered by the session members to be an absolute necessity prior to solidifying final plans.

The chosen baseline Martian acquisition system is the drill, heat, and pump process. The primary option is the mobile miner, which is a system that can process water off the surface of the planet. The primary recommendation for volatile extraction on Mars was to compile more data on Mars' atmosphere because that is a "knowable" factor.

Several choices were examined for acquisition on Phobos and Deimos. The first was a fixed dragline/rail system (driven by the lack of any significant gravity). A hot clamshell was another mechanical system. The mole, underground auger and the anchor dragline were three other

mechanical extraction devices. An electromagnetic device was one non-mechanical system. Three non-mechanical, passive systems were microwave, heater with dome, and boiling with a dome (a variant of the heater).

The selected baseline was a combination dragline/rail device fixed with a hot clamshell system. This combination will heat material and drive off gases. A microwave clamshell with a rototiller and a mole system were two alternative acquisition systems.

5.5 BENEFICIATION

The selected lunar beneficiation method was the screen. The primary option was an alternate version of the screen which uses electrostatic attraction/repulsion as a sizing method.

If the baseline Martian acquisition system is drill, heat, and pump, as described above, the attendees felt that the question of a beneficiation process might not have any validity. However, they felt that if filtering dirty water fell under the rubric of beneficiation, then a screen would be employed for that process. If the mobile vehicle processor option is to be used, then heat would be the beneficiation method because heat would melt the product and boil it off. For the mobile processor, the question arises about what is being mined: ice or water bound up in chemical form.

Questions which arose regarding beneficiation on Phobos and Deimos centered around the choice of acquisition method. If the concept of dragline/rail/hot clamshell is the baseline, the primary question was what beneficiation concept was valid. However, two processes were identified as possibilities. One process consists of scooping, lifting, and vibrating with a device that has a half clamshell on the bottom. The other process is a simple grizzly, which is a large sieve that eliminates large items and passes the smaller ones. The baseline decision was to use the grizzly; no option was selected.

5.6 SPECIES SEPARATION AND PROCESSING

Eight techniques were identified as possibilities for lunar species separation. Liquefaction, diffusing membranes, adsorption, and laser isotope were four techniques. The remaining four were chemical, ionization/plasma, molecular sieves, and pressure swing adsorption.

For lunar processing, the baseline choice was a combination of a liquefaction system and diffusing membranes. This choice of diffusing membranes was justified on the basis of their possible low cost and workability. However, while it is known that the diffusing membranes will work, it is not known how well. The cooling system will also work, but is considered to be costly. Two secondary alternatives to this combination were adsorption and chemical separation. The utilization of only diffusing membranes was a tertiary alternative.

For the case of Mars, separation processing is considered to be of major importance due to the large amount of dissolved material to be encountered. Several techniques were discussed. They are: distillation, crushing, ultrasonics, O2 combustion, X-ray radiation, vibration, chemical, filtration, and floculation (electrochemical). The baseline choice was to use filtration coupled with distillation, and the primary option was to use a chemical/ion exchange.

On Phobos and Deimos, a most

important consideration is the zero-gravity setting. Another question is whether the work will be done on the natural satellites or on an artificial one. A third concern to be kept in mind is the goal of separating CO_2 from H_2O. The separation processes considered were: liquefaction, diffusing membranes, adsorption, laser isotope, chemical, ionization/plasma, molecular sieves, and pressure swing adsorption. The baseline choice was the liquefaction process, with diffusion and molecular sieves as alternatives.

5.7 STORAGE

Hydrogen is the major volatile of interest on the Moon. Storage techniques include storage through chemical bonding (metal hydrides or water), and storage in the solid (chemical), liquid (LH_2), or gaseous (GH_2) state. Session attendees concluded that lunar hydrogen would obviously be dealt with in a liquid state, so the baseline storage method is liquid tankage. Gas would be supplied as needed. If the hydrogen is stored aboveground, then the tanks could be used as radiation shielding. In such a case, the tanks would be filled and transported from Earth in the early lunar base period. Using deep underground tanks for storage is another possibility.

The baseline was to use aboveground tanks for liquid hydrogen storage as radiation shielding. An option was high pressure gas. However, if high pressure gas is used, it would have to be in combination with liquid storage.

Solid and liquid water storage were considered for Mars. Liquid storage was chosen as the baseline for several reasons. First, liquid water is both more versatile and much easier to handle. Second, because end uses will center on fuel and supplying the colony, storage in liquid form is the best choice. Solid storage as ice is the only option.

Storage of H_2O on Phobos and Deimos could be either as a liquid or a solid, and storage of CO_2 could be in any of the three physical states. The decision was to store both in tanks

5.8 RESEARCH AREAS, SCIENCE MISSIONS, AND TECHNICAL SPINOFFS

5.8.1 Research Areas

Seven areas of research for volatile extraction were identified. One was molecular diffusion. Studies of both lunar samples and carbonaceous chondrites are thought to be needed, the latter as Phobos/Deimos analogs. Many questions remain about microwave extraction of regolith volatiles and materials degradation in space. Two additional areas of study involve physical characterization of the Moon and specifics of volatiles extraction methods.

A literature review and assessment is the first type of study necessary for molecular diffusion through different materials, including traps, sieves, and membranes. The knowledge gaps remaining after such a literature review could then be filled in with laboratory research. Estimated cost for the two steps: $50 to $150 K.

For $100 to $300 K, a number of lunar sample studies could be conducted. One needed study is better characterization of volatile rich materials. A second type of investigation would be on the crushing of breccias and subsequent volatile release. Release rate of H and He need to be known under different circumstances. Sulfur separation techniques need study. Finally, cold trapping of lunar soils and soil

simulants is an excellent research area.

Study of carbonaceous chondrites as analogs for Phobos and Deimos (and by extension, other asteroids) involves four steps. The first is a literature search. The second would be study of the release of volatiles by species as a function of temperature. Range of magnetic attraction for material handling is the third. The fourth concerns the range of mechanical strengths on these bodies, possibly obtained as ground truth from Phobos photos. Cost: \$25 to \$40 K.

Microwave extraction of volatiles involves several questions. Whether or not certain frequencies are superior to others is unknown. Also, it is unknown as to whether there is an "intensity ceiling" beyond which further increase in intensity is non-productive. The way in which penetration depth varies with frequency and/or intensity needs study. Comparisons between heating rates for different materials is an open research area. Simulants as well as actual lunar samples need study. The estimated cost for these studies is \$100 to \$150 K.

Materials degradation under "space conditions" can be evaluated. Of particular interest is hydrogen embrittlement in metals. The estimated cost of this study is \$100 K.

Physical characterization of the properties of the bulk regolith on the Moon are needed, as is characterization of the surface and maria. Diffusion of H_2 and He and other volatiles needs to be studied. Estimated cost of these studies is \$100 to \$150 K.

Research concerns which deal primarily with extraction involve three primary areas. One is the volatile-releasing response of lunar samples to radio frequencies, X-rays, and ultrasonic vibration. Estimated cost range for these studies is \$200 to \$300 K. Screening in vacuum, micro-g, and one-sixth gravity is a second research area, but cost was unknown. The third area is drilling for lunar volatiles. First, optimum locations for test drilling in terms of reservoir size, depth, and flow pressures must be determined. Second, develop a drilling method. The cost of the first drilling study was estimated at \$50 K; the cost of the second is thought to be approximatey \$200 K.

5.8.2 Science Missions

Precursor mission objectives were identified for the Moon, Mars, Phobos, and Deimos. They involve both remote and ground investigations.

For the Moon, a systematic return sampling program for both mare samples and core samples is considered necessary. Mare regolith should be seismically profiled. He distribution needs to be explored by means of a robotic explorer with a He "sniffer." Remote studies would include TiO_2 distribution, FeO distribution, polar studies via the lunar polar orbiter, and high resolution studies for the physical characterization of potential mining sites.

Mars precursor studies involve core samples at the poles and elsewhere. The matter of interest is H_2O presence and quantity vs. depth at the poles and vs. depth and latitude elsewhere.

On Phobos and Deimos, a demonstration of penetrator/strain relief (see attachment regarding penetrator) is desired. Volatile release vs. temperature needs to be studied. Rock size distribution and magnetic susceptability are two other areas of interest.

5.8.3 Technical Spinoffs

A variety of technical spinoffs are expected to occur as a result of these studies of volatile extraction techniques. Better membrane separation devices will result from studies of molecular diffusion. Improved technology for separating chemical species will also result.

Development of robotic explorers could occur. This will probably have impacts on terrestrial mining operations, none of which currently use remote operators for mining. Additionally, the research into the new mining and extraction methods will probably lead to more cost-effective terrestrial procedures. Improved construction materials are also likely to be a consequence of these studies. Better solar energy devices are almost certain to occur.

5.9 ANCHORED SURFACE MINING SYSTEM

This mining system is essentially a penetrator which would be shot into the asteroid (such as Phobos or Deimos) by a spacecraft. The penetrator would be capable of penetrating as deep as 20 meters into solid rock.

A dragline or rail system could be implemented with a mobile miner in the middle, which acts as a transport system that substitutes for the gravity. The anchoring system would be a key concern, hence the penetrator is designated as an "Anchored Surface Mining System." The estimated gain is one to two percent water at worst and up to 20% at best.

Important considerations involve the question of whether the asteroid's regolith or the asteroid's body will actually be mined. Some types of ferrosilicates could be "sure things."

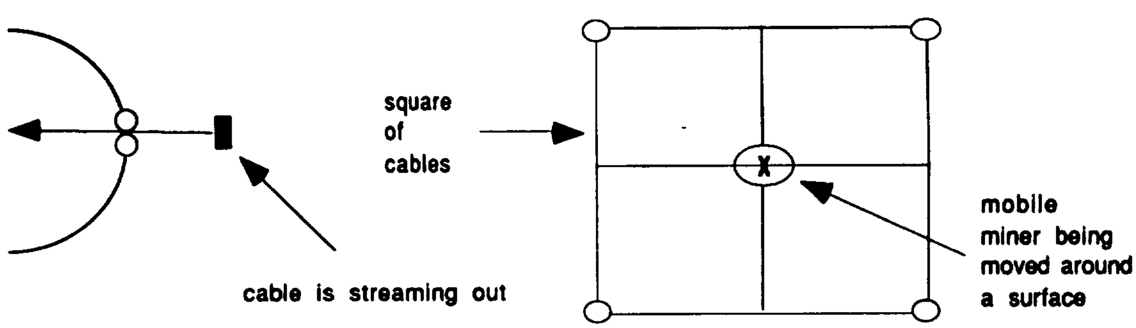

Figure 5.9-1.-Anchored Surface Mining System

SECTION 6

Metals, Ceramics, Semiconductors, Arable Soils, and Other Byproducts

6.1 OVERVIEW

The task of Group 5 was to look at substances that did not fall into the realms of the primary resources of oxygen or volatiles. This includes all the items named in the title, which will be referred to here as secondary materials or resources. Additionally, this group was tasked to look at the possible extent of integration of the manufacture of these materials with the manufacture of primary materials. That is, can a material be produced as a byproduct of an oxygen or volatile-production process or can it only be produced by a dedicated process?

The group was also asked to assess the state of the art of extraction/production of these materials. Parametrics such as sizing, block diagrams of the process, and thermal requirements were requested. Anticipated demands for these secondary substances were estimated, and directions for process development were indicated.

The procedure of Group 5 was to list and describe a broad range of possible secondary materials. This extensive list was then organized according to seven categories of demands. Those seven were: safety, construction, energy management, storage, agriculture, transportation, and waste processing. Demand, rather than abundance or accessibility, was selected as the criterion of characterization because of repeated demands for similar resources in diverse space development scenarios. Integration of secondary and primary materials production was considered along with need, because some resources have multiple uses. Often the final use of a material dictates its method of production. Table 6-I tabulates the uses of all the listed resources, thus giving some indication of the potential for integrating manufacturing processes.

6.2 SAFETY

There is a critical need for shielding in prolonged manned missions. The foremost need is for shielding against galactic and solar cosmic rays, which may be extremely energetic and, for prolonged missions, must be guarded against by shielding. A second need is for shielding against meteoroids and debris. Debris shielding may be particularly important in low-earth orbit.

Regolith is the resource identified for lunar base radiation shielding. It can simply be unprocessed regolith mounded or otherwise used in loose form to cover a structure. It can be bagged and then piled up over a structure. A third method of using regolith is to sinter it and form articles such as bricks. No specific resource was identified for radiation shielding in orbits around planetary bodies, but it has been recommended that radiation shields should not be carried up and down gravity wells (Willoughby, 1989).

6.3 CONSTRUCTION

Structures will be necessary to contain habitat and operational environments, and to support shielding. A variety of both metallic and non-metallic construction materials can be envisioned; these may be either directly produced or made as byproducts from O2 production schemes. Included with construction is the necessary site

preparation (foundations, roads, and dust control).

Both metals and non-metals were identified as resources for construction. Metals would include all those which can be obtained from the lunar regolith. Examples are iron, nickel, aluminum, silicon, and magnesium. Metal alloys such as silicon/aluminum from anorthite may also constitute lunar resources for construction.

The easiest non-metal to use is, of course, regolith in the form of sintered bricks. Glass, glassy ceramics, and crystalline ceramics are other non-metallic resources. Glass would probably be used in fiber or foamed form. Glass could be also combined with metals to form glass/metal foams.

Structural elements could be bonded by cements made from in situ resources. One type of suggested cement would use calcium oxide cement; another would use sulfur.

6.4 ENERGY MANAGEMENT

The topic of energy management includes energy production, energy storage, energy transmission (including heat transfer media), and waste heat rejection. Both the systems which are necessary for fulfilling these functions and elements of those systems were discussed in terms of the available lunar resources.

Energy production in space is likely to be electrical (either solar or nuclear produced) or thermal (solar). Chemical or mechanical forms of energy production are possible but are relatively cumbersome and tend to be driven by resources that are scarce and better used for other purposes in space.

Energy storage in space could be achieved by a broad range of thermal, electrical, mechanical, and chemical methods. One storage technique for mechanical energy which would operate particularly well in the space environment would be the use of flywheels spinning in the ~zero-air-friction "free" vacuum of space. Because of its moderate melting point and low conductivity, regolith could be used to store the thermal energy from the sun. Electrical energy could be stored through the use of sodium sulfide batteries or hydrogen-oxygen fuel cells.

Heat transfer media include helium, sulfur, sulfur dioxide, sodium, air, and water. Thermal radiative barriers may be composed of high-albedo, low-conductivity products. Heat radiators may be made of ferrous alloys comprised of iron and nickel.

Many elements are involved in energy transmission systems, including optical components and photovoltaics, electrical conductors, electrical insulators, magnets, and transformers. Group 5 members identified resources which could be put to use for these elements.

The optical components listed were reflectors (such as solar mirrors) and transmitters. Piping light through optical fibers was considered a means of bringing light into an underground or shielded area, and could be especially important to plant growth. Special treatment of anorthite may produce transparent optics. Photovoltaic resources were identified as silicon and (possibly) ilmenite.

Resources identified for electrical conductors fall into two categories. One category is wires and bars; the other is coatings. For wires and bars, magnesium, aluminum, and iron were identified. For coatings, titanium and magnesium can be

added to the preceding three. Flexible and rigid glasses would make excellent electrical insulators. Ceramics can also be used for insulation, as can oxides which occur as by-products of extraction processes. Iron and nickel are resources for magnets, and transformers can be made from iron and silicon.

6.5 STORAGE

Containers will be needed to hold liquids and gases. Smaller containers may be supplied from Earth, but large containers may have to be fabricated in space. These larger vessels could be made from rigid metals or from pliable glass/composite materials (woven glass fibers with a matrix of either glass or metal).

Metals which could be used to construct these storage vessels include aluminum and magnesium. Ferrous metals are also candidates.

6.6 AGRICULTURE

Agriculture on the Moon requires substrate materials and support structures, nutrients, water, and light sources (just as it does on Earth). Large-scale agriculture will probably become practical only in later, more mature stages of a lunar base, although it may be introduced at an early stage. This is because only in the later stages is it expected that the necessary large-scale closed-environment systems will be in operation.

Although hydroponics will likely be employed, a substrate material will be needed. Regolith was identified as the most accessible resource for it. However, precisely how the regolith would be used is open to question. The regolith must be clean, which could represent a difficult, although mechanical, problem.

One of the results of the group's efforts was the recognition that many of the elements that are surface-correlated in lunar pyroclastic deposits (e.g., S [used as SO4], Zn, Cu, Na, Cl) are necessary trace elements for agriculture. This is good news from the point of view of nutrients, since the the group noted the need for 14 minerals in plant growth and 21 minerals for human health. Further, the 14 minerals must be recycled 20 times over the plant lifetime. Water must also be recycled through the plants 20 times, thus requiring a system to detect contaminant build-up.

It was suggested that artificial light be used for plant growth. However, natural light could also be piped in using optical fibers (see energy management, above).

6.7 TRANSPORTATION

Building aerobrakes would be one of the two primary uses of lunar resources for transportation. The other is the manufacture of rocket fuels.

Sintered regolith is one variety of material which can be used to build aerobrakes. Ceramics are also envisioned as aerobrake media, and some experience exists with ceramics for this use. Slightly more unusual, perhaps, would be the use of glass foams. Finally, and most exotic, would be the use of organic materials such as bamboo. This broad range of aerobrake materials may all be realized, but studies of weight/insulation properties, thermal properties, and ablation behavior are required.

In-situ rocket fuels were envisioned primarily in terms of non-light gases -- metals and silanes. Other groups were directed to deal with the more

familiar light gas H2-O2 and CO-O2 rocket systems.

In terms of metallic rocket fuels, magnesium was considered by the group to be the best resource, followed by aluminum and then silicon. Other metallic fuel possibilities include iron, titanium, and calcium. Sulfur and carbon were also identified as possible fuel components. The group recognized that a tremendous variety of non-light gas fuels to be burned with oxygen can be envisioned, but they also recognized that practical experience with them is very limited. Thus many questions must be resolved about their use.

6.8 WASTE PROCESSING

This resource use category was considered by the group as a topic to be considered as processing schemes are developed, evaluated, and integrated. Wastes will be generated, and they must be dealt with to prevent poisoning of industrial processes, of habitats, and of planetary environments.

This is a particularly complex topic to deal with, because the methods used for waste processing are dependent on environmental factors. Consider, for example, the differing treatments of solid wastes on the Moon, on Mars, on asteroids, or at a Lagrange point. On the Moon, chemical reactions may occur with dry reducing regolith. Chemical reactions with wet oxidizing regolith may occur on Mars. On asteroids, the problem is one of uncontrolled drifting of solid wastes. An at a Lagrange point, wastes will accumulate from all nearby operations.

One suggestion the group made was to make use of non-vacuum environments, such as helium or oxygen, where possible. However, in these environments and at the various possible locations, it will be important to use the available resources in ways which minimize unwanted waste. Another factor which makes waste processing such a crucial part of integrated resource extraction systems is the desirability of combining processes so that the waste stream from one process is the feedstock stream of another.

TABLE 6-I.-RESOURCES AND THEIR USES

USES	RESOURCES							
	bulk regolith	bag regolith	sintered regolith	glass foams	glass compos.	glass ceramic	xline ceramic	"dry" cements
SAFETY								
1. radiation shields	X	X	X					
2. debris shields	X	X	X					
CONSTRUCTION								
1. Metal Forms								
2. Non-Metallic Forms					X	X	X	X
3. Site Preparation			X					
STORAGE VESSELS								
1. Metallic								
2. Glass & Comp.					X			
AGRICULTURE								
1. Substrates & supports	X		X	X		X	X	
2. Universal Nutrients	X							
3. Lite sources								
TRANSPORTATION								
1. Rocket Fuels								
2. Aerobrakes			X	X		X	X	
WASTE PROCESSING (to be considered as systems are developed and integrated)								

TABLE 6-I.-RESOURCES AND THEIR USES

USES	Fe	Ni	Al	Si	Mg	Ca	CaO	Na	S	SO2	Ti	SiH4
CONSTRUCTION												
1. Metal Forms	X	X	X								X	
2. Non-Metallic Forms												
3. Site Preparation												
STORAGE VESSELS												
1. Metallic	X	X	X		X							
2. Glass & Comp.												
TRANSPORTATION												
1. Rocket Fuels	X	X	X	X	X	X		X	X		X	X
2. Aero-brakes												

TABLE 6-I.-RESOURCES AND THEIR USES

USES	RESOURCES: bulk regolith	bag regolith	sintered regolith	glass foams	glass compos.	glass ceramic	xline ceramic	"dry" cements
ENERGY MANAGEMENT								
1)Electrical Conductors								
a. Wires & bars								
b. coatings								
2) Electrical Insulators	X	X		X	X	X	X	
3) Magnets								
4) Trans formers								
5) Heat Conductors								
a. stationary radiators								
b. heat transfer fluids								
6) Thermal Insulators	X	X		X				
7) Optical Elements								
8) Photo voltaics								

TABLE 6-I.-RESOURCES AND THEIR USES

USES	RESOURCES: bulk regolith	bag regolith	sintered regolith	glass foams	glass compos.	glass ceramic	xline ceramic	"dry" cements
9) Energy Storage								
a. thermal *	*							
b. electrical#								
c. mechanical@							@	
d. chemical	(probably best storage in H2-O2 fuel cells)							

* = heat storage in molten regolith

= Na-S used in storage batteries

@ = ceramic flywheels to store kinetic energy

TABLE 6-I.-RESOURCES AND THEIR USES

USES	RESOURCES										
	Fe	Ni	Al	Si	Mg	Ca	CaO	Na	S	SO2	Ti	SiH4
ENERGY MANAGEMENT												
1) Electrical Conductors												
a. Wires & bars	X		X		X							
b. coatings	X		X		X	X					X	
2) Elec Insulator							X					
3) Magnets	X	X										
4) Trnsformers	X			X								
5) Heat Conductors												
a. stationary radiators	X	X										
b. heat transfer fluids								X		X		

TABLE 6-I.-RESOURCES AND THEIR USES

USES	Fe	Ni	Al	Si	Mg	Ca	CaO	Na	S	SO2	Ti	SiH4
ENERGY MANAGEMENT												
6) Thermal Insulators												
7) Optical Elements			X	X								
8) Photovoltaics				X								
9) Energy Storage												
a. thermal*												
b. electrical#								X	X			
c. mechanical@												
d. chemical	(probably best storage in H2-O2 fuel cells)											

* = heat storage in molten regolith

\# = Na-S used in storage batteries

@ = ceramic flywheels to store kinetic energy

SECTION 7

Session Conclusions

7.1 FEEDSTOCK DEFINITION AND PRECURSOR SCIENCE AND EXPLORATION

The task of Group 1 was 1) to identify feedstocks and processes and 2) to identify the required precursor scientific research and exploration which must be accomplished before the various resources can be obtained. Group 1's method was to identify specific categories of resource, i.e., gas, metal, or other material, and then to identify feedstocks and processes for the extraction of that resource.

There were three categories of gaseous resources: oxygen, solar wind implanted volatiles, and magmatic volatiles. Magmatic volatiles include sulfur, chlorine, and others found in coatings on fire-fountain-produced volcanic glass beads and in certain soils. For both types of volatiles, the extraction mechanism of choice was thermal release. The feedstock for solar wind implanted volatiles is mature, high-ilmenite, mare basalt soils. The two feedstocks for magmatic volatiles are mare basalt soils and volcanic glass beads. Table 7.1-I summarizes feedstocks and process methods. Tables 7.1-II and 7.1-III summarize feedstocks and processes or uses. Table 7.1-IV lists required precursor studies.

At least seven areas require research for volatile extraction. Those are molecular diffusion, studies of lunar samples, studies of carbonaceous chondrites as Phobos/Deimos analogs, microwave extraction of volatiles in regolith, materials degradation in space, physical characterization of the Moon, and specifics of volatile extraction techniques.

Several science missions are needed. For the Moon, these are sample returns, lunar seismic profiling, volatile (and other resource) distribution, and remote sensing. For Mars, core samples are needed to investigate the presence of water. And for Phobos and Deimos, concerns are with rock size distribution, magnetic susceptibility, volatile release vs. temperature, and penetrator/strain relief.

Many technical spinoffs will result from this type of research. Most probable spinoffs include better membrane separation devices, improved chemical species separation techniques, development of robotic explorers, more cost-effective terrestrial mining and extraction methods, improved construction materials, better volatile recovery systems, and better solar energy devices.

7.2 GROUP 2: OXYGEN-EXTRACTION PROCESSES

The primary task of this group was to compare oxygen extraction processes in detail. Their procedure was to analyze 15 processes according to eight criteria. Numerical values were assigned to the criteria, and the summed totals represented a ranking of the various processes.

The 15 processes were as follows:

- hydrogen reduction of ilmenite
- carbothermal reduction
- volatile extraction
- hydrogen sulfide reduction
- carbochlorination
- fluorine exchange
- hydrofluoric acid leach
- magma electrolysis
- caustic electrolysis
- lithium reduction
- fluxed electrolysis
- vapor phase reduction

TABLE 7.1-I.- OXYGEN PROCESSES AND FEEDSTOCKS

Feedstock	Process				
	H-reduction of ilmenite	HF leaching	Vapor Pyrolysis	F-flux electrolysis	Magma electrolysis
High-Ti mare soils	X		X		
High-Ti mare basalts	X		X		
silicates		X	X		
mare soil			X	X	
low-Fe highland soil			X	X	
highlands anortho-sitic gabbro soil			X		X
all highlands soils			X		X
low-Fe mare soil			X		X

TABLE 7.1-II.- METAL PROCESSES AND FEEDSTOCKS

METAL	*PROCESS*	*FEEDSTOCK*
Fe, Ni, Co	Carbonyl	mature mare basalt soils
Fe, Ni, Co	magma electrolysis	hi-Fe basalt soils
Si	F-flux electrolysis	anorthositic gabbro highlands soils, lo-Fe mare soils
Si	magma electrolysis	anorthositic gabbro highlands soils, lo-Fe mare soils
Al	electrolysis	maria & highlands soils
Al	HF dissolution	any feedstock
Ti	?	maria soils, Apollo 11 and 17 sites

TABLE 7.1-III.-OTHER MATERIALS, THEIR USES, AND THEIR FEEDSTOCKS

MATERIAL	*USE*	*FEEDSTOCK*
glass	construction	anorthite, pyroclastics, rock wool
ceramics	construction, aerobrakes	highlands soils, waste streams from other processes
shielding	cosmic ray protection	regolith
agricultural substrate	plant growth	regolith - pyroclastics, KREEP-rich rocks

TABLE 7.1-IV.-PRECURSOR STUDIES

RESOURCE	*PRECURSOR NEED*
Oxygen	Mapping of Ti from both Earth and the LGO; evaluation & development of beneficiation methods; remote sensing of maria for ores and sites; verification of discovered ore bodies; evaluation of material for ease of beneficiation
solar-wind implanted volatiles	specific details on volatile concentrations in different soils, soil depths, and particle size groups
magmatic volatiles	study of particle coatings to quantify composition & comprehend origins; evaluation of S content in existing lunar samples; evaluation of S production from products of solar-wind thermal release
metals	extraction techniques for all metals; for iron, its occurrence from lunar sample studies
glasses	evaluation of data on uses of lunar glasses

- ion separation
- magma partial oxidation
- methane/water ilmenite reduction

The eight criteria by which these processes were gauged were: maturity, feedstock requirements, yield, reactant resupply, byproducts, complexity, reliability, and energy requirements. The scoring of each process was based on a scale from 1 to 5, in which 1 represented a low, undesirable value of the criterion in question and 5 represented a high, desirable value. For example, a score of 5 for the byproducts criterion meant many byproducts, whereas a score of 5 for the process complexity criterion meant two or fewer steps. The criteria of process simplicity, maintainability, yield, and resupply requirements were weighted double due to their importance. Finally, a minimum cutoff value of 37 was selected.

Eight processes equalled or exceeded this cutoff. The process which received the highest score (46) was fluxed electrolysis (also known as molten salt electrolysis). The other choices, in order of desirability, were: vapor phase reduction (42), hydrogen reduction of ilmenite (41), volatile extraction(41), carbothermal reduction (40), ion separation (40), magma partial oxidation (37), and magma electrolysis (37).

The fluxed electrolysis process received its high score due to its very high yield, very high process simplicity, high reactant recovery capability, high reliability, and capability of using any feedstock. It requires large amounts of energy, but because it can use any feedstock, actual mining requirements are significantly reduced over more well-known processes such as hydrogen reduction of ilmenite.

Vapor phase reduction ranked very high in terms of process simplicity, reactant recovery, and feedstock use, but its yield is much less than that of fluxed electrolysis. Hydrogen reduction of ilmenite has high simplicity and reliability, but few byproducts and stringent feedstock requirements. Volatile extraction is reliable, moderately simply, and

provides excellent reactant recovery, but its yield and byproducts are low. It has medium feedstock flexibility.

Feedstock flexibility for carbothermal reduction is high, and its simplicity is moderately high. However, its yield, reactant recovery, and byproducts are only medium. Ion separation is a simple process, with great feedstock flexibility and moderately high yield and byproducts. However, its reliability is low, and reactant recovery is moderate.

Magma partial oxidation scored moderately high on reliability, reactant resupply, and feedstock flexibility, but was medium to low in other criteria. Finally, magma electrolysis scored high in simplicity and reactant recovery, but only medium to low in other areas.

Plant equipment needed for these processes was identified and compared in terms of feedstock type, mass, average power, peak power, standby power, volume, estimated operational life, and resupply in tons/yr. A number of trade studies needed for further information on the processes were identified. Justifications for extraterrestrial propellant production for the Moon and Mars system were provided. Finally, each process which the group examined was described in detail.

7.3 PLANT ENGINEERING

Group 3 defined a typical ISRU processing plant, based on a functional analysis of block components. A basic ISRU plant was laid out in a block diagram, and the issues and concerns associated with each block were discussed.

The basic block components are Beneficiation, Processing, and Oxygen Separation and Storage. In the Beneficiation step, ore is received and processed to meet all other plant requirements. In the Processing Block, oxygen is extracted from the beneficiated material, along with any other resources which could be useful. These would include refractories or metals, for example. The extracted oxygen is liquefied for storage. Other gases may also be liquefied for storage or recycled back to the process block, depending on their use.

Specific oxygen processes were not discussed, but five process categories were identified:

- •Thermochemical Reduction - hydrogen reduction of ilmenite, carbothermal reduction, volatile recovery, and hydrogen sulfide reduction.
- •Thermochemical redox - carbochlorination
- •Thermochemical oxidation - fluorine exchange
- •Reactive solvent - HF leaching
- •Electrochemical - magma electrolysis.

7.4 VOLATILE EXTRACTION

Group 4 examined processes for the extraction of volatiles. Such extraction consists of five steps: acquisition, beneficiation, processing, species separation, and storage.

The group first identified the volatiles which are accessible on the Moon and in the Mars system. Next, for each of the five extraction steps, a primary and an alternate method was defined. Finally, research and precursor science missions necessary for volatile extraction and technical spinoffs from such research were discussed.

Tables 7.4.I and 7.4.II summarize baseline choices and major options for volatile extraction on the Moon, Mars, Phobos, and Deimos:

TABLE 7.4-I.-BASELINE CHOICES FOR VOLATILE EXTRACTION

	Moon	Mars	Ph/D/asteroids
Acquisition	Mobile Miner Bucketwheel	Drill, heat, & pump	Dragline/rail/ hot clamshell
Beneficiation	Screen	Screen	Grizzly
Processing	Thermal Heat Heat exchange pipes	Filtration/ distillation	Thermal Heat Heat exchange pipes
Separation	Cooling & Diffusion membrane	N/A	Cooling
Storage	Liquid tanks/ radiation	Liquid tanks	Liquid tanks

TABLE 7.4-II.-MAJOR OPTIONS FOR VOLATILE EXTRACTION

	Moon	Mars	Ph/D/asteroids
Acquisition	Dragline	Mobile Vehicle Processor	Microwave/ rototiller; clamshell -- inertia-active; mole
Beneficiation	Screen; electrostatic	Heat	Grizzly
Processing	Radiation; direct solar	ion exchange/ chemical	radiation
Separation	Diffusion membranes	N/A	Diffusion membranes; molecular sieves
Storage	Liquid tanks; radiation with chemical or high pressure gas storage	solid in container	liquid tanks

7.5 OTHER RESOURCES

Group 5's task was to examine resources for metals, ceramics, semiconductors, arable soils, and other materials. One aspect of this examination was the degree to which the production of these byproducts could be integrated with the manufacture of primary materials such as oxygen and volatiles. State of the art knowledge of production of these materials was to be described, where possible. The procedure was to list resources for these byproducts and then categorize them according to use for safety, construction, energy management, storage, agriculture, transportation, and waste processing. Because the use of a resource often dictates its method of production, concerns with manufacturing integration were also examined at this stage.

Safety requires shielding from galactic and solar cosmic rays, and the primary resource for this shielding is safety. Metallic and non-metallic resources were identified for construction purposes. Metallic resources include iron, nickel, aluminum, silicon, magnesium, and any other metals which the regolith can provide, as well as their alloys. Non-metallic construction resources

include sintered regolith bricks, glasses, glassy and crystalline ceramics, and cements made from in situ resources such as calcium oxide or sulfur.

Energy management is a broad topic, including energy production, distribution, storage, and waste heat rejection. Energy production will most likely to be either solar or nuclear. Both can produce electricity, but solar energy will be used for thermal energy. A broad range of thermal, electrical, mechanical, and chemical methods for energy storage exist, including friction-free flywheels, molten regolith, and fuel cells.

Numerous resources exist for energy transmission, which includes heat transfer. Heat transfer involves transfer media such as helium or sulfur, thermal radiative barriers such as regolith, and thermal radiators made from iron-nickel alloys. Optical components, photovoltaics, electrical conductors and insulators, magnets, and transformers are all needed for energy transmission. Sources for these elements of energy transmission systems are quite varied, including metals, glasses, ceramics, oxides, and silicon.

Three types of materials were identified as storage container resources. One type of material is metals, specifically aluminum, magnesium, and ferrous metals. Glasses are a second resource. The third resource is glass composites, where fiberglass is woven into a matrix of glass or metal.

For agriculture, the chief resources are regolith and sunlight. Regolith will provide not only a substrate for plant growth, but it is also believed to be a source of many of the necessary trace elements needed for plant and human health. Sunlight, piped through optical fibers, will provide the necessary light for photosynthesis. If artificial light is to be used, then an electrical source for it must be built.

Aerobrakes and rocket propellants are the primary transportation resources. Aerobrakes can be made from sintered regolith bricks or from ceramics. Propellants considered by Group 5 did not include H_2-O_2 and CO-O_2 because those were under discussion by other groups. The propellants Group 5 concentrated on involved metals and silanes, such as magnesium, aluminum, silicon, iron, titanium, calcium, sulfur, and carbon.

Waste processing is a very complex subject, due to the varied environments within the solar system. Also, the need to use a waste stream from one process as a feedstock stream of another makes waste processing a crucial part of the integration of manufacturing processes. However, this area was considered to be ripe for further development as knowledge of extraction processes increases.

APPENDIX A

PLENARY PAPER SUMMARIES

Speaker: Mike Gibson
Affiliation: Carbotek
Topic: Oxygen Production From Hydrogen Reduction of Ilmenite

The production of oxygen from the hydrogen reduction of ilmenite was described. First the process itself was described and compared to terrestrial analogs. Then process design considerations for 1000 tonne/yr LOX production were discussed.

The reaction is:

$$FeTiO_3 + H_2 \longleftrightarrow Fe + TiO_2 + H_20 \longleftrightarrow H_2 + (1/2)\ O_2$$

Ilmenite constitutes less than 10% weight of lunar soil, so a beneficiation step is required to process it. Efficient, automated, low-maintenance process system designs are required for lunar operations. Vacuum and 1/6 gravity must dictate designs for solids feed and withdrawal and gas-solid contacts. Terrestrial analogs of this process include fluidized iron ore reduction, limestone calcining and sulfide roasting, and coal gasification. Fluidized-bed, gas-solids contacting was considered important to each of these.

Many considerations are involved in process design. First, reactor design requires more data on reaction thermodynamics, kinetics and solids sintering, fluidized-bed, gas-solid contacting, standpipe solids transport, construction materials, and instrumentation and control/dynamics. Qualitative results on kinetics and sintering were described. Other research results which were described included standpipe behavior predictions and studies of bubble size and growth rate, and the effects of bubbles on gasification. Process design considerations involve beneficiated feed solids in the 20-200 micron range, substantial recycling of hydrogen, smooth solids feeding and withdrawal with minimal gas leakage, heat salvage from the spent solids, a refractory-lined vessel, and an avoidance of thermal cycling.

Equilibrium conditions and process design requirements therefore call for a continuous, staged fluidized bed reactor, a lock hopper/vacuum pump scheme to seal solids feed and withdrawal points, and a vapor-phase water electrolysis at reactor temperature. All these considerations are met in Carbotek's designs. (Adapted from viewgraph presentation at workshop.)

Speaker: I.N. Sviatoslavsky
Affiliation: University of Wisconsin
Topic: Mining Lunar Helium-3

Steps in the mining and extraction of He-3 from regolith, mass of equipment required, power required, and energy payback were discussed. The Lunar Miner Mark II design requirements were described. Prime considerations include deep regolith mining (three meters), minimizing impact on lunar surface by re-depositing regolith, and convenient gas handling. The mobile miner would use a bucket wheel excavator to excavate a wide trench of material which would then be processed by the miner. Rejected regolith would be deposited along the miner's sides and processed regolith would be ejected at the back to refill the trench. Empty gas cylinders placed along one side of the mining route by service vehicles would be picked up by the miner, filled, and placed on the other side of the trench. Service vehicles would retrieve the filled cylinders and take them to the condensing station.

Equipment parameters were defined, including those for the regolith heater, the gas collection system compressor, the cooling radiator, and the cryogenerator. Energy requirements for the miner's operations were estimated at 84 giga-Joules per kg of He3, and energy needed to separate other gas components from the He3 were estimated at 186 giga-Joules per kg. Total energy required to bring mining equipment and people to the Moon were estimated at 1983 giga-Joules. Total energy invested to obtain and transport one kg of He3 to Earth was reckoned at 2253 giga-Joules, but the energy expected to be released from the kg is 600,000 giga-Joules. Therefore the payback ratio is 266. To manufacture the fusion reactor, 5025 giga-Joules per kg of He3 of energy is required. If this energy is added to the costs, the payback ratio becomes 82 (i.e., 600,000/[5025 +2253] = 82).

Conclusions: 1) Obtaining lunar He3 appears to be both technically feasible and economically viable; 2) proposed procedures are state of the art, except for beneficiation 3) the mass of equipment needed from Earth is large, but will eventually be ameliorated by indigenous titanium; 4) the energy payback is about 80, providing real incentive for commercial investment; 5) byproducts can be used to resupply a permanent lunar base and other space establishments, thus significantly enhancing space exploration. (Adapted from workshop viewgraph presentation.)

Speaker: John S. Lewis
Affiliation: Lunar and Planetary Lab., Tucson, AZ 85712
Topic: Importance of Space Resource Utilization for Future Space Development

The cost of large-scale space activities was shown by a graph which compared a 1962 estimate of the cost in 1980 to lift a pound to LEO with the real price in 1980. The 1962 estimate of the 1980 cost was approximately $200; the real 1980 cost (in 1962 dollars) was closer to $1400.

To bring the cost of large-scale space activities down, at least three criteria must be met. First, the cost of launching payloads into LEO must be reduced. Second, the cost of building spacecraft must be decreased by the use of long series of production-line vehicles. Third, the cost of LEO activities must be reduced by using non-terrestrial materials instead of paying the energy penalties to bring materials up from Earth.

Criteria for non-terrestrial resource and process selection were identified: ore/resource abundance; high demand; purification process simplicity; process autonomy; process efficiency; mass/energy transportation efficiency; maximum use of low-grade thermal energy. LEO, GEO, lunar base, and Mars system were identified as sites of high-volume, low-tech material demand for propellant, life support fluids and gases, shielding materials, and structural materials. Resource availability dictates the materials appropriate to each location for shielding, structures, and life support, and of course propellants have specific uses at different locations. For example, propellants at LEO would be used to get to GEO, whereas propellants at GEO would be used for stationkeeping. GEO was not listed as needing life support fluids, and the Mars system does not require shielding or structural materials.

High demand space resources include water, ferrous native metals, shielding, refractory elements, energy, and carbonaceous materials. These resources are distributed unequally throughout the solar system, so the particular resource material and its accessibility varies. Some political and economic drivers for bases at the different locations were discussed, as were extraction processes. Strategic materials distribution was discussed in the context of space resources.

Speaker: Grant Heiken
Affiliation: Los Alamos National Laboratories
Topic: Overview of Lunar Materials-What We Know and What We Probably Don't Know

Lunar regolith, lava, and pyroclastic rocks were discussed and described. Apollo and Luna regolith sample sizes and content were discussed. The five basic particle types comprising regolith are: crystalline rock fragments, breccia, glass particles, mineral grains, and agglutinates. Most unique of these particles is the agglutinate, which consists of fragments of rock and glass bonded by heterogeneous glass droplets containing abundant, well-dispersed 30-100 Angstrom diameter iron droplets. They are rich in solar-wind implanted gases. Lunar regolith is heterogeneous, and its bulk composition reflects that of the local bedrock.

Lunar lavas comprise less than one percent of the lunar crust, but they are both very important and very accessible. Most interesting are the high-titanium basaltic lavas, sampled by Apollo 11 and 17 crews. These lavas are rich in ilmenite (FeTiO3), chromite (FeCr2O4), and troilite (FeS). Over the last ten years, ilmenite has been proposed as an important source of oxygen with iron, titanium, sulfur, and chromium as byproducts. High-titanium lavas make up about 20% of lava flows observed on the nearside, and ilmenite grains make up 10 to 20 percent of the volume of these lavas. But not all of the rocks contain easily accessible ilmenite.

Pyroclastic deposits have been interpreted as the products of fire fountains at vents of mare lava flows. They are completely glassy deposits, and may serve well as feedstock for lunar glass manufacture or as sintered blocks in construction. However, they may have even greater value as sources of sublimates from eruptions of now-extinct lava fountains. Those sublimates include sulfur compounds, Fe, Zn, Cl, Pb, Ge, Au, and Sb (?).

Substantially more data are needed for the Moon, including extensive surface explorations and deep-core drilling. Suggested exploration methods include photogeology, remote sensing, coring, and ground-penetrating radar. Volcanic vents should be explored for evidence of deposits of sublimates. Finally, once base sites have been selected, observations of thicknesses and lateral variations within the regolith and underlying rock units must be explored.

Speaker: Murray Hirschbein
Affiliation: NASA HQ
Topic: Pathfinder Program

Key program elements of Pathfinder are basic production methods for required materials, process engineering, raw material preparation, and pilot plant design. Basic production methods are of particular interest for oxygen, metals, and construction materials. Areas under study in process engineering include engineering methods development and component concept design. Raw material preparation concerns include simulant development and production, materials handling methods, and mining technology. Pilot plant design concerns include component hardware design and development, lunar pilot plant conceptual design, testbed design and development, and system studies.

Eight priorities exist. First, second, and third is the validation of oxygen production methods, construction material production, and metals production. A conceptual pilot plant design is the fourth priority. Volatile extraction is fifth, with ore concentration sixth. Special topics such as Mars constitutes a seventh priority, and eight is the development of large scale mining technology.

In terms of relative effort, 45% of resources is being invested in basic production methods for the various materials. Process engineering consumes 28%, with 14% each going to raw materials preparation and pilot plant design.

Speaker: Chris McKay
Affiliation: NASA/Ames
Topic: Overview of Martian System and Near-Earth Asteroids

Mars and the Moon are very different bodies. Subsequently, what works on the Moon and what works on Mars will be different. The "first foot in the door" on Mars will be the extraction of oxygen from the CO2 in the atmosphere. The emphasis on "mining" is misdirected; it is not simple, nor is it believable as a first step. In situ resource utilization (ISRU) will not drive space exploration. The mass gain factor, Γ, must be considered explicitly:

$\Gamma > 10^2 - 10^3$ (grams resource/gram power plant +)

An essential precursor at Mars for ISRU is the detection of subsurface water.

Available Martian resources were listed. Water comprises one percent by weight of soil and is 10-90 pr. μm in atmosphere. O2 as CO2 forms 95% of the atmosphere; 70-760 nanomoles/cm3 were released from the soil in Viking experiments. As a buffer gas, N2/Ar forms 5% of the atmosphere. As propellant sources, CO/LOX can be made from atmospheric CO2, and CH4/LOS can be made from CO2 and H2O.

Concentrations of various gases in the terrestrial and Martian atmospheres were compared; water availability on Mars and Martian dust storms were discussed. The energy required for extraction of Martian resources was estimated, and lists of materials which can be produced from Mars atmosphere and soil were presented. Potential use of these materials were identified. A Mars gas extractor design was presented.

Presenter: Larry Haskin
Affiliation: Dept. of Earth and Planetary Sciences, Washington Univ.
Topic: Constraints on Chemical Processing Concepts

Feedstock, process, product/residue, and energy and mass constraints were discussed, followed by a detailed discussion of six processes in terms of 18 criteria. The six processes were hydrogen leaching, electrolysis, carbothermal, vapor pyrolysis, hydrogen reduction of ilmenite, and fluxed electrolysis.

An obvious constraint on feedstocks is the existence of the feedstock in question. Preparation is another constraint: should the regolith be scooped up; should drilling or tunneling be used; should rock be ground up; should a sieve be used; is high-grade ore available? Another feedstock constraint is the tolerance of the process for variability; should composition therefore be monitored?

Several process constraints exist. They include equipment complexity, number of steps per product, needed terrestrial imports, need for reagents and their recovery and recycling, energy and feedstock efficiency, ease of input and output, and robustness and tolerance of the process.

Constraints on products and residues involve questions of further purification. Also, should the product be fabricated as needed or stored? Delivery to the consumer is a third constraint.

There are energy and mass constraints for all steps in ISRU. Those steps include mining and preparation of feedstock, factory input and output, processing, storage and delivery, and scaling from laboratory studies.

APPENDIX B

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APPENDIX C

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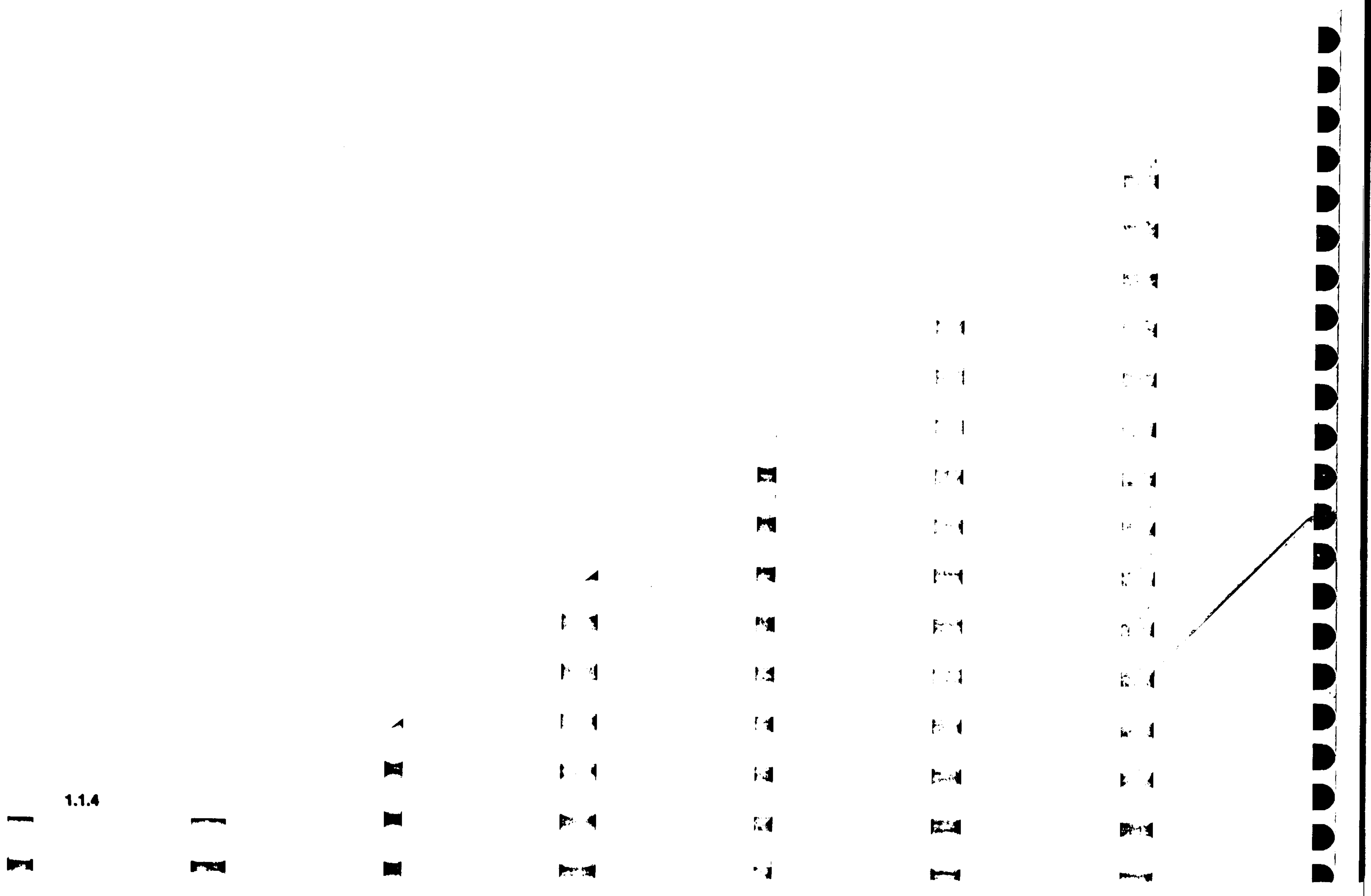

1.1.4

SUPPORTING DATA FOR CONSTRUCTION OPERATIONS ANALYSIS

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SUPPORTING DATA FOR CONSTRUCTION OPERATIONS ANALYSIS

Introduction

The construction of extraterrestrial surface bases will depend on careful coordination of crew and equipment during each phase of base development. A structured approach to the estimation of construction operations requirements was adopted for this study so that the large number of variables and assumptions could be more easily managed. The method used was intended to assist others in the space planning community as they review the progress made to date in the field of extraterrestrial construction and extend the work performed during this study. As a result of this effort, several categories of information were catalogued and organized into data bases that can be used to determine the resource allocations required for lunar and Mars construction activities. Hopefully, these data bases and the equations that relate them to one another and to specific construction projects will serve as basic building blocks for further investigation.

The construction operations analysis progressed in three major phases. The first phase was to identify and characterize the fundamental tasks and resources that would be used in the analysis. The second phase was to use the general task primitives to build and quantify specific task representations of the construction activities for the lunar scenario under investigation. The third and final phase was to combine the tasks in accordance with the constrained availability of resources to arrive at overall project durations. Each of these phases is described and documented in greater detail in the following sections.

Phase I. Fundamentals

The methodology used to quantify the operational requirements for the lunar and Mars surface missions is based on the premise that the majority of construction projects can be built up from a set of elemental construction tasks. A similar approach is being adopted for planning EVA work periods for the Space Station. The elemental tasks used by the EVA planners are referred to as *primitives* and are used to develop EVA timelines. Sequences of generic tasks are developed for specific EVA applications and then each task is quantified on the basis of previous EVA experience or the results of computer simulation.

A set of primitives for extraterrestrial construction operations was developed to facilitate the analysis of lunar and Mars construction requirements. Based on a review of the proposed lunar/Mars program objectives and the results of working sessions held during the NASA Mining and Construction Workshop, the study team determined that 25 basic tasks

were sufficient to characterize the construction activities under investigation. In addition to a set of basic tasks that can be used to define the construction work, a set of resources available to perform the work must also be identified. The set of construction resources used in this study was compiled with the help of the conceptual equipment design team from Pacer Works, Ltd. The tasks and resources listed in table 1 are the fundamental entities that were used in the operations analysis.

The information in table 2 indicates the mapping between tasks and resources. A group of resources is assigned to each task based on the functional capabilities of the equipment and crew. The excavation task, for example, is accomplished by an IVA crew member who teleoperatively manipulates a system that is composed of a reverse clam shell digging implement, a mobile work platform, an energy storage unit,.and a supervisory module. The assignment of IVA crew to any of the tasks listed in table 2 indicates that an IVA crew member would be directly involved as a teleoperator or remote task controller. The assignment of an IVA crewmember to monitor an EVA task is not indicated in table 2, but it is accounted for in the third phase of analysis through the assignment of overhead IVA resources for each task requiring EVA crew

The tasks are further characterized by a productivity measure that is expressed in units of work per hour. The unit of work identifies the parameter deemed to be the primary driver of task duration. The volume of regolith involved in an excavation task, for instance, is a significant driver of the time needed to complete the excavation task. For some tasks, the unit of work is a composite rather than a single measure of task difficulty. In the case of transporting bulk regolith, the unit of work is expressed as the volume of regolith to be moved (expressed in cubic meters) multiplied by the distance the regolith is to be transported (expressed in meters). The productivities indicated in table 2 reflect the estimated performance of the equipment involved and are designed to establish a set of benchmarks for this study. The sensitivity of overall project durations to these individual productivity measures is an issue that should be explored in future efforts

TABLE 1. - BASIC CONSTRUCTION TASKS AND RESOURCES

TASKS

Survey	Hardware Ingress
Excavate	Emplace Utilities
Remove Boulders	Inspect
Break Up Large Boulders	Set Anchors
Transport Bulk Cargo	Elevate Bulk Cargo
Trench	Connect/Disconnect
Grade	Activate/Test
Backfill	Repair/Startup
Offload	Transport Crew
Transport Pallets	Restation Machines
Emplace Large Items	Configure Machines
Emplace Medium Items	Set Up/Tear Down
Emplace Small Items	

RESOURCES (Code Letter)

EVA Crew	(A)	Regolith Bagger	(J)
IVA Crew	(B)	Grader Blade	(K)
Cargo Bin	(C)	Supervisory Module	(L)
Mobile Work Platforms	(D)	Servicing Module	(M)
Casters	(E)	Belt Conveyor	(N)
Crane Assembly	(F)	Energy Storage Unit	(O)
Reverse Clam Shell Digger	(G)	Unpressurized Rover	(P)
Robotic Arm	(H)	Mining Equipment	(Q)
Drill Implement	(I)		

TABLE 2.- PRODUCTIVITY AND RESOURCE ASSIGNMENTS FOR ELEMENTAL CONSTRUCTION TASKS

Elemental Task Name	Task ID	Unit of Work	Productivity*	Resources
Survey	001	points	12 points/hr	AMO
Excavate	002	volume (m3)	3 m3/hr	BDGLO
Remove Boulders	003	pieces	4 pieces/hr	BCDEH
Break Up Large Boulders	004	pieces	.25 pieces/hr	ADIMO
Transport Bulk Cargo	005	volume (m3)* distance (km)	2 m3 @ 4 km/hr	BCELO
Trench	006	volume (m3)	3 m3/hr	BDGLO
Grade	007	volume (m3)	3 m3/hr	BDKLO
Backfill	008	volume (m3)	6 m3/hr	BDGLO
Offload Pallets	009	pallets	.2 pallets/hr	ABDFLO
Transport Pallets	010	distance (km)	1 km/hr	BCELO
Emplace Large Pieces	011	pieces	.2 pcs/hr	ABDFLO
Emplace Medium Pieces	012	pieces	2 pcs/hr	ADHLO
Emplace Small Pieces	013	pieces	4 pcs/hr	AM
Hardware Ingress	014	volume	1.33 m3/hr	AB
Emplace Utilities	015	length (m)	500 m/hr	BDHLO
Inspect	016	points	4 pts/hr	AM
Set Anchors	017	points	1 pt/hr	BDILO
Elevate Bulk Cargo	018	volume (m3)	3 m3/hr	BDGNLO or BDJLO
Connect/Disconnect	019	points	4 pts/hr	AM
Activate/Test	020	systems	4 systems/hr	AM or BM
Repair/Startup	021	systems	.5 systems/hr	AM
Transport Crew	022	distance (km)	10 km/hr	DEO or PO
Restation Machines	023	distance (km)	4 km/hr	DELO or CELO
Configure Machines	024	functions	1 function/hr	AM
Set Up/Tear Down	025	systems	2 systems/hr	AM

* For tasks requiring EVA crew as a resource, the productivity is based on the amount of work that can be accomplished by a two-person EVA crew on average.

Phase II. Task Representations

The elemental construction tasks, resource assignments, and productivities were then used to calculate project durations and resource usage for the lunar scenario developed as a result of the FY89 case studies. Using the planetary surface systems lunar manifest as an indicator of the infrastructure to be installed and the availability of construction resources and materiel, specific task lists were developed for each period following a delivery of lunar hardware and/or crew to the lunar surface. Detailed, bottoms-up analysis was performed for all of the flight periods in the emplacement and consolidation phases (2003-2011). The major construction projects associated with the utilization phase were additional installations of modular LLOX plants and the construction of a low frequency radio telescope on the lunar far side. These projects were quantified using the aggregate results of the detailed analysis.

The tasks for each activity period were sized by the appropriate number of work units and a task duration was calculated based on the assumptions on productivity. The implementation of this phase of the analysis was well-suited to a spreadsheet environment where similar calculations could be easily replicated and referenced to a standard set of productivity data. A total of 16 spreadsheets (Lotus Symphony, Version 2.0) were developed to record and manipulate all of the task-related information. Printouts of the spreadsheets are provided on pages 7 through 25 as a reference for the assumptions made in regard to level of work required for each task.

Phase III. Aggregate Task Scheduling

The task durations calculated in the first phase of the analysis are expressed in hours. In order to determine the project duration in terms of mission days, the resources required for each project must be scheduled within the constraints imposed by human and machine performance, operational procedures, and the mission manifest. A project scheduling package (Computer Associates Super Project Expert, Version 1.1) was used to aggregate the tasks subject to the constraints imposed by the operating environment. The tasks and their individual durations were imported into the scheduling package as the starting framework for the scheduling portion of the analysis. Precedence relationships were established and entered for all of the tasks. The availability of each resource was entered into the project database and resource assignments were made according to the mappings developed in the first phase of the study.

The scheduling package provides several views of the project information, but the most important views for this effort were the Gantt charts and resource histograms. Gantt charts for each activity period are provide on pages 26 - 46.

A large number of assumptions were made in order to develop the elemental construction task database, the construction project task data and the availabilities for each resource. The assumptions can be organized into three major categories: machine performance, human performance, and operational procedures. The assumptions that are not explicitly called out in table 2 are listed in table 3.

TABLE 3 - ASSUMPTIONS USED IN CONSTRUCTION OPERATIONS STUDY

Average Slope at Site	4.7 degrees over 25 m
Allowable Average Slope	2.5 degrees over 25 m
Volume of Regolith to be Moved per Area to be Graded	.15 m3/m2
Angle of Repose for Disturbed Lunar Regolith	36 degrees
Angle of Excavated Wall Face (referenced to verticle)	30 degrees
Mass of Small Piece	m < 100 kg
Mass of Medium Piece	100 kg < m < 1000 kg
Mass of Large Piece	m > 1000 kg
Average Number of Systems to be Tested per Machine	4
Average Mass of Package to be Ingressed	100 kg
Traverse Distance from Landing Zone to Habitation Zone	5 km
Traverse Distance from Landing Zone to ISRU Zone	4 km
Width of Area Cleared for Roadways	10 m
Fraction of Systems Requiring Some Troubleshooting Activity	25 %
Average Duration of Troubleshooting Activity	2 hrs
Machine Duty Cycle, Daily	8 hrs per day
Machine Duty Cycle, Weekly	6 days per week
Machine Duty Cycle, Lunar Cycle	4 weeks per 4 week cycle
IVA Crew Duty Cycle, Daily	8 productive hrs per day
IVA Crew Duty Cycle, Weekly	6 days per week
EVA Crew Duty Cycle, Daily	6 productive hrs per day
EVA Crew Duty Cycle, Weekly	6 days per week
Airlock Capacity for Equipment Ingress	8 m3 per ingress cycle
Average Density of Equipment to be Ingressed	.18 t/m3

Year: 2004, Month: 2

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Activate Equip		0				
2	Offload MWP	1	1	2	med items	2	1
3	Test MWPs	2	1	8	systems	4	2
4	Offload Trucks	1	1	2	med items	2	1
5	Test Trucks	2	1	8	systems	4	2
6	Offload FCP Cart	5	1	1	lg item	0.2	5
7	Test FCP Cart	1	1	4	systems	4	1
8	Offload Implmts	3	1	10	sml items	4	3
9	Offload TC Cart	5	1	1	lg item	0.2	5
10	Test TC Cart	1	1	4	systems	4	1
11	Offload Liqf Tks	5	1	1	lg items	0.2	5
12	Test Liqf Tks	1	1	4	systems	4	1
13	PVA/RFC		0				
14	Layout PVA/RFC	1	1	12	points	12	1
15	Surf Prep PVARFC		0				
16	Level PVA/RFC	20	1	60	cubic meters	3	20
17	Remove Sml Rocks	2	1	6	pieces	4	2
18	Remove Lg Rocks	2	1	0.6	pieces	0.25	2
19	Unload PVA-RFC	5	1	1	lg item	0.2	5
20	Xport PVA-RFC	1	1	300	meters	1000	1
21	Emplace PVA	2	1	4	med items	2	2
22	Emplace RFC	5	1	1	lg item	0.2	5
23	Anchor RFC	4	1	4	points	1	4
24	Connect PVA	2	1	8	points	4	2
25	Anchor PVA	4	1	4	points	1	4
26	Connect PVA/RVC	1	1	2	points	4	1
27	Inspect PVA/RFC	1	1	4	points	4	1
28	Test PVA/RFC	1	1	4	systems	4	1
29	Repair/Startup	2	1	1	systems	0.5	2
30	Final Prep/Clnup	1	1	250	square meter	250	1
31	Habitat		0				
32	Layout Hab Mod	1	1	12	points	12	1
33	Surf Prep Hab		0				
34	Level Hab Area	10	1	30	cubic meters	3	10
35	Remove Sml Rocks	1	1	3	pieces	4	1
36	Remove Lg Rocks	1	1	0.3	pieces	0.25	1
37	Offload Hab Mod	5	1	1	lg items	0.2	5
38	Xport Hab Mod	1	1	300	meters	1000	1
39	Emplace Hab Mod	5	1	1	lg items	0.2	5
40	Anchor Hab Mod	4	1	4	points	1	4
41	Unload ALock&TCS	10	1	2	lg items	0.2	10
42	Xport ALock &TCS	1	1	300	meters	1000	1
43	Emplace TCS	1	1	1	med items	2	1
44	Anchor TCS	4	1	4	points	1	4
45	Emplace Airlock	5	1	1	lg item	0.2	5
46	Connect Airlock	2	1	8	points	4	2
47	Anchor Airlock	4	1	4	points	1	4
48	Emplace Tent	2	1	4	med items	2	2
49	Anchor Tent	4	1	4	points	1	4
50	Inspect Hab Sys	3	1	10	points	4	3
51	Test Hab Sys	3	1	10	systems	4	3
52	Final Prep/Clnup	1	1	250	square meter	250	1
53	Utilities		0				

Year: 2004, Month: 2

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
54	Power Cable		0				
55	Layout Cab Trnch	1	1	4	points	12	1
56	Dig Cable Trench		0				
57	Excavate Trench	2	1	6.25	cubic meters	3	2
58	Remove Sml Rocks	1	1	0.625	pieces	4	1
59	Remove Lg Rocks	1	1	0.0625	pieces	0.25	1
60	Install Cables	1	1	100	meters	500	1
61	Cover Cables	1	1	12.5	cubic meters	100	1
62	Connect Cables	1	1	4	connections	4	1
63	Inspect Cables	1	1	4	points	4	1
64	Final Prep/Clnup	1	1	250	square meter	250	1
65	TCS Piping		0				
66	Layout Trench	1	1	3	points	12	1
67	Excavate Cab Trn		0				
68	Dig Piping Trenc	1	1	3.125	cubic meters	3	1
69	Remove Sml Rocks	1	1	0.3125	pieces	4	1
70	Remove Lg Rocks	1	1	0.03125	pieces	0.25	1
71	Install Piping	1	1	50	meters	100	1
72	Cover Trench	1	1	3.125	cubic meters	6	1
73	Connect Piping	1	1	4	connections	4	1
74	Inspect Piping	1	1	4	points	4	1
75	Final Prep/Clnup	1	1	250	square meter	250	1
76	Test All Systems	8	1	30	systems	4	8
77	Repair/Startup	14	1	7	systems	0.5	14
78	Stock Hab		0				
79	Xport Supplies	1	1	250	meters	1000	1
80	Ingress Supplies	49	1	65	sml items	1.33	49
81	Stow Supplies	16	1	65	sml items	4	16
82	Launch/Land Area		0				
83	Layout LL Area	2	1	20	points	12	2
84	SurfPrep LL Area		0				
85	Level LL Area	192	1	576.975	cubic meters	3	192
86	Remove Sml Rocks	14	1	57.6975	pieces	4	14
87	Remove Lg Rocks	23	1	5.76975	pieces	0.25	23
88	Blast Barriers	60	1	180	cubic meters	3	60
89	Anchor Pad Marke	1	1	1	points	1	1
90	Anchor Nav Becon	3	1	3	points	1	3
91	Science Area		0				
92	Layout Sci Area	1	1	12	points	12	1
93	Surf Prep Sci		0				
94	Level Sci Area	20	1	60	cubic meters	3	20
95	Remove Sml Rocks	1	1	0.6	pieces	4	1
96	Remove Lg Rocks	1	1	0.06	pieces	0.25	1
97	Cable		1				
98	Layout	1	1	17	points	12	1
99	Trench		1				
100	Dig Trench	2	1	4.6875	cubic meters	3	2
101	Remove Sml Rocks	1	1	0.046875	pieces	4	1
102	Remove Lg Rocks	1	1	0.004687	pieces	4	1
103	Lay Cable	1	1	500	meters	500	1
104	Cover Cable	5	1	31.25	cubic meters	6	5
105	Connect Cable	1	1	4	points	4	1
106	Inst Solar Obser	2	1	3	med items	2	2

Year: 2004, Month: 2

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
107	Insp Solar Obser	1	1	4	points	4	1
108	Test Solar Obser	1	1	4	systems	4	1
109	Inst Geo Station	1	1	5	sml items	4	1
110	Insp Geo Station	2	1	6	points	4	2
111	Test Geo Station	2	1	6	systems	4	2
112	Final Prep/Clnup	1	1	250	square meter	250	1
113	LLOX Demo Prep		0				
114	Install LLOXDemo	2	1	4	med items	2	2
115	Lay Pwr Cables	1	1	100	meters	500	1
116	Connect Cables	1	1	4	connections	4	1
117	Inspect LLOX POC	1	1	4	points	4	1
118	Test LLOX POC	1	1	4	systems	4	1
119	Final Prep/Clnup	1	1	250	square meter	250	1

Year: 2005, Month: 1

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Activate Equip		0				
2	Unload UP Rover	1	1	1	med item	2	1
3	Test UP Rover	1	1	4	systems	4	1
4	Offload Balance		0				
5	Lg Items	10	1	2	lg items	0.2	10
6	Med items	3	1	6	med items	2	3
7	Transport Cargo	1	1	300	meters	4000	1
8	Ingress Supplies	75	1	100	sml items	1.33	75
9	Stow Supplies	25	1	100	sml items	4	25
10	Optical Tel #1		0				
11	Emplace O T #1	2	1	3	med items	2	2
12	Connect OT # 1	2	1	6	points	4	2
13	Anchor OT #1	4	1	4	points	1	4
14	Test OT # 1	1	1	4	systems	4	1
15	Repair/Start up	2	1	1	systems	0.5	2
16	PVA		0				
17	Emplace PVA # 3	2	1	4	med items	2	2
18	Anchor PVA #3	4	1	4	points	1	4
19	Connect PVA # 3	2	1	8	points	4	2
20	Inspect PVA #3	1	1	4	points	4	1
21	Test PVA # 3	1	1	4	systems	4	1
22	Repair/Start up	2	1	1	systems	0.5	2
23	Cable		0				
24	Install Calbes	1	1	10	meters	500	1
25	Connect Cables	2	1	8	points	4	2
26	Cover Cables	1	1	0.625	cubic meters	6	1
27	M/E Radio Intrf		0				
28	Emplace MERI	2	1	4	med items	2	2
29	Anchor	4	1	4	points	1	4
30	Connect	1	1	4	points	4	1
31	Inspect	1	1	4	points	4	1
32	Test MERI	2	1	8	systems	4	2
33	Repair/Start up	4	1	2	systems	0.5	4
34	Final Prep/Clnup	1	1	50	square meter	250	1
35	Prep next phase		0				
36	Mon Teles 2 Prep		0				
37	Layout Telescops	1	1	12	points	12	1
38	Sur Prep Teles		0				
39	Level	5	1	15	cubic meters	3	5
40	Remove Sml Rocks	1	1	1.5	Sml rocks	4	1
41	Remove Lg Rocks	1	1	0.15	Lg Rocks	0.25	1
42	Earth Observator		0				
43	Layout EO	1	1	12	points	12	1
44	Surf Prep EO		0				
45	Level	5	1	15	cubic meters	3	5
46	Remove Sml Rocks	1	1	1.5	Sml rocks	4	1
47	Remove Lg Rocks	1	1	0.15	Lg Rocks	0.25	1
48	Road to LL Area		0				
49	Layout LL Road	28	1	333	points	12	28
50	Surface Prep		0				
51	Level LL Road	833	1	2500	cubic meters	3	833
52	Remove Sml Rocks	63	1	250	Sml rocks	4	63
53	Remove Lg Rocks	100	1	25	Lg Rocks	0.25	100

Year: 2005, Month: 7

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Activate Equip		0				
2	Unload P Uti Veh	5	1	1	lg item	0.2	5
3	Test P Util Veh	2	1	8	systems	4	2
4	Unload Tun Ramp	5	1	1	lg item	0.2	5
5	Test Tun Ramp	1	1	4	systems	4	1
6	Unload Pwr Tlr	5	1	1	lg item	0.2	5
7	Test Pwr Tlr	1	1	4	systems	4	1
8	Unload FC P Cart	5	1	1	lg item	0.2	5
9	Test FC P Cart	1	1	4	systems	4	1
10	Unload TC Carts	5	1	1	lg item	0.2	5
11	Test TC Carts	1	1	4	systems	4	1
12	Unload Lab Tlr	5	1	1	lg item	0.2	5
13	Unload Rover	1	1	1	med item	2	1
14	Test Rover	1	1	4	systems	4	1
15	Liq Plant/Tanks		0				
16	Emplace P/Tanks	1	1	1	med items	2	1
17	Test Plant/Tanks	1	1	4	systems	4	1
18	Emplace Tents	2	1	4	med items	2	2
19	Unload Balance		0				
20	Lg items	15	1	3	lg items	0.2	15
21	Med items	1	1	1	med items	2	1
22	Transport Bal	1	1	300	meters	1000	1
23	Telescopes		0				
24	Emplace Teles	3	1	6	med item	2	3
25	Anchor Tele	8	1	8	points	1	8
26	Connect	2	1	8	points	4	2
27	Test Telescopes	2	1	8	systems	4	2
28	Repair/Start up	4	1	2	systems	0.5	4
29	Final Prep	1	1	100	square meter	250	1
30	Lab Trailer		0				
31	Transport Lab Tr	1	1	300	meters	4000	1
32	Emplace Lab Tlr	5	1	1	lg item	0.2	5
33	Conn Lab Tlr	3	1	10	points	4	3
34	Test Lab Tlr	1	1	5	systems	4	1
35	Ingress Bio Lab	28	1	37	items	1.33	28
36	Emplace Biomed L	9	1	37	items	4	9
37	Ingress Sci Lab	9	1	12	items	1.33	9
38	Emplace Sci Lab	3	1	12	items	4	3
39	Earth Observator		0				
40	Emplace EO	5	1	1	lg item	0.2	5
41	Anchor EO	4	1	4	points	1	4
42	Inspect EO	1	1	4	points	4	1
43	Test EO	1	1	4	systems	4	1
44	Repair/Start up	2	1	1	systems	0.5	2
45	Prep next phase		0				
46	SP-100 Prep		0				
47	Layout SP-100	1	1	12	points	12	1
48	Surf Prep SP-100		0				
49	Level	9	1	26.49375	cubic meters	3	9
50	Remove Sml Rocks	1	1	2.649375	sml rocks	4	1
51	Remove Lg Rocks	1	1	0.264937	lg rocks	0.25	1
52	Excavate		0				
53	Dig Hole	8	1	24	cubic meters	3	8

Year: 2005, Month: 7

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
54	Remove Sml Rocks	1	1	2.4	sml rocks	4	1
55	Remove Lg Rocks	1	1	0.24	lg rocks	0.25	1
56	Cable for SP 100		0				
57	Layout Cab SP100	1	1	17	points	12	1
58	Dig Trench		0				
59	Trench	10	1	31.25	cubic meters	3	10
60	Remove Sml Rocks	1	1	3.125	sml rocks	4	1
61	Remove Lg Rocks	1	1	0.3125	lg rocks	0.25	1
62	Road To Nuc Plts		0				
63	Layout	6	1	67	points	12	6
64	Surface Prep		0				
65	Level	167	1	500	cubic meters	3	167
66	Remove Lg Rocks	13	1	50	sml rocks	4	13
67	Remove Sml Rocks	20	1	5	lg rocks	0.25	20
68	LL Area		0				
69	Layout	5	1	60	points	12	5
70	Surface Prep		0				
71	Level	144	1	432.7312	cubic meters	3	144
72	Remove Sml Rocks	11	1	43.27312	sml rocks	4	11
73	Remove LG Rocks	17	1	4.327312	lg rocks	0.25	17
74	Blast Barriers	220	1	660	cubic meters	3	220

Year: 2006, Month: 1

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Unload Lander	10	1	2	lg items	0.2	10
2	Xport to Base	5	1	5000	meters	1000	5
3	Ingress Cons	84	1	112	sml items	1.33	84
4	Stow Consumables	27	1	108	sml items	4	27
5	Emplace Sc Resup	1	1	4	sml items	4	1
6	SP-100		0				
7	Emplace SP-100	5	1	1	Lg item	0.2	5
8	Emplace Cable	1	1	500	meters	500	1
9	Cover Trench	5	1	31.25	Cubic meters	6	5
10	Connect SP-100	1	1	5	points	4	1
11	Inspect SP-100	3	1	10	points	4	3
12	Test SP-100	3	1	10	systems	4	3
13	Repair/Start up	4	1	2	systems	0.5	4
14	Final Prep/Clnup	1	1	250	square meter	250	1
15	Prep Next Phase		0				
16	Inf Hab Prep		0				
17	Layout Inf Hab	1	1	12	points	12	1
18	Excavate Inf Hab	667	1	2000	cubic meters	3	667
19	Remove Sml Rocks	50	1	200	sml rocks	4	50
20	Remove Lg Rocks	80	1	20	lg rocks	0.25	80
21	IR Tele Prep		0				
22	Layout IR Tele	1	1	12	points	12	1
23	Surface Prep		0				
24	Level	5	1	15	cubic meters	3	5
25	Remove Sml Rocks	1	1	1.5	sml rocks	4	1
26	Remove Lg Rocks	1	1	0.15	lg rocks	0.25	1
27	UV Tele Prep		0				
28	Layout UV Tele	1	1	12	points	12	1
29	Surface prep		0				
30	Level	5	1	15	cubic meters	3	5
31	Remove Sml Rocks	1	1	1.5	sml rocks	4	1
32	Remove Lg Rocks	1	1	0.15	lg rocks	0.25	1
33	Rd To ISRU Area		0				
34	Layout	2	1	24	points	12	2
35	Surface Prep		0				
36	Level	5	1	15	cubic meters	3	5
37	Remove Sml Rocks	1	1	1.5	sml rocks	4	1
38	Remove Lg Rocks	1	1	0.15	lg rocks	0.25	1

Year: 2006, Month: 7

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Inf Hab		0				
2	Offload Inf Hab	5	1	1	lg items	0.2	5
3	Xport Inf Hab	5	1	5000	meters	1000	5
4	Emplace Found.	1	1	1	med item	2	1
5	Anchor Found		1	8	points	27	
6	Emplace Inf Hab	5	1	1	lg item	0.2	5
7	Inflate Hab	20	1	20	hours	1	20
8	Conn to Found.	2	1	6	points	4	2
9	Conn Power	1	1	4	points	4	1
10	Offload Tunnel	5	1	1	lg item	0.2	5
11	Xport Tunnel	5	1	5000	meters	1000	5
12	Emplace Tunnel	5	1	1	lg item	0.2	5
13	Connect Tunnel	2	1	6	points	4	2
14	Anchor Tunnel	4	1	4	points	1	4
15	Inspect Hab/Tunn	3	1	10	points	4	3
16	Test Hab/tunnel	1	1	4	systems	4	1
17	Repair/Startup	2	1	1	systems	0.5	2
18	Backfill Hab	167	1	1000	cubic meters	6	167
19	IR Telescope		0				
20	Emplace IRT	2	1	4	med items	2	2
21	Connect	2	1	6	points	4	2
22	Anchor	4	1	4	points	1	4
23	Inspect	1	1	4	points	4	1
24	Test IRT	1	1	4	systems	4	1
25	Repair/Startup	2	1	1	systems	0.5	2
26	Final Prep	1	1	50	square meter	250	1
27	UV Telescope		0				
28	Emplace UVT	2	1	4	med item	2	2
29	Connect	2	1	6	points	4	2
30	Anchor	4	1	4	points	1	4
31	Inspect	1	1	4	points	4	1
32	Test UVT	1	1	4	systems	4	1
33	Repair/startup	2	1	1	system	0.5	2
34	Final Prep	1	1	50	square meter	250	1
35	Road Nuc to ISRU		0				
36	Layout	4	1	50	points	12	4
37	Surface Prep		0				
38	Level	167	1	500	cubic meters	3	167
39	Remove Sml Rocks	13	1	50	sml rocks	4	13
40	Remove Lg Rocks	20	1	5	lg rocks	0.25	20

Year: 2007, Month: 1

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Offload Lander	5	1	1	lg item	0.2	5
2	Xport Cargo	5	1	5000	meters	1000	5
3	Stow Consumables	25	1	100	sml items	4	25
4	TCS Const Hab		0				
5	Offload TCS	5	1	1	lg item	0.2	5
6	Xport TCS	20	1	5000	meters	1000	20
7	Emplace TCS ext	5	1	1	lg item	0.2	5
8	Anchor TCS ext	8	1	8	points	1	8
9	TCS Ingress	11	1	14	items	1.33	11
10	Emplace TCS int	4	1	14	sml items	4	4
11	Piping Trench		0				
12	Dig Trench	2	1	6.25	cubic meters	3	2
13	Remove sml rocks	1	1	0.625	sml rocks	4	1
14	Remove lg rocks	1	1	0.0625	lg rocks	0.25	1
15	Emplace Piping	1	1	100	meters	100	1
16	Cover Trench	1	1	6.25	cubic meters	6	1
17	Connect Piping	1	1	4	points	4	1
18	Inspect TCS	10	1	38	points	4	10
19	Test TCS	29	1	116	systems	4	29
20	Repair/Startup	58	1	29	systems	0.5	58
21	Rd ISRU To L/L		0				
22	Layout	3	1	40	points	12	3
23	Surface Prep		0				
24	Level	667	1	2000	cubic meters	3	667
25	Remove Sml Rocks	50	1	200	sml rocks	4	50
26	Remove Lg Rocks	80	1	20	lg rocks	0.25	80

Year: 2007, Month: 7

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Activate Equip		0				
2	Offload Pwr Tlr	5	1	1	lg item	0.2	5
3	Test Pwr Tlr	1	1	4	systems	4	1
4	Offload PUV # 2	5	1	1	lg item	0.2	5
5	Test PUV # 2	2	1	8	systems	4	2
6	Offload Bal	15	1	3	lg item	0.2	15
9	Export Balance	5	1	5000	meters	1000	5
10	RLSS		0				
11	Install RLSS ext	25	1	5	lg items	0.2	25
12	Inspect RLSS ext	25	1	100	points	4	25
13	RLSS Ingress	44	1	59	items	1.33	44
14	Install RLSS int	15	1	59	sml items	4	15
15	Inspect int	25	1	100	points	4	25
16	Test RLSS	100	1	400	systems	4	100
17	Repair/Startup	200	1	100	systems	0.5	200
18	Outfitting		0				
19	Outfitting Ingr	67	1	89	items	1.33	67
20	Inst Outfitting	22	1	89	sml items	4	22
21	Test Outfitting	11	1	45	systems	4	11
22	Repair/Start Up	24	1	12	systems	0.5	24
23	Opt Tel # 2		0				
24	Layout	1	1	12	points	12	1
25	Surf Prep		0				
26	Level	5	1	15	cubic meters	3	5
27	Remove sml Rocks	1	1	1.5	sml rocks	4	1
28	Remove Lg Rocks	1	1	0.15	lg rocks	0.25	1
29	Emplace Opt Tel2	1	1	1	med item	2	1
30	Anchor OT #2	4	1	4	points	1	4
31	Inspect OT # 2	2	1	6	points	4	2
32	Test O T # 2	1	1	4	systems	4	1
33	Repair/StUp OT2	2	1	1	systems	0.5	2
34	Cover Hab	567	1	1700	cubic meters	3	567

Year: 2008, Month: 1

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Offload Lander	5	1	1	lg item	0.2	5
2	Xport Cargo	5	1	5000	meters	1000	5
3	Stow Consumables	25	1	100	sml items	4	25
4	Opt Tel # 3		0				
5	Layout	1	1	12	points	12	1
6	Surf Prep		0				
7	Level	5	1	15	cubic meters	3	5
8	Remove Sml Rocks	1	1	1.5	sml rocks	4	1
9	Remove Lg Rocks	1	1	0.15	lg rocks	0.25	1
10	Emplace OT # 3	1	1	1	med item	2	1
11	Anchor OT # 3	4	1	4	points	1	4
12	Inspect	2	1	6	points	4	2
13	Test	1	1	4	systems	4	1
14	Repair/Startup	2	1	1	systems	0.5	2

Year: 2008, Month: 7

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Lander Fac Addns		0				
2	Offload Addns	25	1	5	lg items	0.2	25
3	Emplace Addns	25	1	5	lg items	0.2	25
4	Test Addns	3	1	10	systems	4	3
5	Comm Tower		0				
6	Offload Com Twr	1	1	1	med items	2	1
7	Xport Com Twr	1	1	5000	meters	4000	1
8	Emplace Found	2	1	4	med items	2	2
9	Anchor Found	4	1	4	points	1	4
10	Emplace Twr	1	1	1	med item	2	1
11	Anchor Twr	4	1	4	points	1	4
12	Connect Twr	1	1	4	points	4	1
13	Test Twr	1	1	4	systems	4	1
14	Repair/Startup	2	1	1	systems	0.5	2

Year: 2009, Month: 1

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Offload Lander	10	1	2	lg items	0.2	10
2	Xport Cargo	5	1	5000	meters	1000	5
3	Stow Consumables	25	1	100	sml items	4	25
4	Prep Next Phase		0				
5	Nuc Pwr Plt		0				
6	Layout	2	1	20	points	12	2
7	Excavate		0				
8	Dig pit	12	1	35	cubic meters	3	12
9	Remove sml rocks	1	1	4	sml rocks	4	1
10	Remove lg rocks	4	1	1	lg rocks	0.25	4
11	Surf Prep		0				
12	Level	141	1	423.9	cubic meters	3	141
13	Remove sml Rocks	1	1	4	sml rocks	4	1
14	Remove Lg rocks	4	1	1	lg rocks	0.25	4
15	PMAD		0				
16	Layout	4	1	50	points	12	4
17	Trench		0				
18	Dig trench	15	1	43.75	cubic meters	3	15
19	Remove Sml Rocks	1	1	4	sml rocks	4	1
20	Remove lg rocks	4	1	1	lg rocks	0.25	4

Year: 2009, Month: 7

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Consumables		0				
2	Offload Consums	5	1	1	lg item	0.2	5
3	Xport Consums	5	1	5000	meters	1000	5
4	Ingress Consums	81	1	108	sml items	1.33	81
5	Stow Consums	27	1	108	sml items	4	27
6	Nuclear Pwr Plt		0				
7	Offload Nuc	5	1	1	lg item	0.2	5
8	Xport Nuc	6	1	6000	meters	1000	6
9	Emplace bulkhead	5	1	1	large piece	0.2	5
10	Inspect	3	1	12	points	4	3
11	Backfill pit	2	1	14	m3	6	2
12	Inspect	3	1	12	points	4	3
13	Emplace reactor	5	1	1	large piece	0.2	5
14	Emplace I/O Mani	1	1	1	medium piece	2	1
15	Connect I/O Mani	1	1	2	points	4	1
16	Emplace engine p	4	1	8	med pieces	2	4
17	Emplace engines	40	1	8	large pieces	0.2	40
18	Connect heat rej	4	1	16	points	4	4
19	Connect shunts	4	1	16	points	4	4
20	Connect converte	2	1	8	points	4	2
21	Inspect	14	1	56	points	4	14
22	Emplace reflecti	40	1	8	large pieces	0.2	40
23	Emplace panels	20	1	40	med pieces	2	20
24	Connect panels	10	1	40	points	4	10
25	Anchor radiators	48	1	48	points	1	48
26	Connect manifold	2	1	8	points	4	2
27	Emplace switchin	2	1	3	med piece	2	2
28	Emplace utilitie	1	1	700	meters	500	1
29	Connect utilitie	2	1	8	points	4	2
30	Inspect	9	1	35	points	4	9
31	Backfill trench	7	1	43.75	cubic meters	6	7
32	Inspect	9	1	35	points	4	9
33	Final Prep/Clean	1	1	140	square meter	250	1
34	Activate & Test	8	1	32	systems	4	8
35	Repair Startup	16	1	8	systems	0.5	16

Year: 2010, Month: 1

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Consumables		0				
2	Offload Consums	5	1	1	lg item	0.2	5
3	Xport Consums	5	1	5000	meters	1000	5
4	Stow Consums	25	1	100	sml items	4	25

Year: 2010, Month: 7

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Consumables		0				
2	Offload Consums	5	1	1	lg item	0.2	5
3	Xport Consums	5	1	5000	meters	1000	5
4	Stow Consums	25	1	100	sml items	4	25
5	LLOX Facility		0				
6	LLOX Plant		0				
7	Layout	2	1	20	points	12	2
8	Surf Prep		0				
9	Level	3	1	10.05	cubic meters	3	3
10	Remove sml rocks	1	1	1.005	sml rocks	4	1
11	Remove lg rocks	1	1	0.1005	lg rocks	0.25	1
12	Offload	5	1	1	lg item	0.2	5
13	Xport	5	1	5000	meters	1000	5
14	Emplace	5	1	1	lg item	0.2	5
15	Anchor	3	1	3	points	1	3
16	TCS		0				
17	Emplace	5	1	1	lg item	0.2	5
18	Anchor	8	1	8	points	1	8
19	Connect	1	1	4	points	4	1
20	Bene Equip		0				
21	Offload	5	1	1	lg item	0.2	5
22	Emplace Lg Item	1	1	0	lg items	21	1
23	Emplace Med Item	2	1	3	med items	2	2
24	Emplace Sml Item	1	1	4	sml items	4	1
25	Inspect	3	1	10	points	4	3
26	Connect cables	2	1	6	points	4	2
27	Storage Tanks		0				
28	Excavate		0				
29	Dig	20	1	60	cubic meters	3	20
30	Remove sml Rocks	2	1	6	sml rocks	4	2
31	Remove Lg rocks	2	1	0.6	lg rocks	0.25	2
32	Emplace Tanks	1	1	2	med items	2	1
33	Connect	2	1	8	points	4	2
34	Cover Tanks	5	1	30	cubic meters	6	5
35	PMAD		0				
36	Layout	3	1	35	points	12	3
37	Trench		0				
38	Dig trench	21	1	62.5	cubic meters	3	21
39	Remove sml rocks	2	1	6.25	sml rocks	4	2
40	Remove lg rocks	3	1	0.625	lg rocks	0.25	3
41	Emplace Cable	2	1	1000	meters	500	2
42	Cover Cable	10	1	62.5	cubic meters	6	10
43	Connect Cable	2	1	8	points	4	2
44	Inspect all syst	5	1	20	points	4	5
45	Activate/test	1	1	10	systems	30	1
46	Repair/start up	6	1	3	systems	0.5	6
47	Final prep/cl up	1	1	250	square meter	250	1
48	Mining Equip		0				
49	Offload	15	1	3	lg items	0.2	15
50	Xport	5	1	5000	meters	1000	5
51	Test	3	1	12	systems	4	3
52	Repair/Start up	6	1	3	systems	0.5	6
53	G-R Telescope		0				

Year: 2010, Month: 7

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
54	Layout	1	1	12	points	12	1
55	Surf Prep		0				
56	Level	11	1	33.75	cubic meters	3	11
57	Remove sml rocks	1	1	3.375	sml rocks	14	1
58	remove lg rocks	1	1	0.3375	lg rocks	0.25	1
59	Offload	10	1	2	Lg item	0.2	10
60	Xport	5	1	5000	meters	1000	5
61	Emplace	10	1	2	lg items	0.2	10
62	Anchor	6	1	6	points	1	6
63	Inspect	1	1	4	points	4	1
64	Test	1	1	4	systems	4	1
65	Repair/start up	2	1	1	systems	0.5	2

Year: 2011, Month: 1

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	Consumables		0				
2	Offload Consums	5	1	1	lg item	0.2	5
3	Xport Consums	5	1	5000	meters	1000	5
4	Stow Consums	40	1	160	sml items	4	40

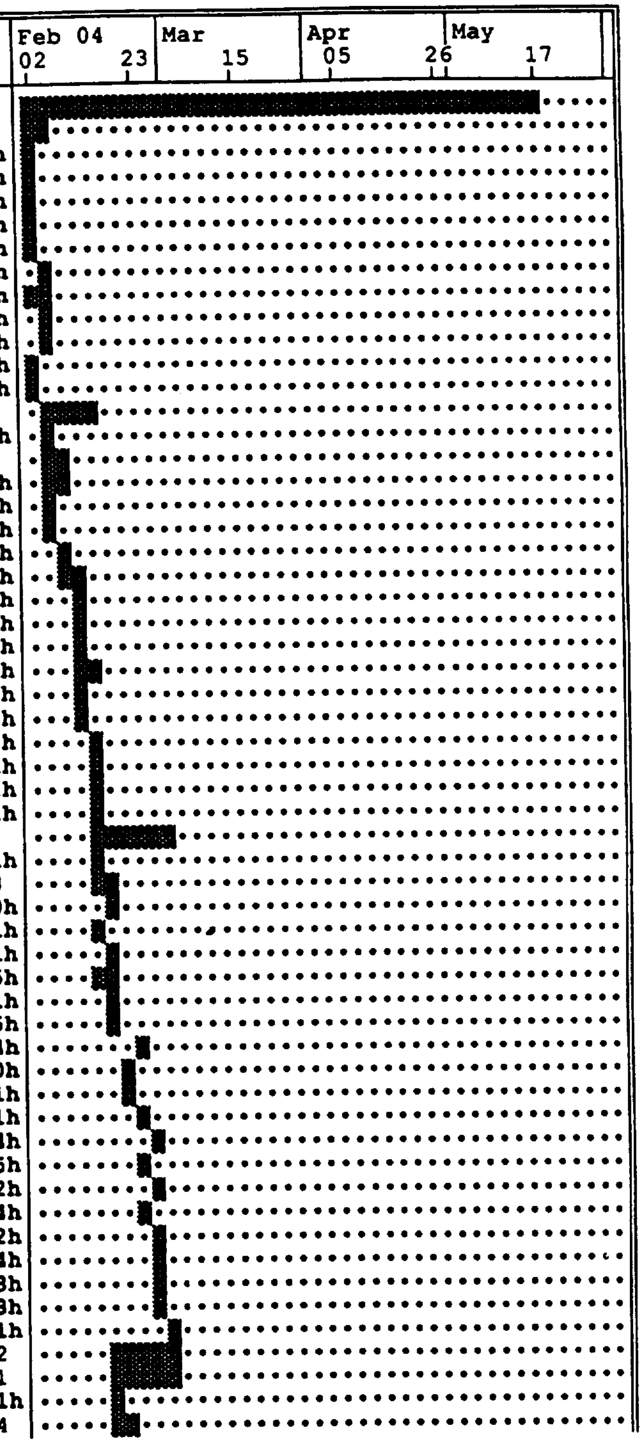
Year: 2011, Month: 7

Task ID	Task Name	Duration	Type	Quantity Of Work	Units	Production Rate/Hour	Task Duration Hours
1	PRV		0				
2	Offload	5	1	1	lg item	0.2	5
3	Test	1	1	4	systems	4	1
4	Offload Bal	15	1	3	lg items	0.2	15
5	Transport Bal	5	1	5000	meters	1000	5
6	Stow Sci Resup	1	1	4	sml items	4	1
7	Ingress Lab Inst	75	1	100	sml items	1.33	75
8	Install Lab inst	25	1	100	sml items	4	25
9	X-Ray Tele		0				
10	Layout	1	1	12	points	12	1
11	Surf prep		0				
12	Level	5	1	15	cubic meters	3	5
13	Remove sml rocks	1	1	1.5	sml rocks	4	1
14	Remove lg rocks	1	1	0.15	lg rocks	0.25	1
15	Transport	1	1	1000	meters	4000	1
16	Emplace	1	1	2	med items	2	1
17	Anchor	4	1	4	points	1	4
18	Inspect	1	1	4	points	4	1
19	Test	2	1	6	systems	4	2
20	Repair/startup	4	1	2	systems	0.5	4

Task Gantt
11-29-89 2:07p

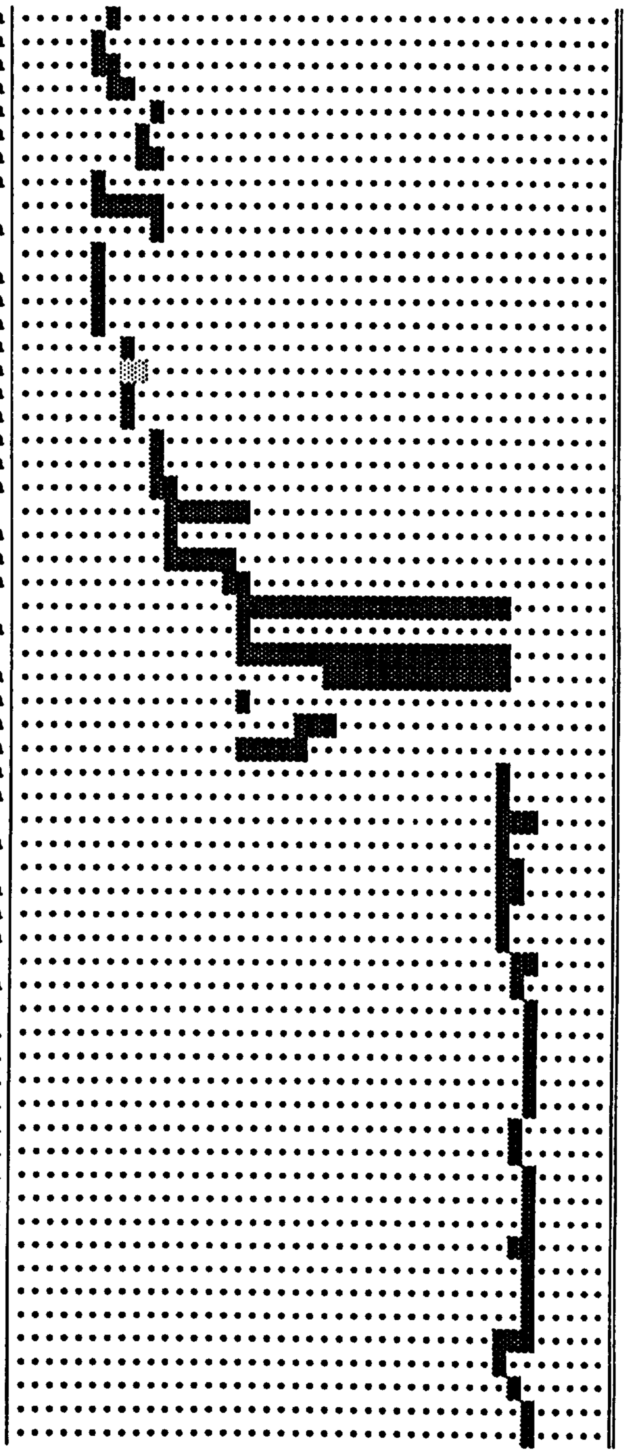
Project: I2004_2.PJ
Revision: 28

Task ID	Heading/Task	Dur
P1	**I2004_2.PJ**	92
001	**Activate Equip**	5
002	± Offload MWP	1h
003	± Test MWPs	2h
004	± Offload Trucks	1h
005	± Test Trucks	2h
006	± Offload FCP Cart	5h
007	± Test FCP Cart	1h
008	± Offload Implmts	3h
009	± Offload TC Cart	5h
010	± Test TC Cart	1h
011	± Offload Liqf Tks	5h
012	± Test Liqf Tks	1h
013	**PVA/RFC**	8
014	± Layout PVA/RFC	1h
015	**Surf Prep PVARFC**	4
016	± Level PVA/RFC	20h
017	± Remove Sml Rocks	2h
018	± Remove Lg Rocks	2h
019	± Unload PVA-RFC	5h
020	± Xport PVA-RFC	1h
021	± Emplace PVA	2h
022	± Emplace RFC	5h
023	± Anchor RFC	4h
024	± Connect PVA	2h
025	± Anchor PVA	4h
026	± Connect PVA/RVC	1h
027	± Inspect PVA/RFC	1h
028	± Test PVA/RFC	1h
029	± Repair/Startup	2h
030	± Final Prep/Clnup	1h
031	**Habitat**	14
032	± Layout Hab Mod	1h
033	**Surf Prep Hab**	3
034	± Level Hab Area	10h
035	± Remove Sml Rocks	1h
036	± Remove Lg Rocks	1h
037	± Offload Hab Mod	5h
038	± Xport Hab Mod	1h
039	± Emplace Hab Mod	5h
040	± Anchor Hab Mod	4h
041	± Unload ALock&TCS	10h
042	± Xport ALock &TCS	1h
043	± Emplace TCS	1h
044	± Anchor TCS	4h
045	± Emplace Airlock	5h
046	± Connect Airlock	2h
047	± Anchor Airlock	4h
048	± Emplace Tent	2h
049	± Anchor Tent	4h
050	± Inspect Hab Sys	3h
051	± Test Hab Sys	3h
052	± Final Prep/Clnup	1h
053	**Utilities**	12
054	**Power Cable**	11
055	± Layout Cab Trnch	1h
056	**Dig Cable Trench**	4



Across: 1 Down: 2

057	±	Excavate Trench	2h
058	±	Remove Sml Rocks	1h
059	±	Remove Lg Rocks	1h
060	±	Install Cables	1h
061	±	Cover Cables	1h
062	±	Connect Cables	1h
063	±	Inspect Cables	1h
064	±	Final Prep/Clnup	1h
065		**TCS Piping**	12
066	±	Layout Trench	1h
067		**Excavate Cab Trn**	3
068	±	Dig Piping Trenc	1h
069	±	Remove Sml Rocks	1h
070	±	Remove Lg Rocks	1h
071	±	Install Piping	1h
072	±	Cover Trench	1h
073	±	Connect Piping	1h
074	±	Inspect Piping	1h
075	±	Final Prep/Clnup	1h
076	±	Test All Systems	8h
077	±	Repair/Startup	14h
078		**Stock Hab**	12
079	±	Xport Supplies	1h
080	±	Ingress Supplies	49h
081	±	Stow Supplies	16h
082		**Launch/Land Area**	47
083	±	Layout LL Area	2h
084		**SurfPrep LL Area**	47
085	±	Level LL Area	192h
086	±	Remove Sml Rocks	14h
087	±	Remove Lg Rocks	23h
088	±	Blast Barriers	60h
089	±	Anchor Pad Marke	1h
090	±	Anchor Nav Becon	3h
091		**Science Area**	8
092	±	Layout Sci Area	1h
093		**Surf Prep Sci**	4
094	±	Level Sci Area	20h
095	±	Remove Sml Rocks	1h
096	±	Remove Lg Rocks	1h
097		**Cable**	5
098	±	Layout	1h
099		**Trench**	3
100	±	Dig Trench	2h
101	±	Remove Sml Rocks	1h
102	±	Remove Lg Rocks	1h
103	±	Lay Cable	1h
104	±	Cover Cable	5h
105	±	Connect Cable	1h
106	±	Inst Solar Obser	2h
107	±	Insp Solar Obser	1h
108	±	Test Solar Obser	1h
109	±	Inst Geo Station	1h
110	±	Insp Geo Station	2h
111	±	Test Geo Station	2h
112	±	Final Prep/Clnup	1h
113		**LLOX Demo Prep**	6
114	±	Install LLOXDemo	2h
115	±	Lay Pwr Cables	1h
116	±	Connect Cables	1h
117	±	Inspect LLOX POC	1h



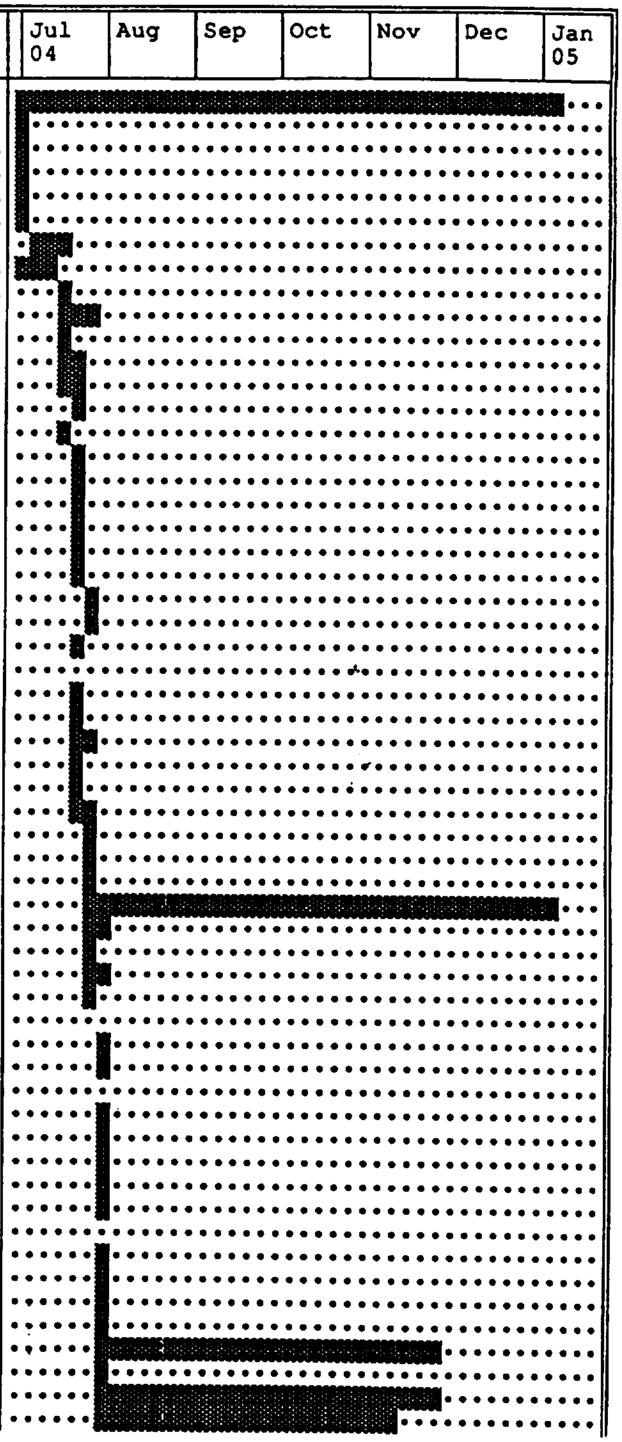
Across: 1 Down: 3

118	± Test LLOX POC	1h	
119	± Final Prep/Clnup	1h	

Non Critical m Milestone ___ Interrupt
Critical M Critical MS

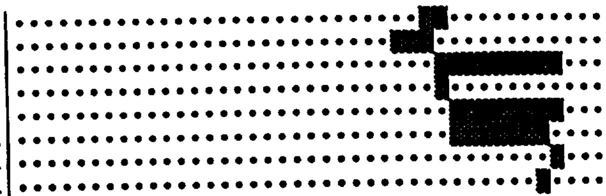
Emplacement Phase - Personnel Flight 2

Task ID	Heading/Task	Dur
P2	I2004_7.PJ	160
001	**Receive Cargo**	4
002	± Offload Rover	1h
003	± Test Rover	1h
004	± Offload Lander	20h
005	± Transport All	1h
006	**Supplies**	12
007	± Ingress Supplies	50h
008	± Stow Supplies	17h
009	**PVA-RFC # 2**	9
010	± Layout PVA/RFC	1h
011	**Surf Prep PVARFC**	4
012	± Level PVA/RFC	20h
013	± Remove Sml Rocks	2h
014	± Remove Lg Rocks	2h
015	± Emplace RFC	5h
016	± Emplace PVA	2h
017	± Connect PVA	2h
018	± Connect PVA/RFC	1h
019	± Inspect PVA/RFC	2h
020	± Test PVA/RFC	1h
021	± Repair/Startup	2h
022	± Final Prep/Clnup	1h
023	**Comm Equip**	3
024	± Emplace Comm Eq	2h
025	± Inspect Comm Eq	1h
026	± Test Comm Equip	1h
027	**Node**	6
028	± Emplace Node	5h
029	± Connect Node	2h
030	± Anchor Node	4h
031	± Inspect Node	2h
032	± Test Node	1h
033	± Repair/Startup	1h
034	**Prep Next Phase**	139
035	**PVA # 3 Prep**	6
036	± Layout PVA 3	1h
037	**Surf Prep**	6
038	± Level PVA 3	20h
039	± Remove Sml Rocks	2h
040	± Remove Lg Rocks	2h
041	**Opt Tel # 1 Prep**	3
042	± Layout Op Tel 1	1h
043	**Surf Prep Tel 1**	3
044	± Level Tel 1	5h
045	± Remove Sml Rocks	1h
046	± Remove Lg Rocks	1h
047	**M-E R Inf Prep**	3
048	± Layout ME Inf	2h
049	**Surf Prep**	2
050	± Level ME Inf	5h
051	± Remove Sml Rocks	1h
052	± Remove lg Rocks	1h
053	**Cmp Hb Area Prep**	101
054	± Layout Hab Area	3h
055	**Surf Prep**	100
056	± Level Hab Area	500h



Across: 1 Down: 2

057	±	Remove Sml Rocks	38h
058	±	Remove Lg Rocks	60h
059		**Hab Area Roads**	34
060	±	Layout Roads	6h
061		**Surf Prep**	33
062	±	Level Roads	167h
063	±	Remove Sml Rocks	13h
064	±	Remove Lg Rocks	20h



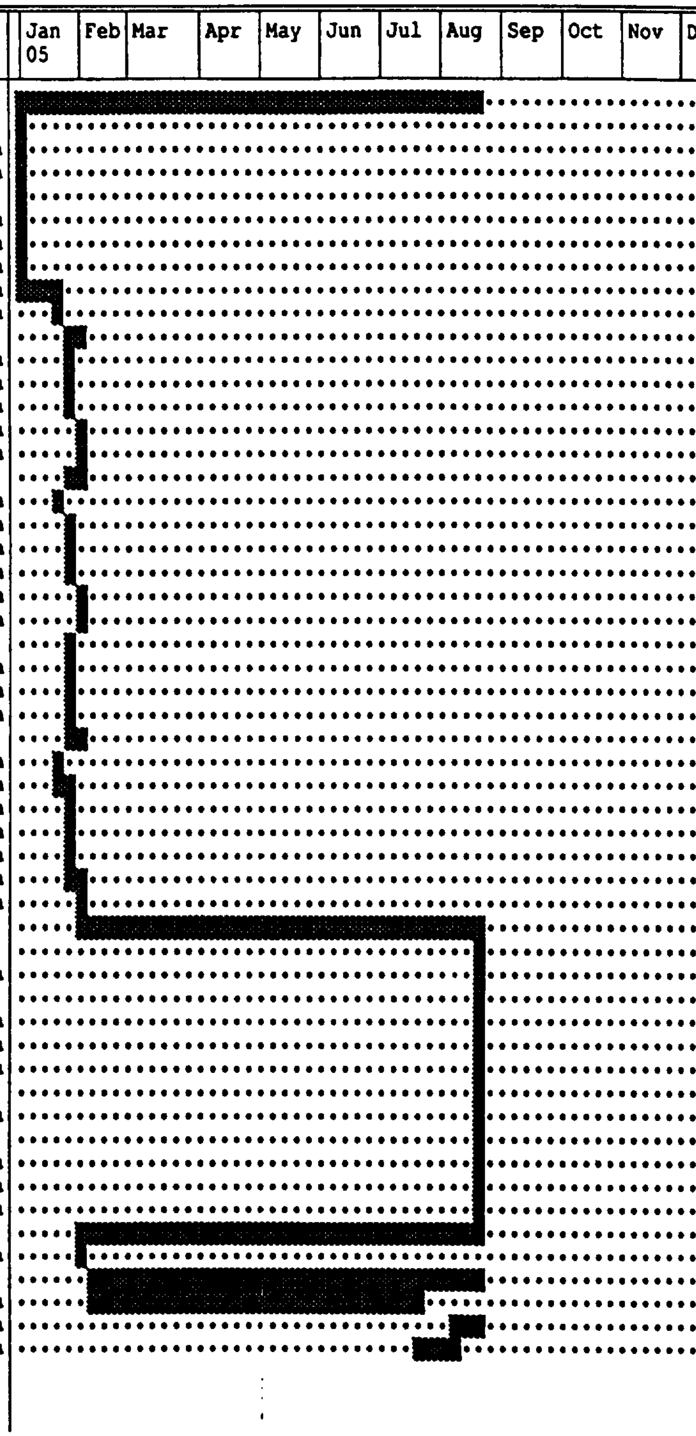
Non Critical m Milestone ___ Interrupt
Critical M Critical MS

Task Gantt
11-29-89 5:38p

Project: I2005_1.PJ
Revision: 14

Begin Consolidation Phase

Task ID	Heading/Task	Dur
P3	I2005_1.PJ	198
001	**Activate Equip**	1
002	± Unload UP Rover	1h
003	± Test UP Rover	1h
004	**Offload Balance**	3
005	± Lg Items	10h
006	± Med items	3h
007	±Transport Cargo	1h
008	±Ingress Supplies	75h
009	±Stow Supplies	25h
010	**Optical Tel #1**	7
011	± Emplace O T #1	2h
012	± Connect OT # 1	2h
013	± Anchor OT #1	4h
014	± Test OT # 1	1h
015	± Repair/Start up	2h
016	**PVA**	7
017	± Emplace PVA # 3	2h
018	± Anchor PVA #3	4h
019	± Connect PVA # 3	2h
020	± Inspect PVA #3	1h
021	± Test PVA # 3	1h
022	± Repair/Start up	2h
023	**Cable**	1
024	± Install Calbes	1h
025	± Connect Cables	2h
026	± Cover Cables	1h
027	**M/E Radio Intrf**	7
028	± Emplace MERI	2h
029	± Anchor	4h
030	± Connect	1h
031	± Inspect	1h
032	± Test MERI	2h
033	± Repair/Start up	4h
034	± Final Prep/Clnup	1h
035	**Prep next phase**	173
036	**Mon Teles 2 Prep**	2
037	± Layout Telescops	1h
038	**Sur Prep Teles**	2
039	± Level	5h
040	± Remove Sml Rocks	1h
041	± Remove Lg Rocks	1h
042	**Earth Observator**	3
043	± Layout EO	1h
044	**Surf Prep EO**	3
045	± Level	5h
046	± Remove Sml Rocks	1h
047	± Remove Lg Rocks	1h
048	**Road to LL Area**	171
049	± Layout LL Road	28h
050	**Surface Prep**	167
051	± Level LL Road	833h
052	± Remove Sml Rocks	63h
053	± Remove Lg Rocks	100h

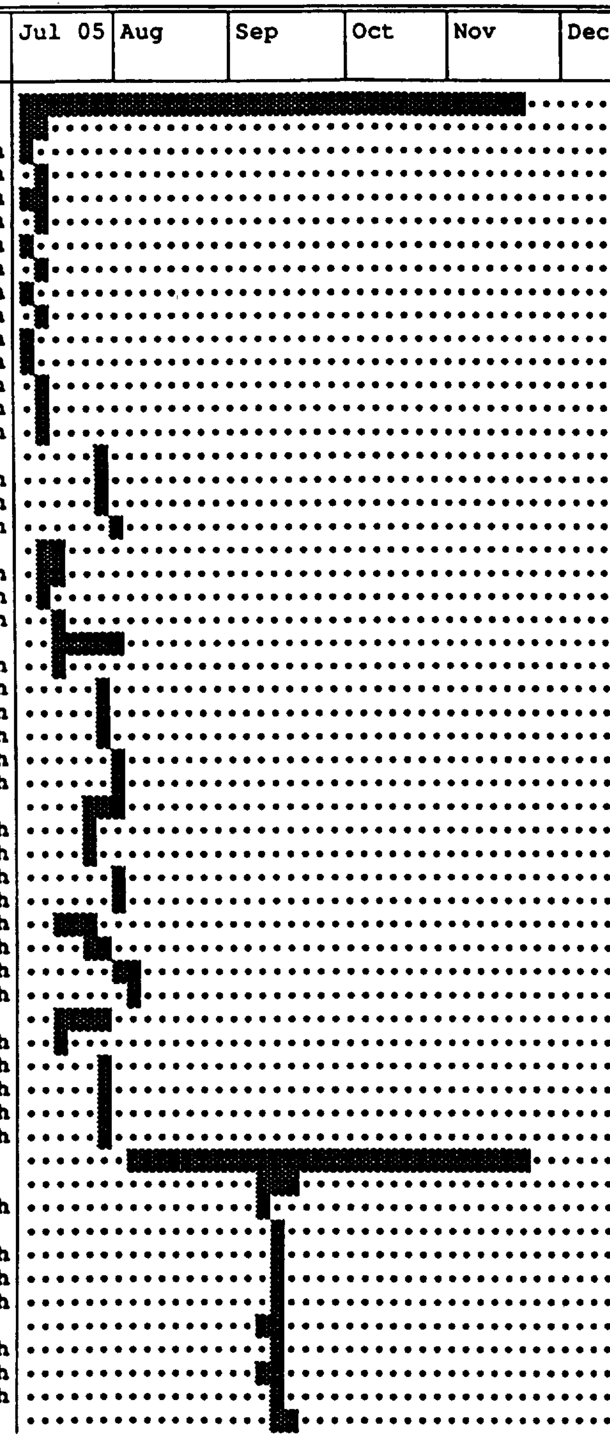


Task Gantt
11-29-89 5:42p

Project: I2005_7.PJ
Revision: 18

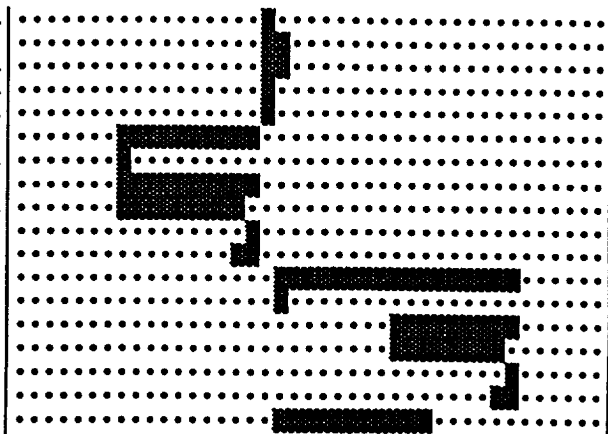
Consolidation Phase, First Cargo Flight

Task ID	Heading/Task	Dur
P4	I2005_7.PJ	118
001	**Activate Equip**	7
002	± Unload P Uti Veh	5h
003	± Test P Util Veh	2h
004	± Unload Tun Ramp	5h
005	± Test Tun Ramp	1h
006	± Unload Pwr Tlr	5h
007	± Test Pwr Tlr	1h
008	± Unload FC P Cart	5h
009	± Test FC P Cart	1h
010	± Unload TC Carts	5h
011	± Test TC Carts	1h
012	± Unload Lab Tlr	5h
013	± Unload Rover	1h
014	± Test Rover	1h
015	**Liq Plant/Tanks**	1
016	± Emplace P/Tanks	1h
017	± Test Plant/Tanks	1h
018	±Emplace Tents	2h
019	**Unload Balance**	3
020	± Lg items	15h
021	± Med items	1h
022	±Transport Bal	1h
023	**Telescopes**	14
024	± Emplace Teles	3h
025	± Anchor Tele	8h
026	± Connect	2h
027	± Test Telescopes	2h
028	± Repair/Start up	4h
029	± Final Prep	1h
030	**Lab Trailer**	8
031	± Transport Lab Tr	1h
032	± Emplace Lab Tlr	5h
033	± Conn Lab Tlr	3h
034	± Test Lab Tlr	1h
035	±Ingress Bio Lab	28h
036	±Emplace Biomed L	9h
037	±Ingress Sci Lab	9h
038	±Emplace Sci Lab	3h
039	**Earth Observator**	13
040	± Emplace EO	5h
041	± Anchor EO	4h
042	± Inspect EO	1h
043	± Test EO	1h
044	± Repair/Start up	2h
045	**Prep next phase**	95
046	**SP-100 Prep**	5
047	± Layout SP-100	1h
048	**Surf Prep SP-100**	2
049	± Level	9h
050	± Remove Sml Rocks	1h
051	± Remove Lg Rocks	1h
052	**Excavate**	3
053	± Dig Hole	8h
054	± Remove Sml Rocks	1h
055	± Remove Lg Rocks	1h
056	**Cable for SP 100**	4



Across: 1 Down: 2

057	±	Layout Cab SP100	1h
058		**Dig Trench**	4
059	±	Trench	10h
060	±	Remove Sml Rocks	1h
061	±	Remove Lg Rocks	1h
062		**Road To Nuc Plts**	35
063	±	Layout	6h
064		**Surface Prep**	34
065	±	Level	167h
066	±	Remove Lg Rocks	13h
067	±	Remove Sml Rocks	20h
068		**LL Area**	57
069	±	Layout	5h
070		**Surface Prep**	28
071	±	Level	144h
072	±	Remove Sml Rocks	11h
073	±	Remove LG Rocks	17h
074	±	Blast Barriers	220h



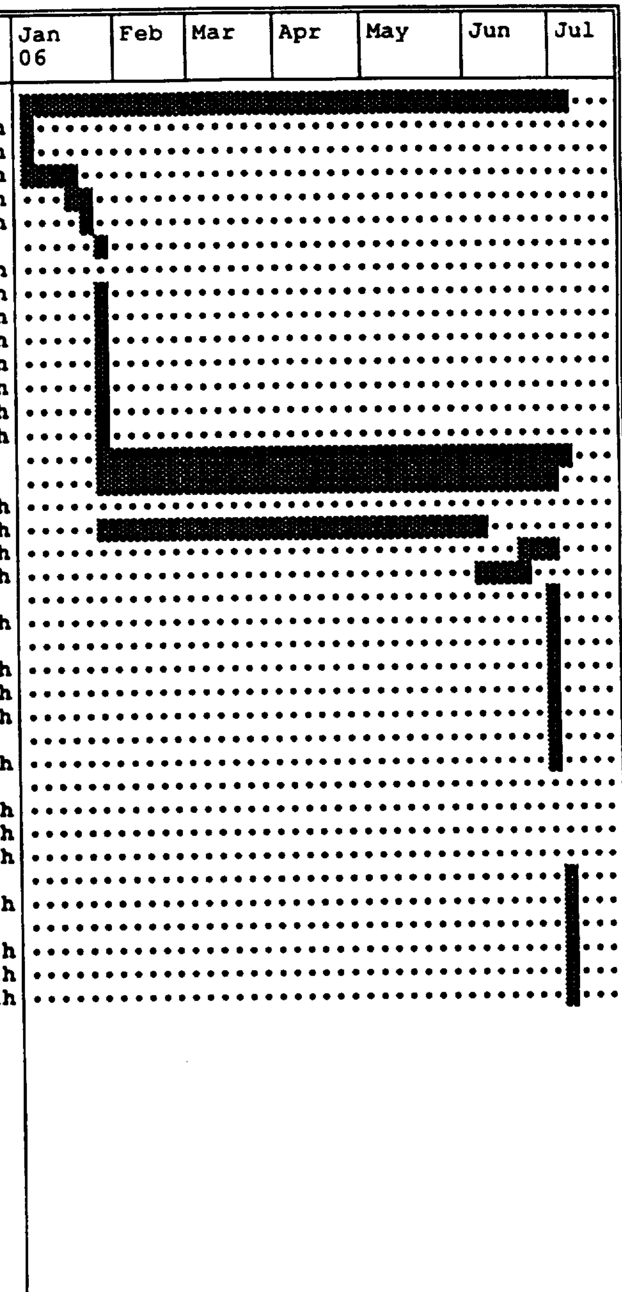
Non Critical m Milestone ___ Interrupt
Critical M Critical MS

Task Gantt
11-29-89 5:45p

Project: I2006_1.PJ
Revision: 14

Consolidation Phase Continued 1

Task ID	Heading/Task	Dur
P5	**I2006_1.PJ**	160
001	±Unload Lander	10h
002	±Xport to Base	5h
003	±Ingress Cons	84h
004	±Stow Consumables	27h
005	±Emplace Sc Resup	1h
006	**SP-100**	4
007	± Emplace SP-100	5h
008	± Emplace Cable	1h
009	± Cover Trench	5h
010	± Connect SP-100	1h
011	± Inspect SP-100	3h
012	± Test SP-100	3h
013	± Repair/Start up	4h
014	± Final Prep/Clnup	1h
015	**Prep Next Phase**	137
016	**Inf Hab Prep**	133
017	± Layout Inf Hab	1h
018	± Excavate Inf Hab	667h
019	± Remove Sml Rocks	50h
020	± Remove Lg Rocks	80h
021	**IR Tele Prep**	2
022	± Layout IR Tele	1h
023	**Surface Prep**	2
024	± Level	5h
025	± Remove Sml Rocks	1h
026	± Remove Lg Rocks	1h
027	**UV Tele Prep**	2
028	± Layout UV Tele	1h
029	**Surface prep**	1
030	± Level	5h
031	± Remove Sml Rocks	1h
032	± Remove Lg Rocks	1h
033	**Rd To ISRU Area**	2
034	± Layout	2h
035	**Surface Prep**	2
036	± Level	5h
037	± Remove Sml Rocks	1h
038	± Remove Lg Rocks	1h

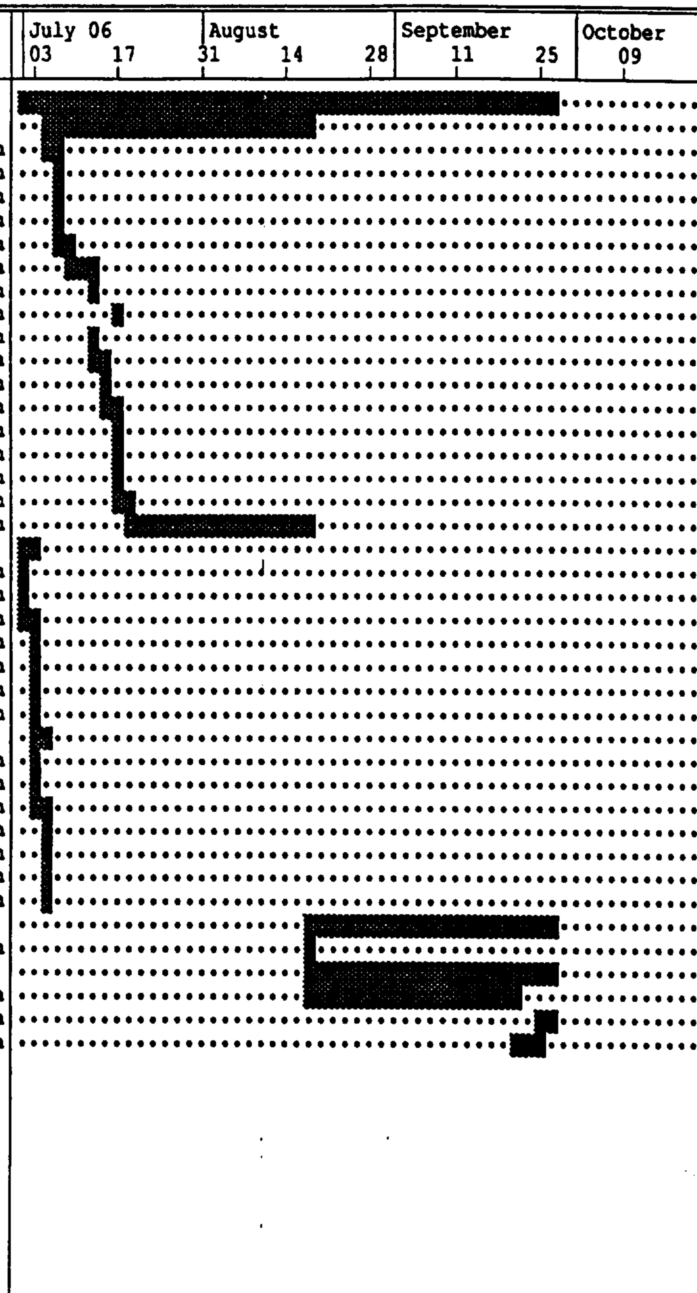


Non Critical m Milestone ___ Interrupt
Critical M Critical MS

Task Gantt
11-29-89 3:40p

Project: I2006_7.PJ
Revision: 22

Task ID	Heading/Task	Dur
P6	I2006_7.PJ	75
001	Inf Hab	39
002	± Offload Inf Hab	5h
003	± Xport Inf Hab	5h
004	± Emplace Found.	1h
005	± Anchor Found	1h
006	± Emplace Inf Hab	5h
007	± Inflate Hab	20h
008	± Conn to Found.	2h
009	± Conn Power	1h
010	± Offload Tunnel	5h
011	± Xport Tunnel	5h
012	± Emplace Tunnel	5h
013	± Connect Tunnel	2h
014	± Anchor Tunnel	4h
015	± Inspect Hab/Tunn	3h
016	± Test Hab/tunnel	1h
017	± Repair/Startup	2h
018	± Backfill Hab	167h
019	**IR Telescope**	3
020	± Emplace IRT	2h
021	± Connect	2h
022	± Anchor	4h
023	± Inspect	1h
024	± Test IRT	1h
025	± Repair/Startup	2h
026	± Final Prep	1h
027	**UV Telescope**	3
028	± Emplace UVT	2h
029	± Connect	2h
030	± Anchor	4h
031	± Inspect	1h
032	± Test UVT	1h
033	± Repair/startup	2h
034	± Final Prep	1h
035	**Road Nuc to ISRU**	34
036	± Layout	4h
037	**Surface Prep**	34
038	± Level	167h
039	± Remove Sml Rocks	13h
040	± Remove Lg Rocks	20h

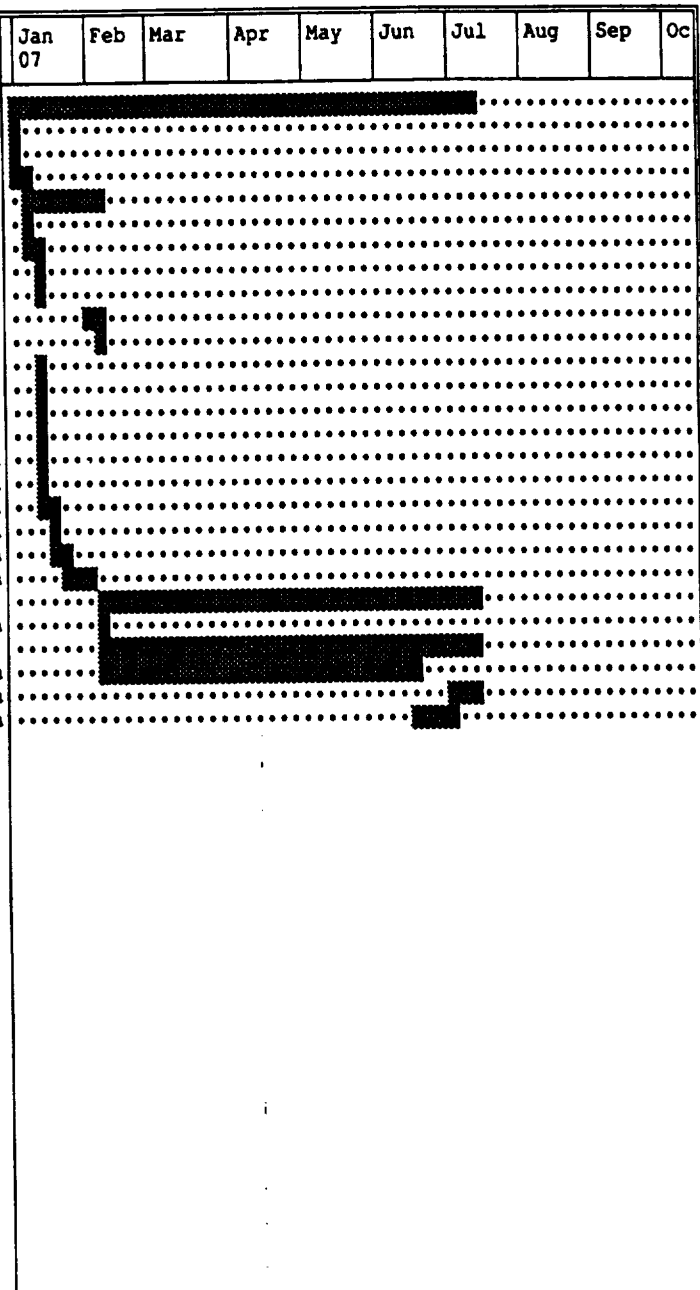


Non Critical ■ Milestone ___ Interrupt
Critical M Critical MS

Task Gantt
11-29-89 5:46p

Project: I2007_1.PJ
Revision: 14

Task ID	Heading/Task	Dur
P7	I2007_1.PJ	163
001	±Offload Lander	5h
002	±Xport Cargo	5h
003	±Stow Consumables	25h
004	**TCS Const Hab**	26
005	± Offload TCS	5h
006	± Xport TCS	20h
007	± Emplace TCS ext	5h
008	± Anchor TCS ext	8h
009	± TCS Ingress	11h
010	± Emplace TCS int	4h
011	**Piping Trench**	1
012	± Dig Trench	2h
013	± Remove sml rocks	1h
014	± Remove lg rocks	1h
015	± Emplace Piping	1h
016	± Cover Trench	1h
017	± Connect Piping	1h
018	± Inspect TCS	10h
019	± Test TCS	29h
020	± Repair/Startup	58h
021	**Rd ISRU To L/L**	134
022	± Layout	3h
023	**Surface Prep**	133
024	± Level	667h
025	± Remove Sml Rocks	50h
026	± Remove Lg Rocks	80h

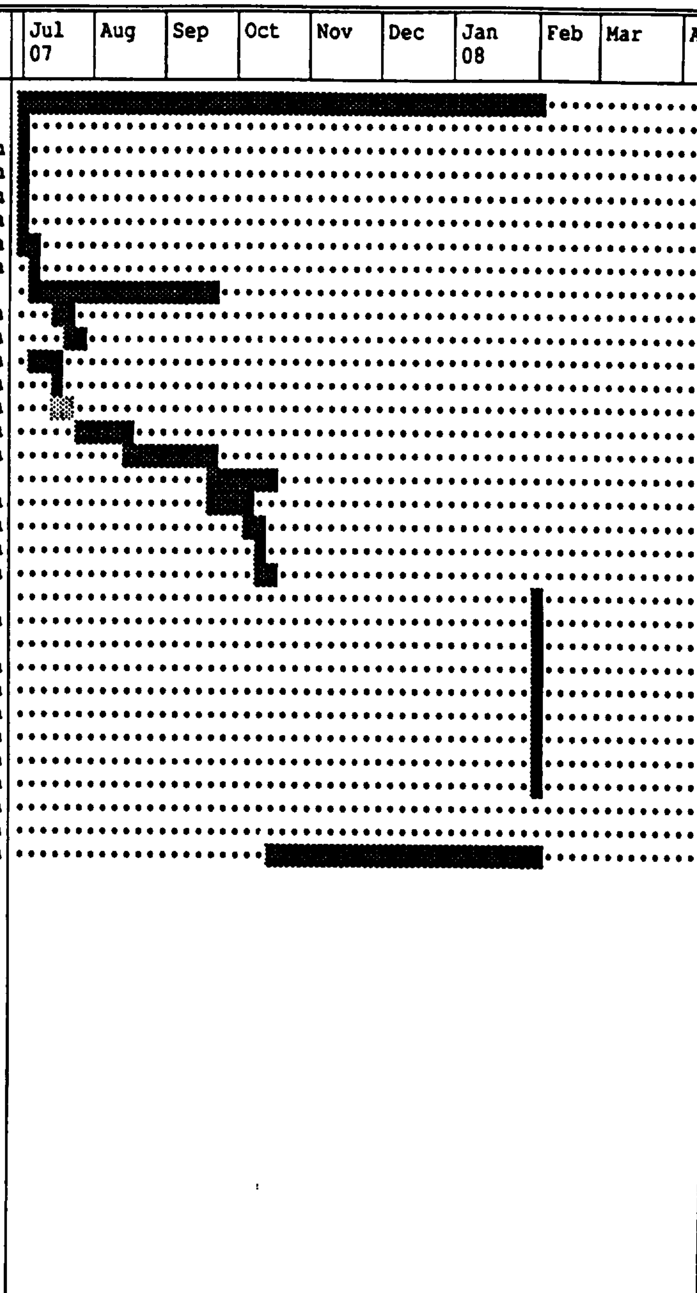


Non Critical m Milestone ___ Interrupt
Critical M Critical MS

Task Gantt
11-29-89 3:46p

Project: I2007_7.PJ
Revision: 17

Task ID	Heading/Task	Dur
P8	I2007_7.PJ	187
001	**Activate Equip**	2
002	± Offload Pwr Tlr	5h
003	± Test Pwr Tlr	1h
004	± Offload PUV # 2	5h
005	± Test PUV # 2	2h
006	±Offload Bal	15h
009	±Export Balance	5h
010	**RLSS**	67
011	± Install RLSS ext	25h
012	± Inspect RLSS ext	25h
013	± RLSS Ingress	44h
014	± Install RLSS int	15h
015	± Inspect int	25h
016	± Test RLSS	100h
017	± Repair/Startup	200h
018	**Outfitting**	20
019	± Outfitting Ingr	67h
020	± Inst Outfitting	22h
021	± Test Outfitting	11h
022	± Repair/Start Up	24h
023	**Opt Tel # 2**	3
024	± Layout	1h
025	**Surf Prep**	2
026	± Level	5h
027	± Remove sml Rocks	1h
028	± Remove Lg Rocks	1h
029	± Emplace Opt Tel2	1h
030	± Anchor OT #2	4h
031	± Inspect OT # 2	2h
032	± Test O T # 2	1h
033	± Repair/StUp OT2	2h
034	±Cover Hab	567h

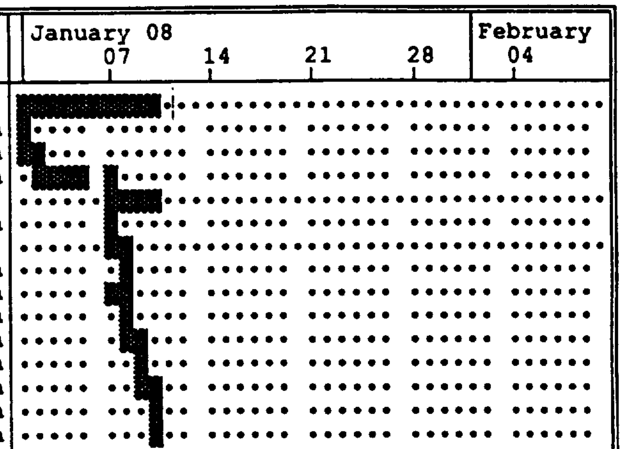


Non Critical ■ Milestone ___ Interrupt
Critical M Critical MS

Task Gantt
11-29-89 3:48p

Project: I2008_1.PJ
Revision: 10

Task ID	Heading/Task	Dur
P9	**I2008_1.PJ**	8
001	±Offload Lander	5h
002	±Xport Cargo	5h
003	±Stow Consumables	25h
004	**Opt Tel # 3**	4
005	± Layout	1h
006	**Surf Prep**	2
007	± Level	5h
008	± Remove Sml Rocks	1h
009	± Remove Lg Rocks	1h
010	± Emplace OT # 3	1h
011	± Anchor OT # 3	4h
012	± Inspect	2h
013	± Test	1h
014	± Repair/Startup	2h

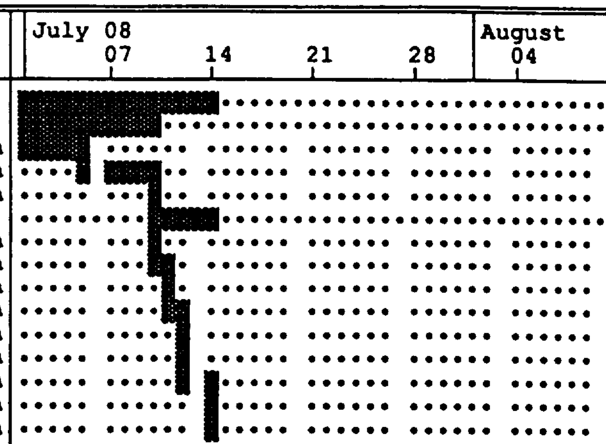


Non Critical m Milestone ____ Interrupt
Critical M Critical MS

Task Gantt
11-29-89 3:50p

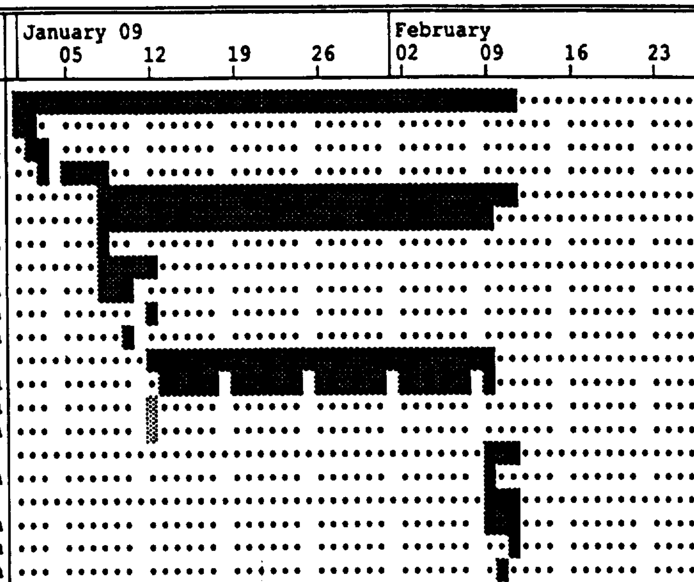
Project: I2008_7.PJ
Revision: 10

Task ID	Heading/Task	Dur
P10	I2008_7.PJ	11
001	**Lander Fac Addns**	9
002	± Offload Addns	25h
003	± Emplace Addns	25h
004	± Test Addns	3h
005	**Comm Tower**	4
006	± Offload Com Twr	1h
007	± Xport Com Twr	1h
008	± Emplace Found	2h
009	± Anchor Found	4h
010	± Emplace Twr	1h
011	± Anchor Twr	4h
012	± Connect Twr	1h
013	± Test Twr	1h
014	± Repair/Startup	2h



Non Critical m Milestone ___ Interrupt
Critical M Critical MS

Task ID	Heading/Task	Dur
P11	I2009_1.PJ	35
001	±Offload Lander	10h
002	±Xport Cargo	5h
003	±Stow Consumables	25h
004	**Prep Next Phase**	30
005	**Nuc Pwr Plt**	28
006	± Layout	2h
007	**Excavate**	4
008	± Dig pit	12h
009	± Remove sml rocks	1h
010	± Remove lg rocks	4h
011	**Surf Prep**	25
012	± Level	141h
013	± Remove sml Rocks	1h
014	± Remove Lg rocks	4h
015	**PMAD**	3
016	± Layout	4h
017	**Trench**	3
018	± Dig trench	15h
019	± Remove Sml Rocks	1h
020	± Remove lg rocks	4h



Non Critical ■ Milestone ___ Interrupt
Critical M Critical MS

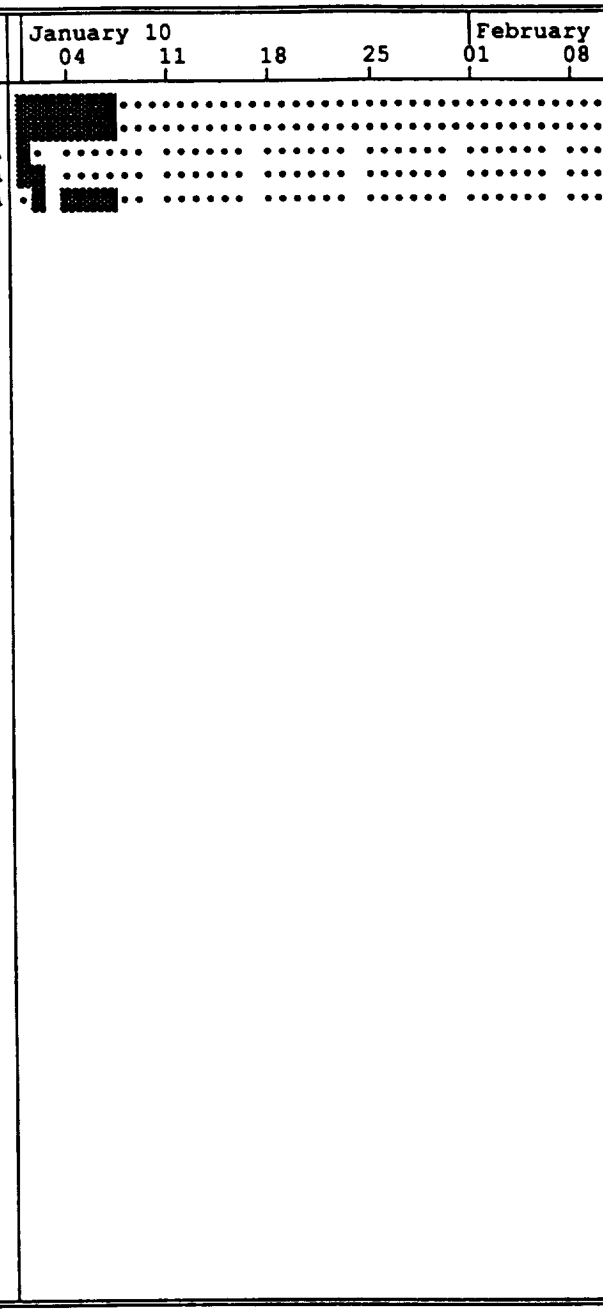
Task ID	Heading/Task	Dur
P12	**I2009_7.PJ**	63
001	**Consumables**	19
002	± Offload Consums	5h
003	± Xport Consums	5h
004	± Ingress Consums	81h
005	± Stow Consums	27h
006	**Nuclear Pwr Plt**	46
007	± Offload Nuc	5h
008	± Xport Nuc	6h
009	± Emplace bulkhead	5h
010	± Inspect	3h
011	± Backfill pit	2h
012	± Inspect	3h
013	± Emplace reactor	5h
014	± Emplace I/O Mani	1h
015	± Connect I/O Mani	1h
016	± Emplace engine p	4h
017	± Emplace engines	40h
018	± Connect heat rej	4h
019	± Connect shunts	4h
020	± Connect converte	2h
021	± Inspect	14h
022	± Emplace reflecti	40h
023	± Emplace panels	20h
024	± Connect panels	10h
025	± Anchor radiators	48h
026	± Connect manifold	2h
027	± Emplace switchin	2h
028	± Emplace utilitie	1h
029	± Connect utilitie	2h
030	± Inspect	9h
031	± Backfill trench	7h
032	± Inspect	9h
033	± Final Prep/Clean	1h
034	± Activate & Test	8h
035	± Repair Startup	16h

Non Critical m Milestone ___ Interrupt
Critical M Critical MS

Task Gantt
11-29-89 4:06p

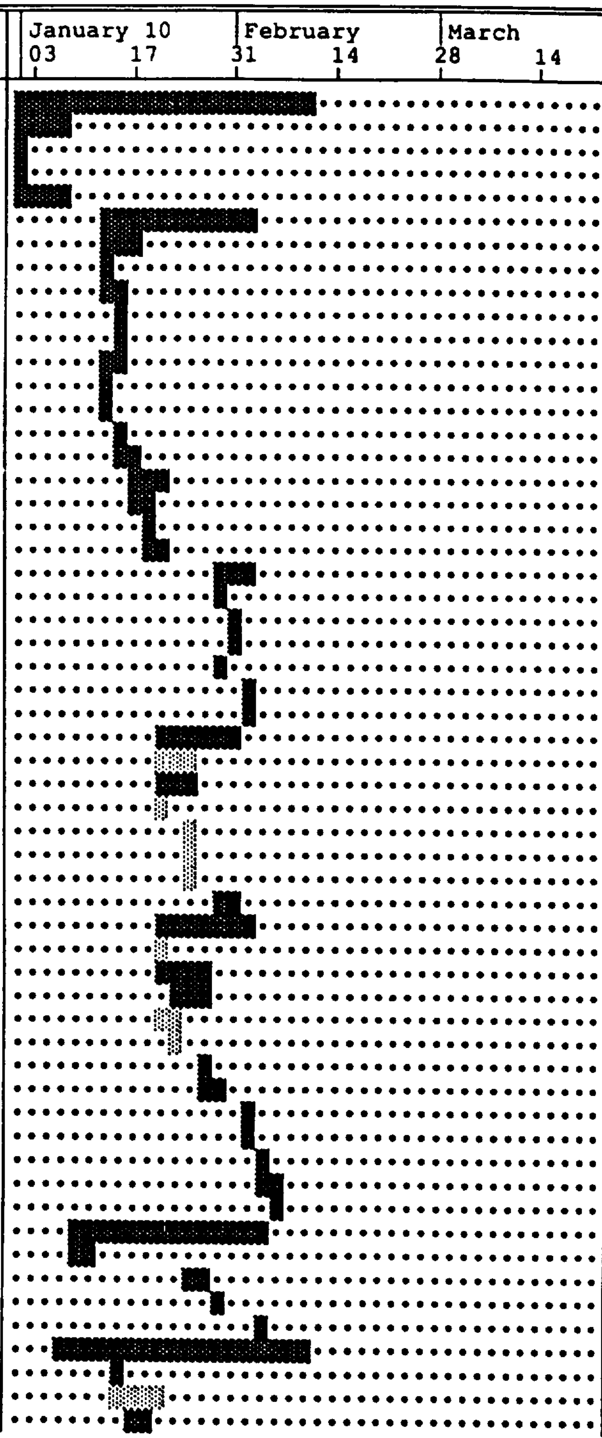
Project: I2010_1.PJ
Revision: 9

Task ID	Heading/Task	Dur
P13	I2010_1.PJ	5
001	**Consumables**	6
002	± Offload Consums	5h
003	± Xport Consums	5h
004	± Stow Consums	25h



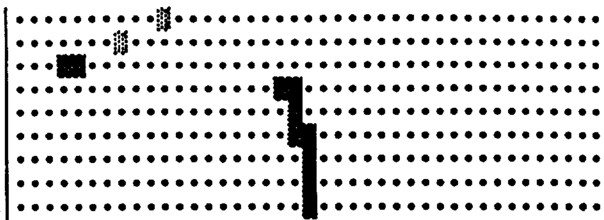
Non Critical m Milestone ___ Interrupt
Critical M Critical MS

Task ID	Heading/Task	Dur
P14	**I2010_7.PJ**	35
001	**Consumables**	6
002	± Offload Consums	5h
003	± Xport Consums	5h
004	± Stow Consums	25h
005	**LLOX Facility**	19
006	**LLOX Plant**	5
007	± Layout	2h
008	**Surf Prep**	2
009	± Level	3h
010	± Remove sml rocks	1h
011	± Remove lg rocks	1h
012	± Offload	5h
013	± Xport	5h
014	± Emplace	5h
015	± Anchor	3h
016	**TCS**	4
017	± Emplace	5h
018	± Anchor	8h
019	± Connect	1h
020	**Bene Equip**	5
021	± Offload	5h
022	± Emplace Lg Item	1h
023	± Emplace Med Item	2h
024	± Emplace Sml Item	1h
025	± Inspect	3h
026	± Connect cables	2h
027	**Storage Tanks**	10
028	**Excavate**	4
029	± Dig	20h
030	± Remove sml Rocks	2h
031	± Remove Lg rocks	2h
032	± Emplace Tanks	1h
033	± Connect	2h
034	± Cover Tanks	5h
035	**PMAD**	11
036	± Layout	3h
037	**Trench**	6
038	± Dig trench	21h
039	± Remove sml rocks	2h
040	± Remove lg rocks	3h
041	± Emplace Cable	2h
042	± Cover Cable	10h
043	± Connect Cable	2h
044	±Inspect all syst	5h
045	±Activate/test	1h
046	±Repair/start up	6h
047	±Final prep/cl up	1h
048	**Mining Equip**	23
049	± Offload	15h
050	± Xport	5h
051	± Test	3h
052	± Repair/Start up	6h
053	**G-R Telescope**	31
054	± Layout	1h
055	**Surf Prep**	7
056	± Level	11h



Across: 1 Down: 2

057	±	Remove sml rocks	1h
058	±	remove lg rocks	1h
059	±	Offload	10h
060	±	Xport	5h
061	±	Emplace	10h
062	±	Anchor	6h
063	±	Inspect	1h
064	±	Test	1h
065	±	Repair/start up	2h



Non Critical m Milestone ___ Interrupt
Critical M Critical MS

Task Gantt
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Project: I2011_1.PJ
Revision: 6

Task ID	Heading/Task	Dur
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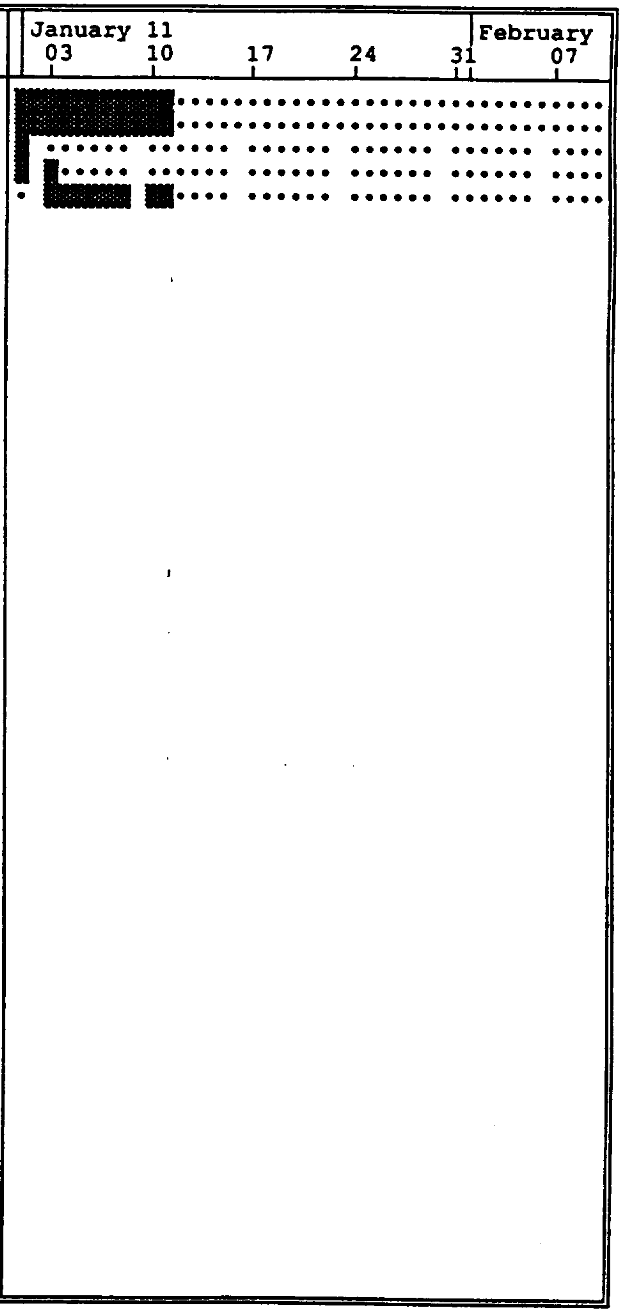
002	± Offload Consums	5h
003	± Xport Consums	5h
004	± Stow Consums	40h

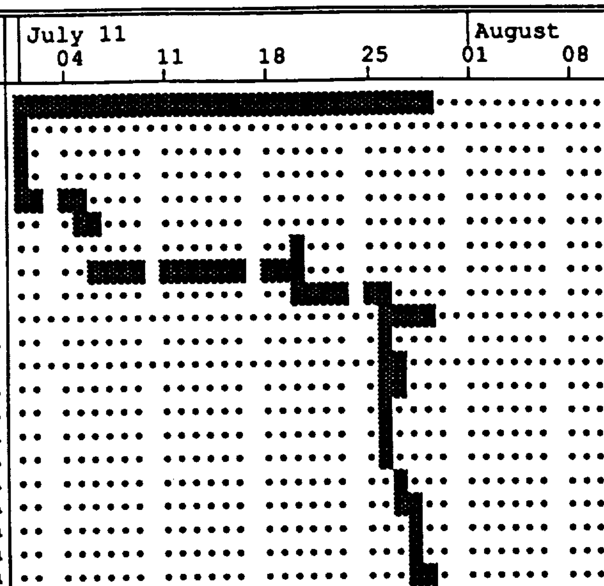


Non Critical m Milestone ___ Interrupt
Critical M Critical MS

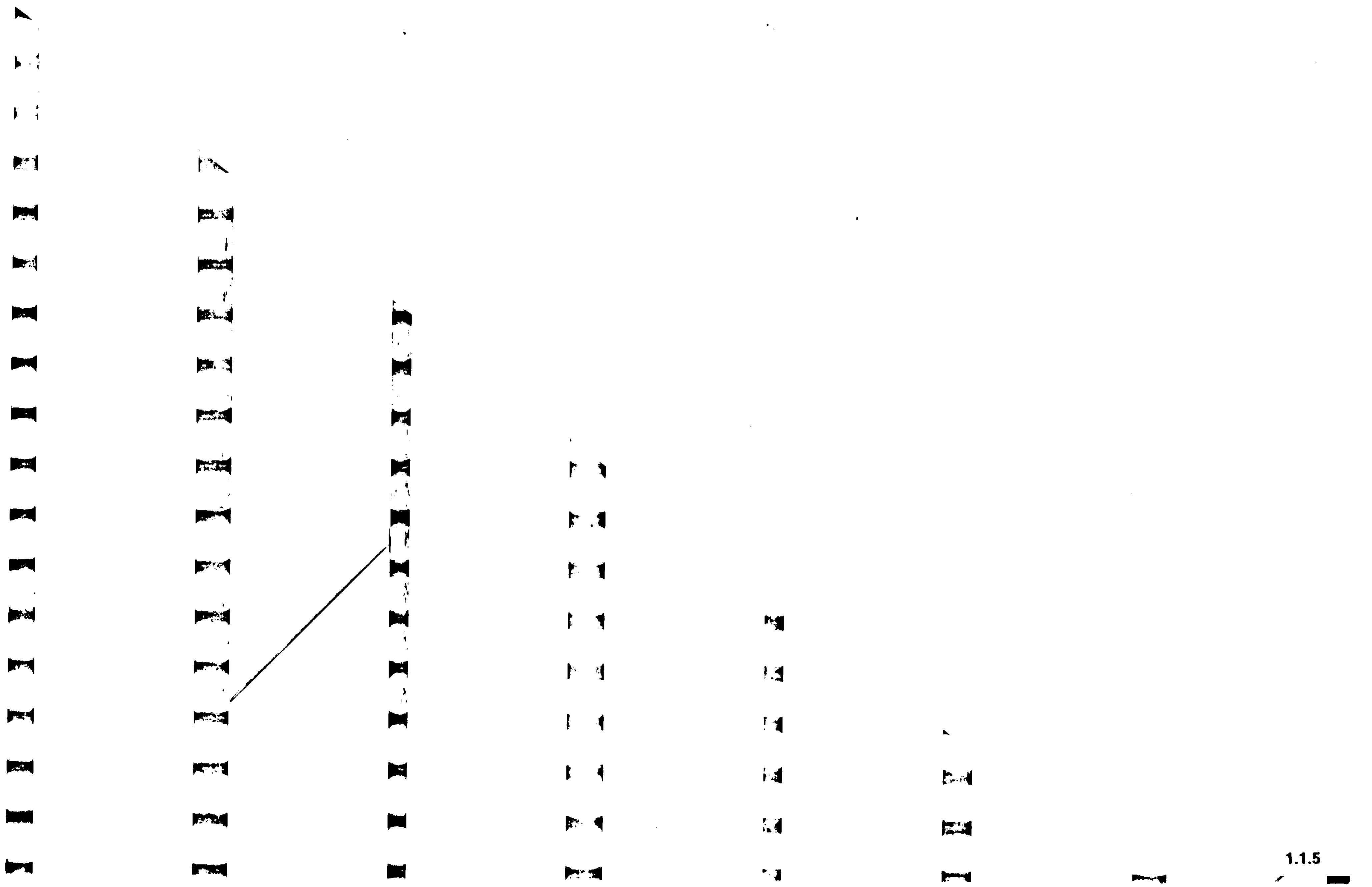
Task Gantt
11-29-89 4:45p

Project: I2011_7.PJ
Revision: 11

Task ID	Heading/Task	Dur
P16	I2011_7.PJ	24
001	**PRV**	1
002	± Offload	5h
003	± Test	1h
004	±Offload Bal	15h
005	±Transport Bal	5h
006	±Stow Sci Resup	1h
007	±Ingress Lab Inst	75h
008	±Install Lab inst	25h
009	**X-Ray Tele**	4
010	± Layout	1h
011	**Surf prep**	2
012	± Level	5h
013	± Remove sml rocks	1h
014	± Remove Lg rocks	1h
015	± Transport	1h
016	± Emplace	1h
017	± Anchor	4h
018	± Inspect	1h
019	± Test	2h
020	± Repair/startup	4h



Non Critical m Milestone ___ Interrupt
Critical M Critical MS



1.1.5

LUNAR LAUNCH AND LANDING VEHICLE TURNAROUND SCENARIO

Rex Shaffer
July, 1989

McDonnell Douglas Space Systems Company
Kennedy Space Center

CONTENTS

FIGURES

TABLES

ACRONYMS AND ABBREVIATIONS

BIT/BITE - built in test/built in test equipment

ECLSS - environmental control and life support system
EVA - extravehicular activity

FDD - functional flow diagram

GH2 - gaseous hydrogen
GDX - gaseous oxygen

H2 - hydrogen

IVA - intravehicular activity

KSC - John F. Kennedy Space Center

L&L - launch and landing
LCL - lunar cargo lander
LSCV - lunar crew sortie vehicle
LH2 - liquid hydrogen
LLO - low lunar orbit
LOX - liquid oxygen

MSFC - George C. Marshall Space Flight Center

NASA - National Aeronautics and Space Administration

O2 - oxygen

PUV - pressurized utility vehicle

RF - radio frequency

SSCMS - surface systems control and monitor system
SSE - surface support equipment

Lunar Launch and Landing Vehicle Turnaround Scenario

This section presents a lunar surface scenario for launch and landing vehicle turnaround. The scenario responds to the 1989 Exploration Study Requirements Document Lunar Evolution focused case study. It is generic in that it includes tasks that could apply to a variety of operational scenarios, depending on the lunar base development phase, base capabilities, and flight frequency. For example, during the emplacement phase, an expendable cargo vehicle may only require cargo off-loading, while a flight during the consolidation phase will require more extensive processing to include such things as the loading of lunar LOX. The turnaround scenario is for a composite vehicle in that it includes some tasks, such as crew egress, that would only apply to a manned vehicle, and others, such as cargo removal, that may not apply to manned flights.

Another objective was for the turnaround scenario to require minimum effort on the lunar surface. Our approach was to review all applicable KSC turnaround tasks and scrub these tasks down to a minimum set of turnaround requirements. A perfect flight vehicle was assumed in order to drive out those tasks that could not be deleted through improved design. The resulting scenario has been presented to the following groups; the NASA-KSC Study Team on June 23rd, the Planet Surface Systems Working Group on June 29th, the Exploration Working Group on July 13th, and the MSFC Support Team for Exploration Initiative on August 18th. Questions have been asked but no one has challenged the basic scenario content.

In developing the turnaround scenario, several assumptions were made in addition to that of a perfect flight vehicle. The more significant ones follow:

- No lunar base manpower was included for active participation in rendezvous, docking and other activities performed at the LLO node, flight vehicle approach control and landing control or ascent, except for potential range safety intervention on unmanned cargo missions. However, the presence of trained pilots and the lack of transmission delays may make the lunar base the preferred location for some or all of these activities.
- The times shown on the following flow charts are Earth ground equivalent times, and do not compensate for inherent EVA inefficiencies.
- No attempt has been made to identify the potential for parallel operations as crew size limitations may not permit them.
- Uplink capability from lunar base to flight vehicle will be provided to call up stored programs for vehicle functional testing (i.e., tank pressurization, avionics checkout, verify flight programs, cabin pressurization, etc.).
- Flight vehicles will have BIT/BITE with capability to automatically detect and isolate problems and recommend corrective action. These data will be transmitted to personnel at the lunar base.
- Time and cycle limit tracking data will be maintained automatically on the flight vehicles and surface systems elements, and will be relayed to both the lunar base and Earth.
- A database of corrective action procedures, which will identify facilities, equipment, time, and manpower required to perform tasks will be maintained at both the lunar base and at Earth. Updates will be incorporated into these procedures based on the hands-on experience of lunar crews.
- A parts inventory and location database will be maintained on Earth, but the database will be available for queries and updates by the lunar base personnel.

The Lunar Launch Vehicle Turnaround Scenario is illustrated in figure 1. The boxes of the scenario represent individual Functional Flow Diagrams (FFD) that detail the required tasks, and the times above each box equal the sums of the individual task times on each FFD. The numbers in the lower right corner of each box correspond to the number of the FFD that it represents. This is a generic scenario, and those items that apply only to a manned Lunar Crew Sortie Vehicle are indicated by an asterisk (*).

The maintenance and retest task is a violation of the perfect vehicle assumption, and is included as a reminder that no vehicle is totally perfect. Lunar surface maintenance was required on the Apollo program Lunar Excursion Module, and some level of maintenance will certainly be required for future lunar landers. Total time required for maintenance and retest will depend upon the quantity and types of failures experienced.

The total required turnaround time is estimated to be 55 hours 15 minutes, exclusive of flight and surface systems maintenance. Individual unplanned maintenance activities could range from 4 hours 50 minutes for replacing an avionics box to 38 hours 35 minutes to replace an engine.

1.0 LANDING

The landing phase FFD is shown in figure 2. This phase will begin with a visual inspection of surface systems in the landing area. The purpose of this inspection is to

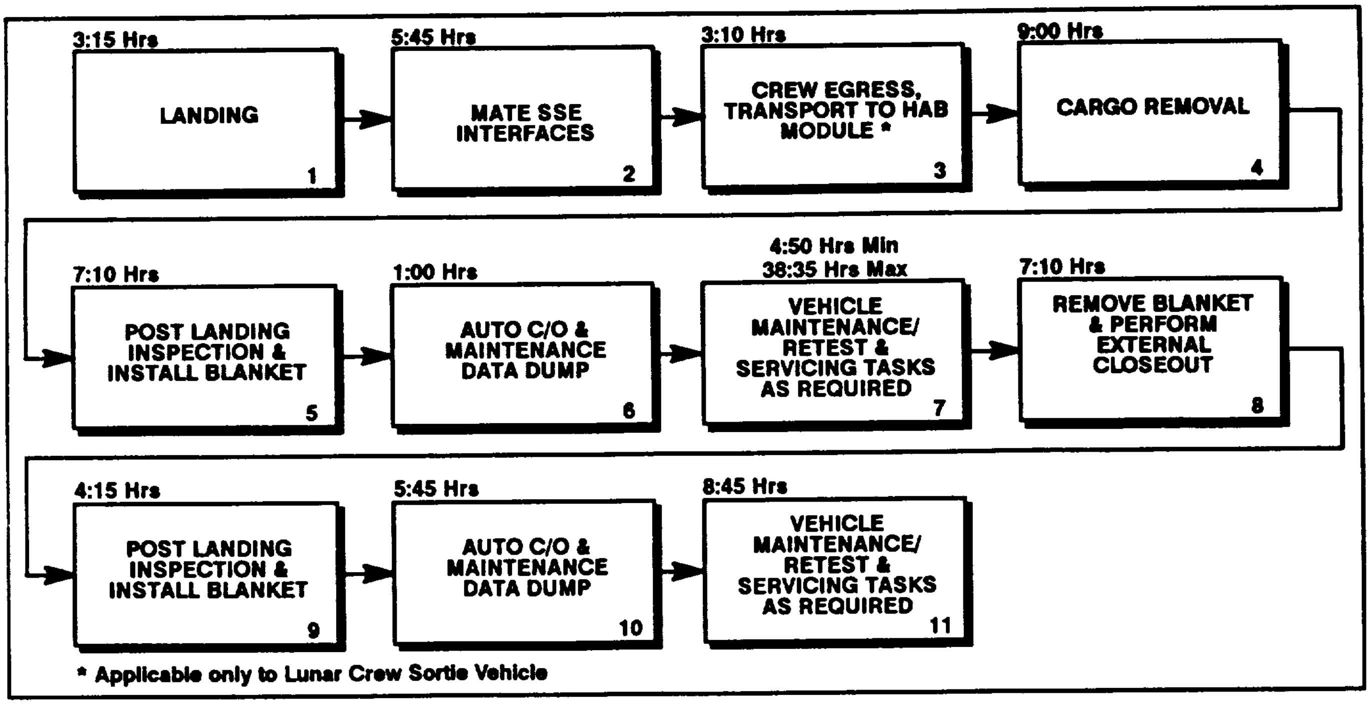


Figure 1. Lunar launch vehicle turnaround scenario

ensure that launch and landing (L&L) systems have sustained no obvious physical damage or dislocation since last checked. It will be performed remotely from the base using teleoperated television cameras with pan, tilt and zoom mounted either at multiple blast-proof pad locations or on a mobile teleoperated assistant. The estimated time is only for inspection, and includes no travel time for the teleoperated assistant, equipment activation time, or time that may be required for corrective action.

The visual inspection will be followed by an operational check of the various L&L systems and equipment to ensure its readiness to support the landing. It is assumed that these checks can be conducted remotely from the base by two people. It includes the time to

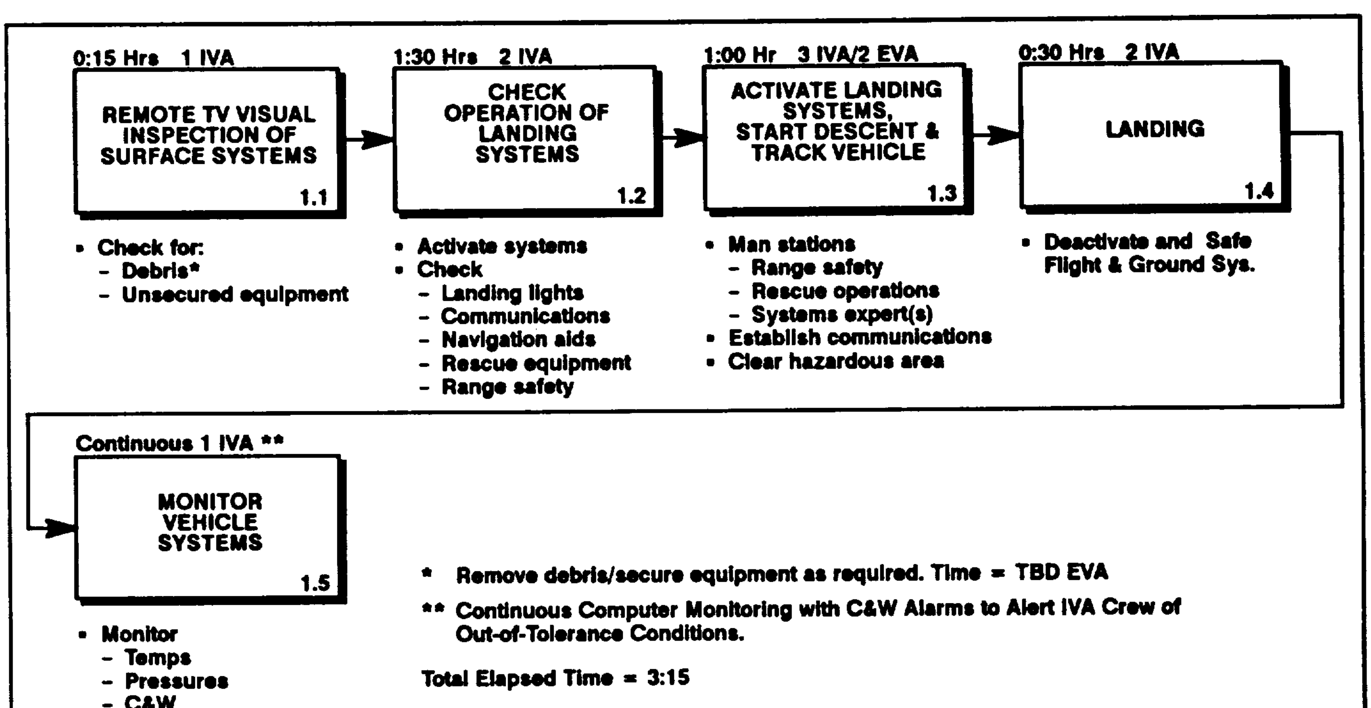


Figure 2. Landing.

activate the Surface Systems Control and Monitor System (SSCMS) from a cold-start, run self test diagnostics, and then perform similar activations and diagnostics on each of the several landing systems.

Base personnel will track and monitor the vehicle during the descent phase through touchdown, one monitoring vehicle systems and one monitoring for range safety. It is assumed that an uplink thrust termination capability will exist for errant unmanned cargo vehicles that have projected impact points on the lunar base. During landing, one person (IVA) will man the Pressurized Utility Vehicle (PUV) and two persons (EVA) will be suited up to perform rescue/contingency operations, if necessary. It is assumed that rescue equipment will be in place and ready for transport by the Pressurized Utility Vehicle.

Upon landing, voice, uplink and downlink communications will be established with the flight vehicle, and systems will be deactivated as necessary to safe the vehicle. Vehicle safing essentially completes the landing phase, which will consume an estimated 3 hours 15 minutes. After safing, one person will be on standby continuously, to respond to any caution and warning alarms or out-of-tolerance conditions flagged by the SSCMS.

2.0 MATE SSE INTERFACES

Immediately after landing, the next phase will mate interfaces between the flight vehicle and surface support equipment (SSE). The FFD for this phase is shown in figure 3. Two EVA personnel will prepare the SSE for transport to the pad. It is assumed that all required SSE can be towed, trailered, or carried to the L&L pad on the Unpressurized Rover in one 30 minute trip. At the pad, the SSE will be positioned and deployed as necessary.

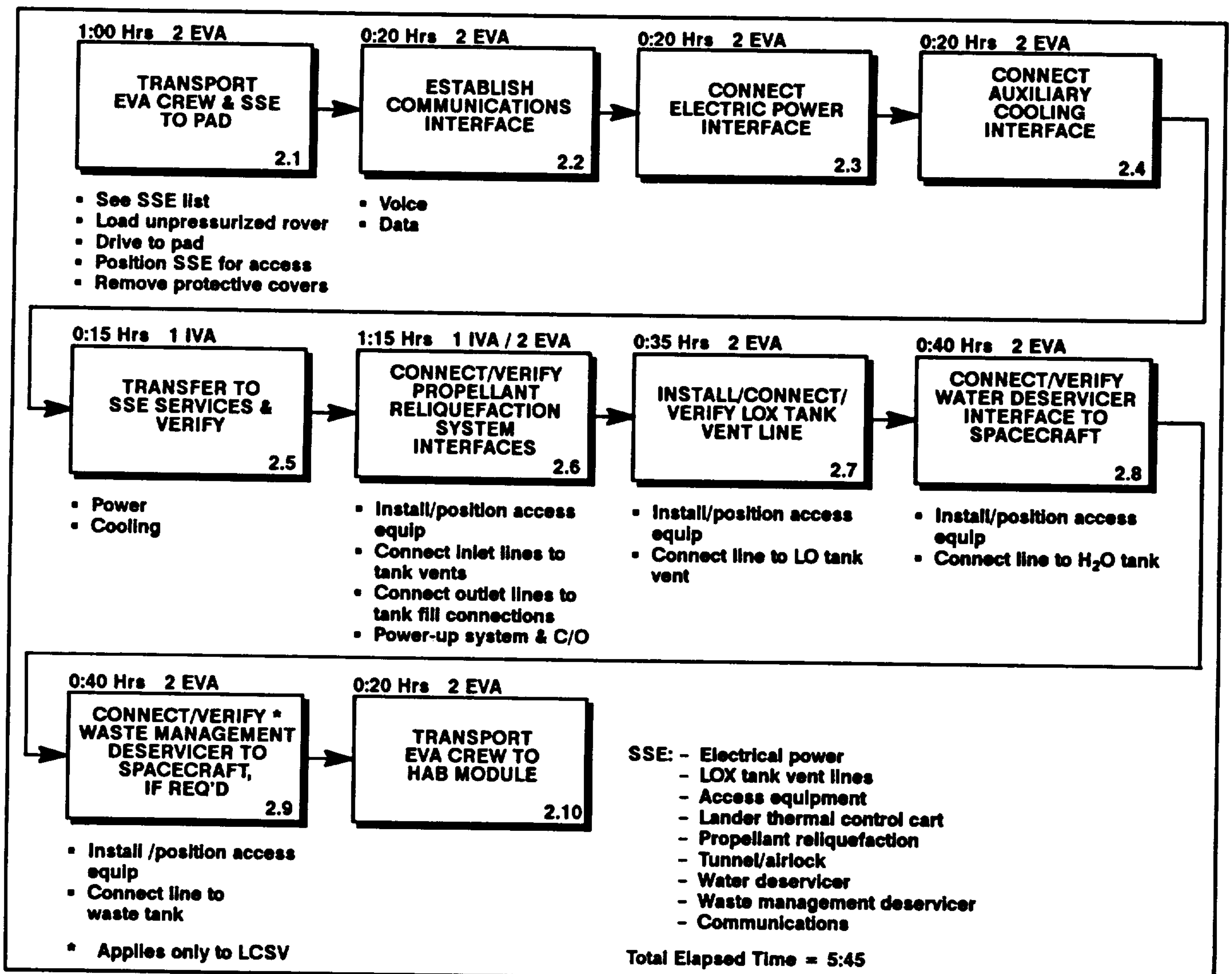


Figure 3. Landing

After deployment communications will be established, both voice and data, over either RF or hardware links assumed to exist to the pad. Access equipment will be positioned, as necessary, static electrical potentials equalized, the surface systems electrical power interfaces will be connected, and no-load verified through the flight vehicle umbilical. The Lander Thermal Control Cart will be connected to flight vehicle quick-disconnects and checked for leakage. Flight vehicle systems will then be transferred to and operation verified on surface systems power and cooling.

Access equipment will be repositioned as required to gain access to the flight vehicle propellant vent and fill connections. It is assumed that the flight vehicle will have a single ground interface each for GH2 vent, LH2 fill, GOX vent and LOX fill. The Hydrogen and Oxygen Boiloff Capture and Reliquefaction Systems will be connected and activated. These systems will permit propellants to be stored on the surface in the flight vehicle tanks. When lunar LOX becomes available, late in the lunar base consolidation phase, LOX reliquefaction may no longer be necessary, and a LOX Boiloff Vent Line can be used to carry GOX from beneath and prevent any collection under the thermal blanket.

Access equipment will be repositioned and connections made between the flight vehicle water tank and the Fuel Cell Water Deservicer. Similarly, interfaces will be mated to the Waste Management System Deservicer, if required. The short duration of round trips between the lunar surface and LLO may not warrant inclusion of a full Waste Management System for the Lunar Crew Sortie Vehicle (LCSV).

Upon completing the connection and verification of the SSE interfaces, the EVA crew will return to the lunar base. Total elapsed time for this phase is estimated to be 5 hours 45 minutes.

The next FFD assumes that the PUV will have a capability for elevating its pressurized cabin to the level of the LCSV hatch, thereby permitting it to mate directly with the LCSV and obviating the need for a separate Tunnel Ramp. This capability, illustrated in figure 4, may necessitate the need to include deployable stabilizers and/or a leveling system.

For mating with pressurized structures, it is assumed that the PUV will have a Space Station Freedom docking mechanism on the end. The docking mechanism will contain a bellows, which will permit sufficient freedom of movement to effect a proper mating.

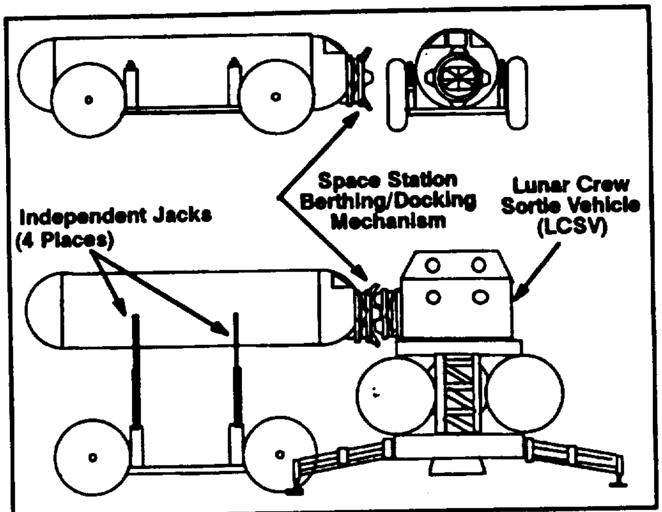


Figure 4. Pressurized Utility Vehicle docking with LCSV crew module.

3.0 CREW EGRESS & TRANSPORT TO HABITAT

Once the flight vehicle is operating on surface utilities, the next phase, applicable only for a manned LCSV, will be to transfer the flight crew to the habitat. The FFD for this phase is shown in figure 5. An IVA operator will enter the PUV, activate and prepare it for travel by sealing its hatches, undocking from the habitat, retracting stabilizers and lowering the cabin, if elevated.

The operator will drive to the pad, position the utility vehicle beneath the LCSV hatch and prepare it for docking by extending stabilizers and leveling the PUV as necessary, equalizing static electrical potentials and elevating the pressurized cabin approximately 6 meters to the level of the LCSV crew module hatch. He will activate the docking hatch mechanism and leak check the pressure seal. Pressures between the PUV and the LCSV crew compartments will be equalized, hatches will be opened, the LCSV crew will egress into the PUV, and hatches will be closed. The PUV will be undocked from the LCSV and prepared for travel as described above.

The operator will drive to the habitat, position the PUV at the hatch, prepare the vehicle for docking, and dock as described above. The operator will power-down and secure the PUV, and the flight vehicle crew and operator will egress to the habitat, closing hatches behind them.

Total elapsed time for this phase of the turnaround scenario is estimated to be 3 hours 10 minutes. Only one IVA operator is identified; the flight vehicle crew members are not considered to be "base personnel" until their first entry into the habitat.

4.0 CARGO REMOVAL

Cargo removal, illustrated by the FFD in figure 6, is the next phase of the turnaround scenario. It was assumed that the cargo will consist of a package or container that will be removed from the flight vehicle as a single unit using a single point lift in one removal

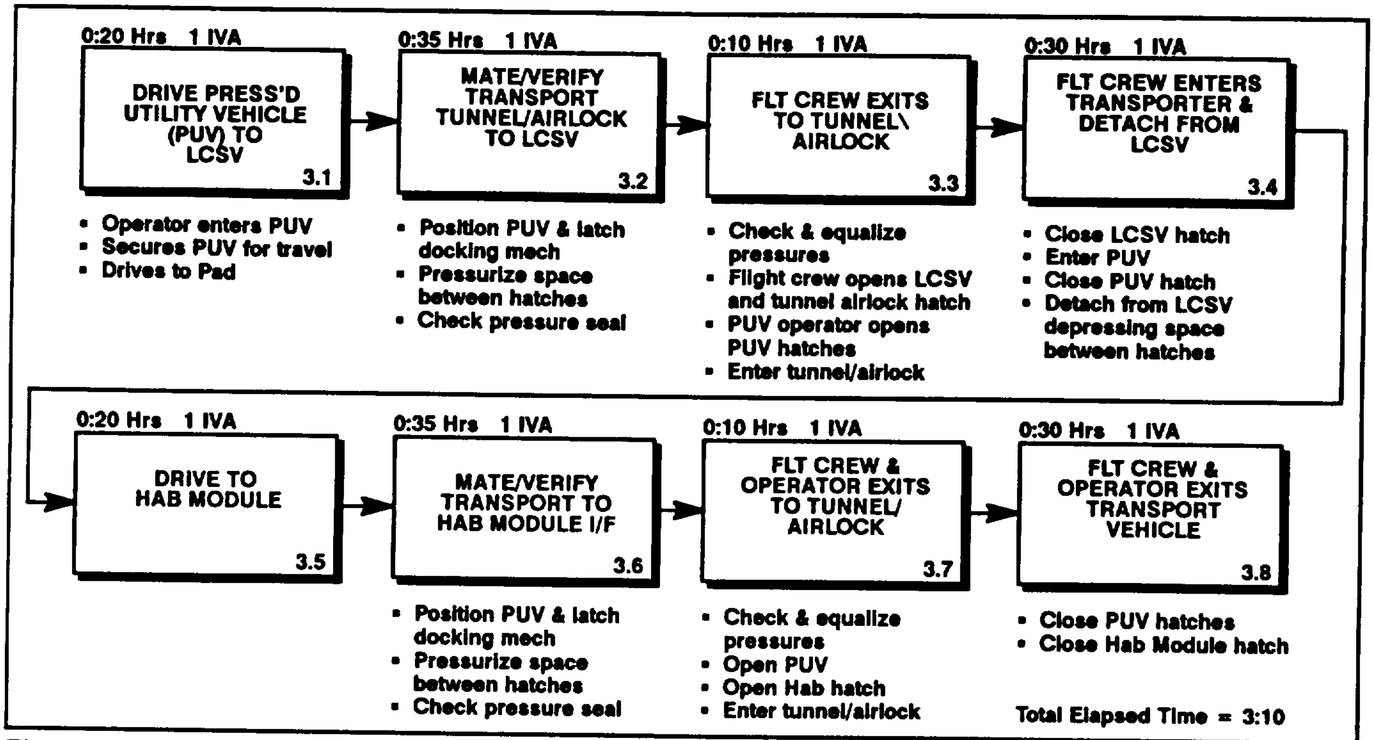


Figure 5. Crew egress and transport to habitat

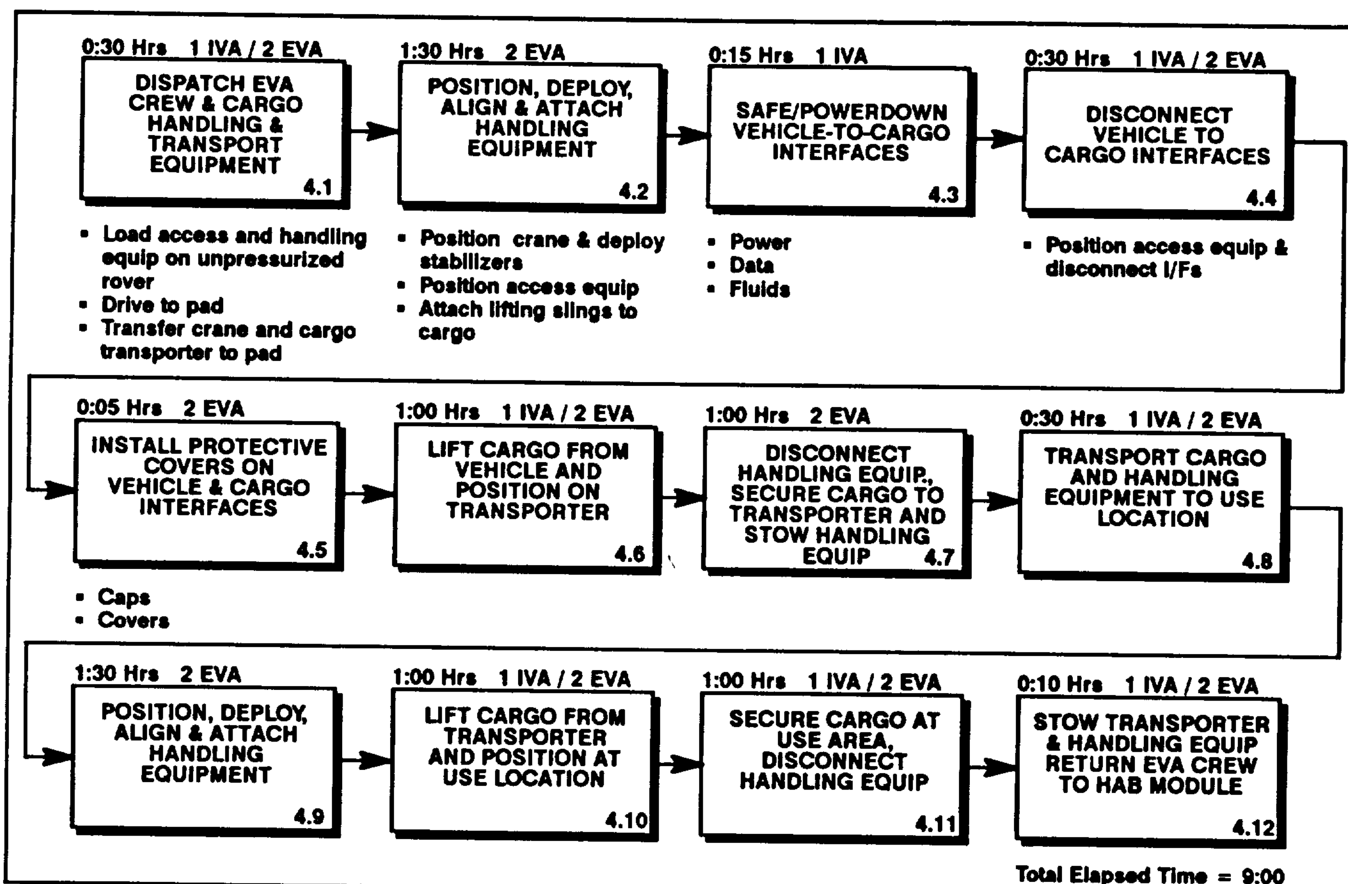


Figure 6. Cargo removal

operation. The lift was assumed to be accomplished by a crane similar to the Eagle Engineering concept shown in figure 7. It was also assumed that the cargo weight and center of gravity will be within handling tolerances, so that cargo shifts that will occur when it lifts clear, can be controlled by a two-man tag line crew. Another assumption was that the cargo must have power and cooling as long as possible.

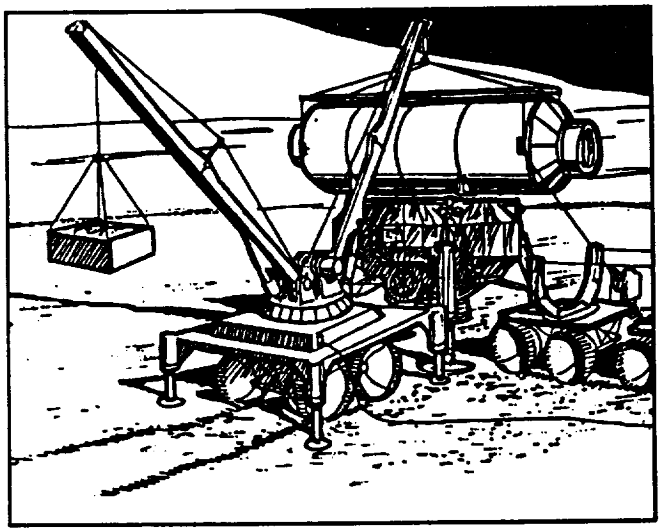


Figure 7. Cargo unloading with mobile crane

To initiate this phase, the two man EVA crew will load access and handling equipment onto the Unpressurized Rover, and will drive to the pad. Meanwhile, the IVA teleoperator crew member will drive the Crane and the Cargo Transporter to the pad. The Crane will be positioned and stabilizers deployed by teleoperation, and the EVA crew will position access equipment and attach lifting equipment to the Crane and then to the cargo. Cargo power and cooling will be shutdown and the interfaces safed by the IVA crewman, and the EVA crew will break and place protective covers on the cargo-flight vehicle interfaces connectors. It was assumed that these interfaces are in a common, easily accessed location. Tag lines will be installed and access platforms moved, as necessary.

The IVA crewman will remotely release the cargo retention latches. While the EVA crewmen stabilize the cargo with tag lines, the IVA crewman will lift the cargo vertically, traverse, and then lower the cargo onto the transporter. The EVA crewmen will reposition access equipment, remove and stow the lifting equipment, and secure the cargo to its transporter. The availability of remotely operated cargo retention latches on the Cargo Transporter and tapered guide pins to aid alignment, will simplify this operation.

The EVA crewmen will stow all handling and access equipment and return to the base on the Unpressurized Rover. Meanwhile, the IVA crewman will prepare the Crane for travel and drive it and the Cargo Transporter to their base destination. Upon arrival he will position the Crane and prepare it for lifting while the EVA crewmen again position access equipment and reconnect the lifting hardware between the cargo and the Crane. With the EVA crewmen again manning tag lines, the IVA crewman will release cargo retention latches, lift, traverse, and position the cargo at its use location. The EVA crewmen will secure the cargo, as necessary, and disconnect and stow all handling and access equipment. They will uncap and mate interface connectors, and the IVA crewman will activate and verify system operation using base utility services. The EVA crewmen will then return the Unpressurized Rover and equipment to its storage location, and re-enter the habitat, while the IVA crewman parks the Crane and the Cargo Transporter.

Total elapsed time for this phase is estimated to be 9 hours. Several areas of concern are the cargo center-of-gravity with respect to the cargo lift points, and the ability of the EVA crewmen to control the cargo using tag lines as the cargo is lifted from the flight vehicle.

5.0 POST LANDING INSPECTION AND BLANKET INSTALLATION

After the cargo has been removed, the flight vehicle will be inspected and its thermal blanket installed to protect the flight vehicle from solar heat and micrometeoriods during its long-term storage on the lunar surface. The FFD for this phase is shown in figure 8. Two EVA crewmen will remove the blanket and related equipment from storage, load it onto the Unpressurized Rover, (the availability of lifting device such as a fork lift is assumed), and drive to the launch and landing pad. Upon arrival at the pad, the crew will drive around the flight vehicle, inspecting it for visual damage and noting any maintenance items required. A television camera, with pan, tilt, and zoom, will be mounted on a telescoping mast attached to the Unpressurized Rover. It is assumed that flight vehicle surfaces will not require pre-inspection cleaning. The video inspection could be performed as a parallel operation from inside the habitat if a suitable teleoperated vehicle or teleoperated assistant were available.

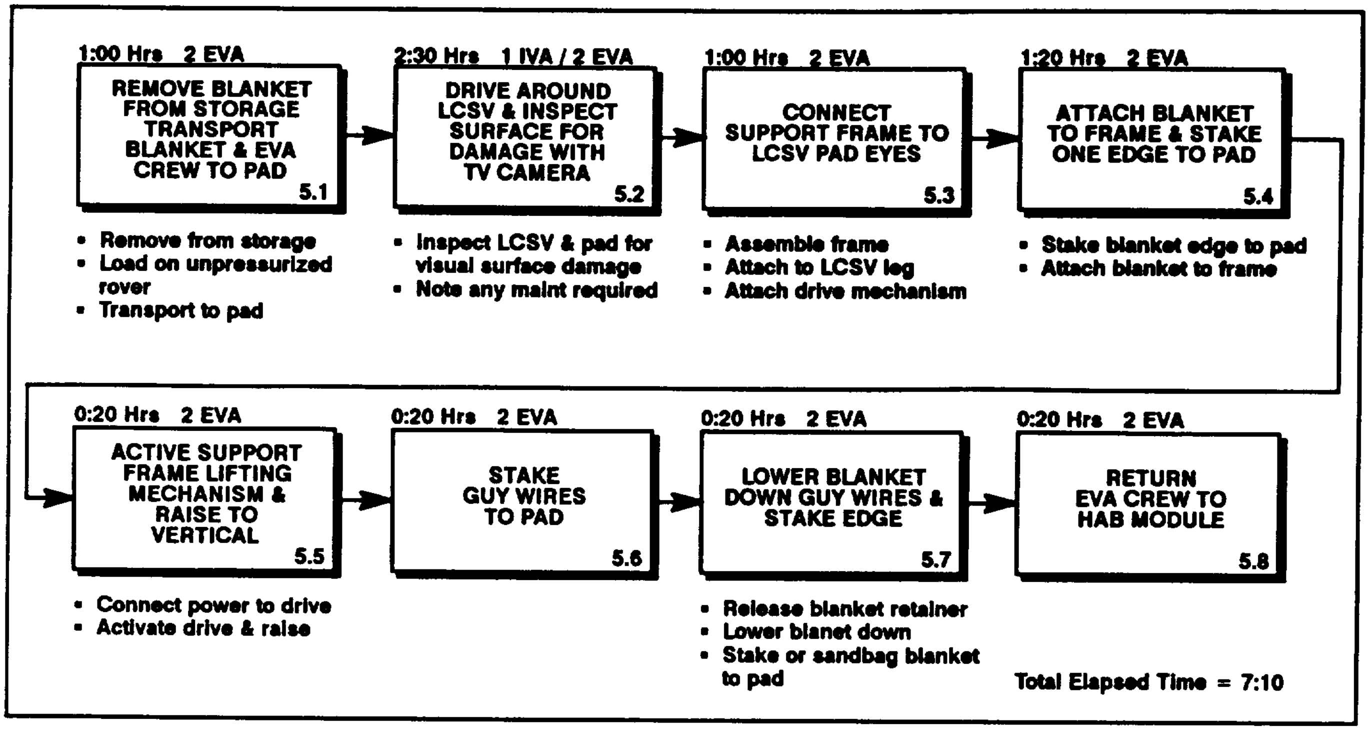


Figure 8. Post landing inspection and blanket installation.

Upon completion of the visual inspection, thermal blanket erection will be initiated. This scenario was based upon an assumed configuration, illustrated in figure 9, and different configurations may cause significant timeline variations. The blanket support frame will be assembled, positioned on the surface, attached to opposite flight vehicle lander legs, and the erection drive mechanism attached. The near-side edge of the blanket will be secured to the surface by stakes or weights and the far-side attached to the frame guy wires. The frame will be erected to an upright position, deploying the near-side half of the blanket and its guy wires in the process. The far-side guy wires will be extended to form an A-frame and staked to the surface, and then the far-side edge of the blanket will be lowered to the surface and secured. The weight of the blanket compared to that of the crew will determine how lowering is to be accomplished. The erected blanket will be approximately 17 meters (55 feet) tall.

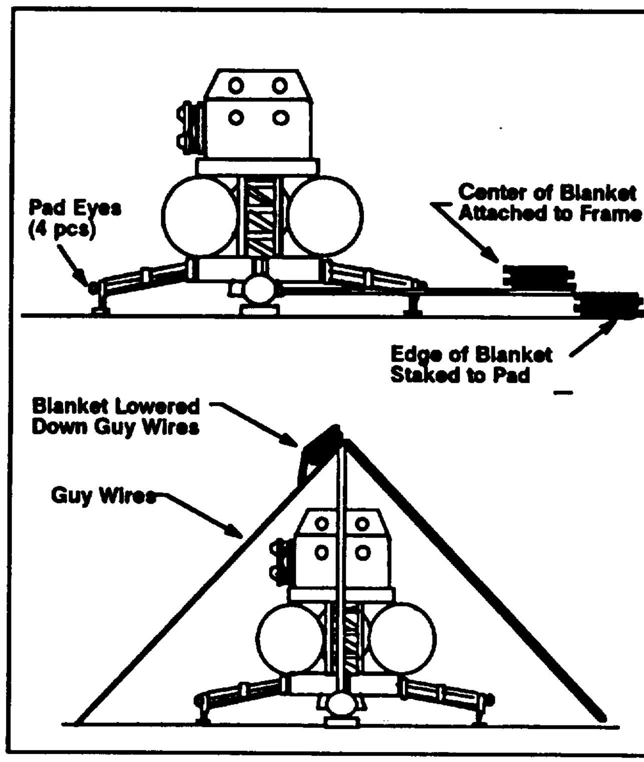


Figure 9. Blanket installation using A-frame concept.

The two-man EVA crew will return to the habitat in the Unpressurized Rover. Total elapsed time is projected to be 7 hours 10 minutes. At this point the flight vehicle is ready for long-term surface storage.

It must be realized that the time required for thermal blanket errection could change significantly with different blanket and frame designs. For example, it is conceivable that a completely automated deployment and erection design, with a folding frame similar to an automobile convertible top, could be designed. An obvious weight penalty is the cost of such an operationally efficient design. One optional concept using flexible graphite epoxy rods for support is shown in figure 10. The end opening is sufficient for erect entry of a crew member.

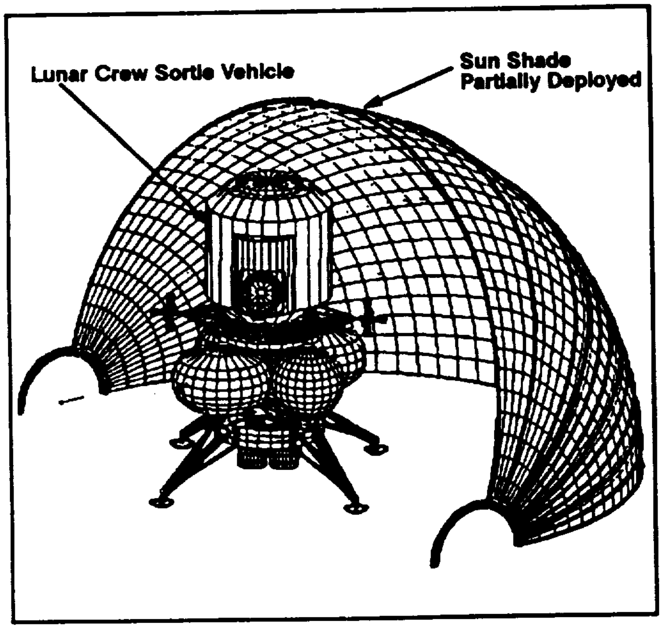


Figure 10. Optional deployable sun shade concept.

6.0 AUTOMATIC CHECKOUT AND MAINTENANCE DATA DUMP

The FFD for Automatic Checkout and Maintenance Data Dump is shown in figure 11. These operations can be scheduled any time during the flight vehicle surface storage period; however, they should be conducted during the early part of the period to acquire an early definition of the maintenance tasks that must be completed before the next flight.

The flight vehicle and surface support equipment will be activated and monitored IVA from the habitat using the SSCMS. On-board BIT/BITE will be activated for the various systems.

The results of the BIT/BITE runs will be transmitted to and displayed by the SSCMS. Anomalous performance by any system or component will be identified and required corrective action will be recommended. The

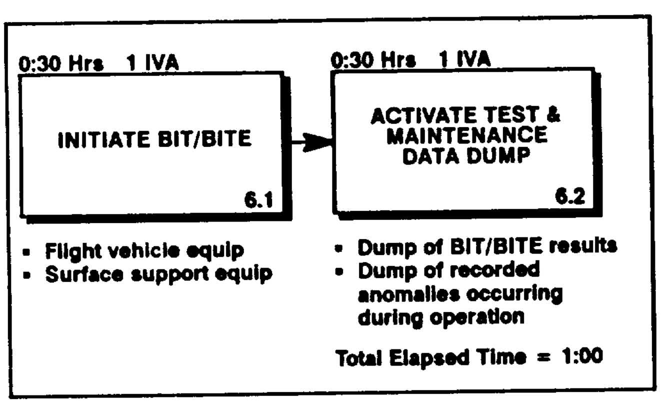


Figure 11. Automatic checkout and maintenance data dump.

system will identify any component that is projected to reach its operating time or cycle limits by the end of the next flight, and will display trend data for any component with a performance parameter trend that is approaching a performance limit.

Total task time is estimated to require one hour. Analysis of the data acquired will define the total complement of maintenance tasks for planning and scheduling.

7.0 FLIGHT VEHICLE MAINTENANCE AND RETEST

The FFD for the maintenance and retest task for flight vehicle turnaround is shown in figure 12. The scope and duration of the task (which violates the assumption of a perfect vehicle) will be determined by the number and type of failures experienced, and will vary from vehicle-to-vehicle and from flight-to-flight. The definition of the task will result from the combination of the visual inspection and the maintenance data dump. Also, the SSCMS will monitor system performance and flag any additional maintenance or repair items as they occur. The SSCMS will open problem and maintenance action files to ensure that the required maintenance tasks are accomplished. It will also identify the appropriate repair/maintenance procedures and the required manpower skills and equipment resources.

Since this scenario was to assume a perfect vehicle, the entire spectrum of maintenance tasks has not been addressed; however, what are believed to be the two ends of the maintenance spectrum have been assessed. The removal, replacement, and retest of an avionics box in the pressurized crew compartment of the Lunar Crew Sortie Vehicle, requiring 4 hours 50 minutes, represents the low end of the spectrum, and the removal, cannibalism, and replacement of an engine, requiring 38 hours 35 minutes, represents the high end.

Upon completion of the maintenance task, the SSCMS will log the location and disposition of the replaced equipment as to whether it is to be repaired, replaced, or scrapped. It will update inventory records to reflect any changes in inventory status, quantities, and/or location, and it will close the previously opened action file to complete the maintenance action.

Total elapsed time for the entire maintenance and retest task will be a variable depending upon the quantity and types of maintenance actions required. All maintenance items should be closed before the end of the flight vehicle long-term surface storage period.

The detailed FFD for the removal, replacement, and retest of an avionics box is shown in figure 13. This

sub-task was assumed to use the PUV, as described in Section 3, to avoid the need for EVA operations.

The detailed FFD for the removal, cannibalism, and replacement of an engine is shown in figure 14. The performance of such an operation in the dusty environment of the lunar surface, by persons in EVA suits which limit mobility and access, would be extremely risky, and is not recommended.

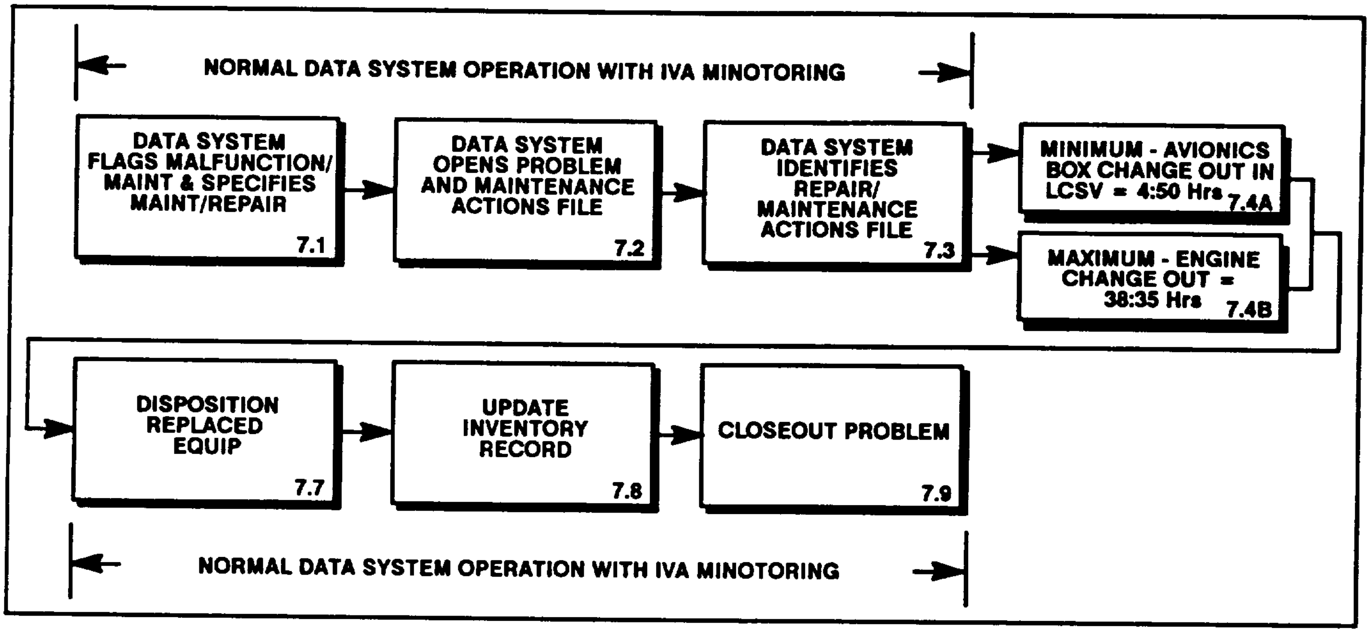


Figure 12. Flight vehicle maintenance and retest.

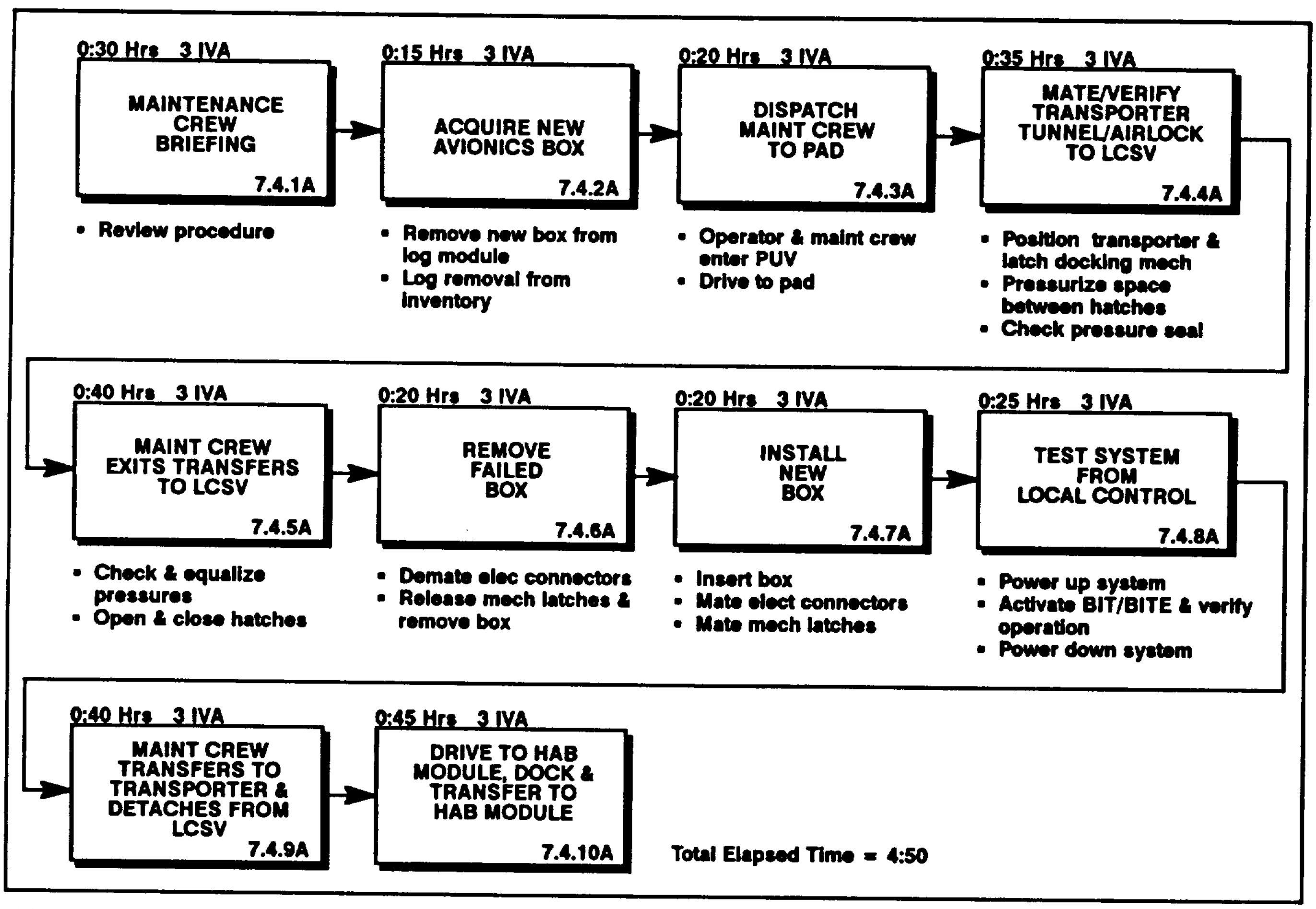


Figure 13. Avionics box removal, replacement, and retest.

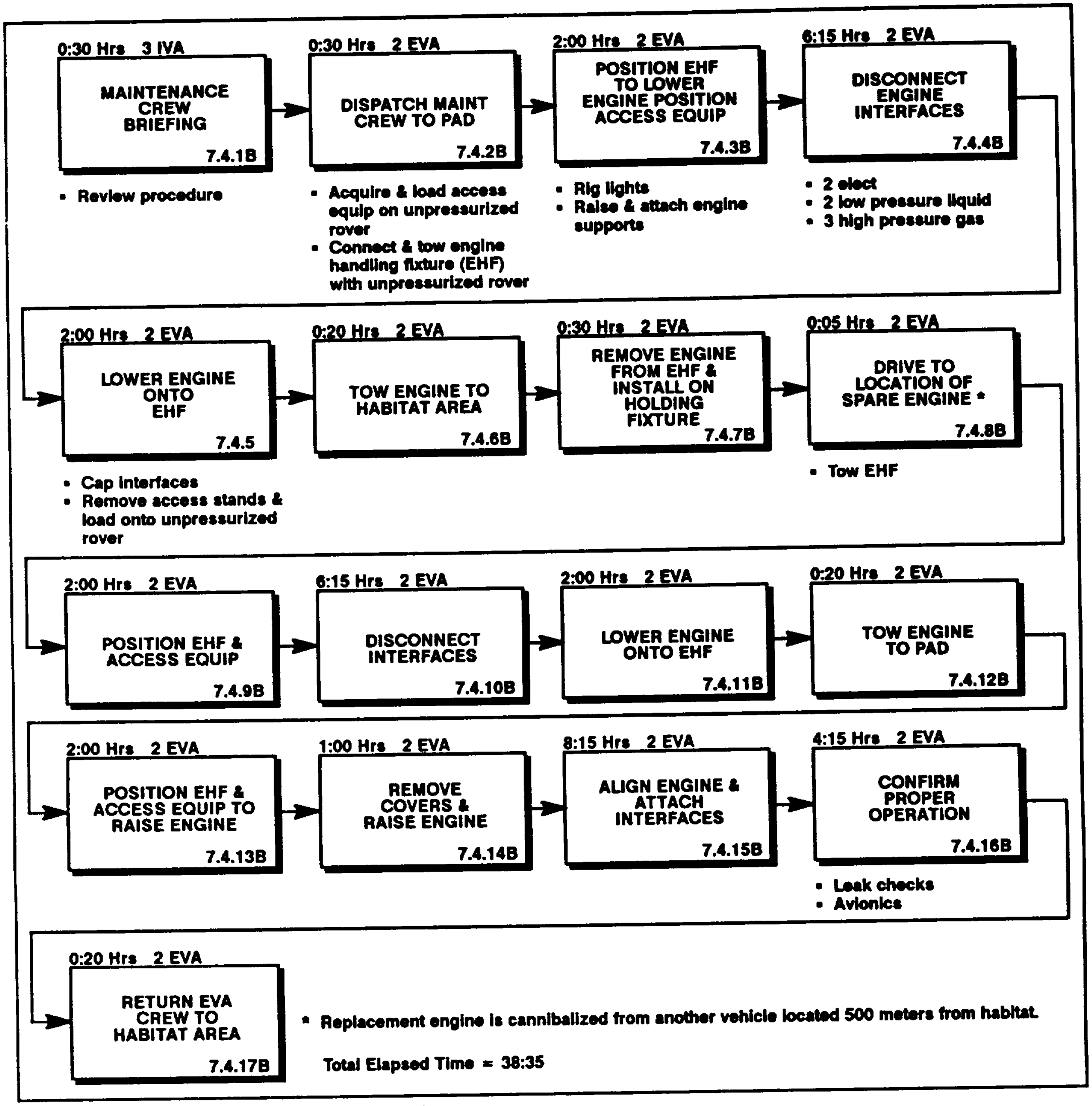


Figure 14. Engine removal, cannibalism, and replacement.

There is a continuum spectrum of possible maintenance and retest sub-tasks which could be necessary for continued use of a failed flight vehicle, and at some point it will become more economical to replace the vehicle rather than perform the maintenance. Also, the probability of in-flight failures increase with vehicle use. Programmatic decisions will be required to determine both the level of lunar maintenance to be accomplished and the frequency of flight vehicle replacement.

8.0 BLANKET REMOVAL AND EXTERNAL CLOSEOUT

The end of the flight vehicle long-term storage period will be marked by removal of its protective thermal blanket, and will be preceded by a check of the SSCMS database to ensure that all required maintenance items are closed. The FFD is shown in figure 15. A two-man EVA crew will drive to the launch and landing pad in the Unpressurized Rover. They will release the far-side

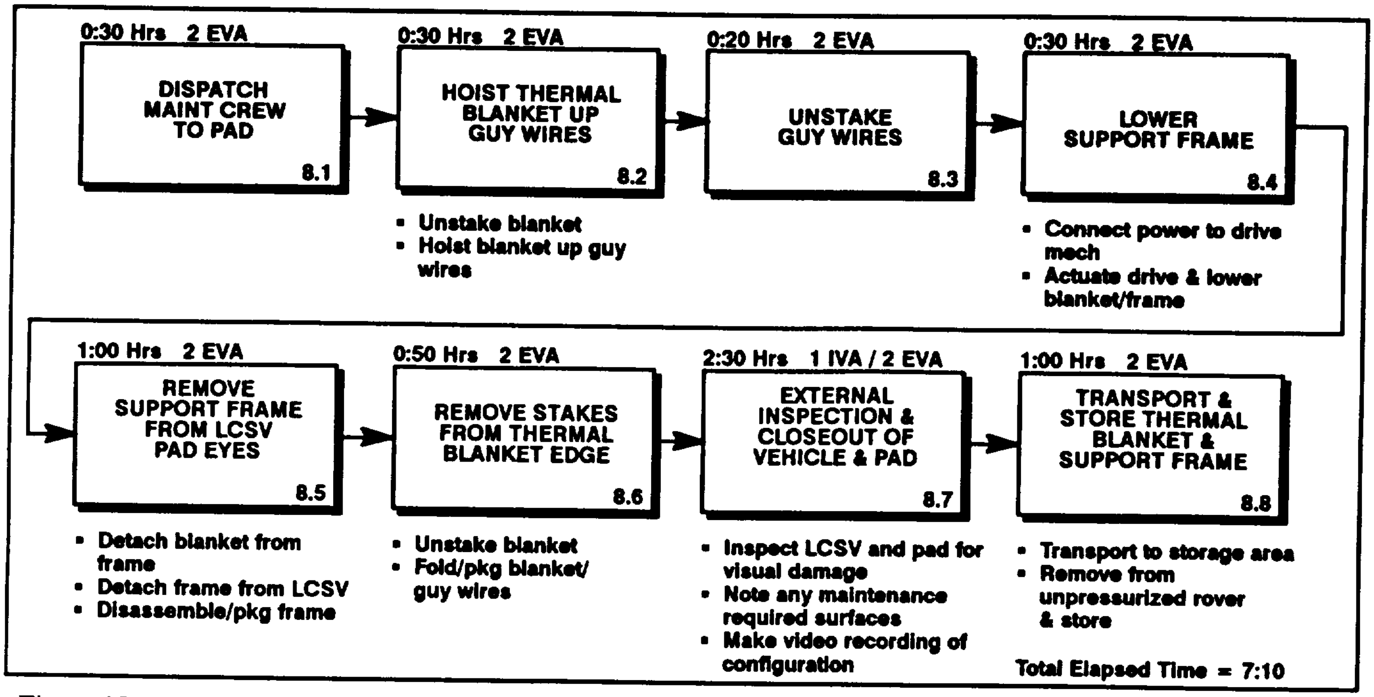


Figure 15. Blanket removal and external closeout.

surface attachments (e.g., stakes, or sandbags), and hoist the blanket far-side up the guy wires to the peak. They will then unstake the far-side guy wires, and using the drive mechanism, carefully lower the blanket support frame, with blanket attached, to the pad surface.

After the frame and blanket have been lowered, the blanket will be detached from the frame, the frame detached from the flight vehicle legs, and the frame disassembled. The surface attachments for the near-side of the blanket will be removed, and the blanket and guy wires will be folded.

Once the blanket and frame have been packaged for storage and loaded onto the Unpressurized Rover, the flight vehicle and surface systems equipment in the pad area will be visually inspected and any late maintenance items noted. A video recording of the flight vehicle external configuration and condition will be made as a closeout record. This could be accomplished as a parallel operation from inside the base by teleoperation if a suitable teleoperated vehicle or assistant were available.

The crew will then drive to the storage area, remove and store the thermal blanket and frame, and return to the habitat. Total elapsed time is projected to be 7 hours 10 minutes, but it must be realized that this task time, like that for blanket installation, could change significantly with different blanket and frame designs and removal scenarios. At this time the flight vehicle will be ready to receive any up-cargo.

9.0 CARGO–VEHICLE INTEGRATION

Upon removal of the thermal blanket, the flight vehicle will be ready for the installation of any bulk cargo to be shipped from the lunar surface, and for cargo-vehicle interface verification. The FFD for this activity is shown in figure 16. At the base logistics area, the two-man EVA crew will position access equipment and man tag lines, while the IVA crewman, via teleoperation, will position the crane and prepare it for cargo lifting (as described in Section 4, Cargo Removal). It is assumed that the cargo pallet/container will have been previously loaded and that its center-of-gravity will be within handling tolerances. Cargo-base interfaces will be deactivated, safed, broken, and protective covers installed. The cargo will be lifted onto and secured to its transporter and lifting equipment loaded onto the Unpressurized Rover. All equipment will then be driven to the pad and deployed for cargo installation.

At the pad, the EVA/IVA crew will jointly deploy and position equipment to prepare for cargo hoisting. The cargo lift will be executed by the IVA crewman via teleoperation with two EVA crewmen manning tag lines. It is assumed that the flight vehicle will have alignment aids, such as tapered pins, to assist in the alignment and mechanical mate of the cargo. Once the cargo is in place, mechanical latches will be remotely activated by the IVA crewman, and will be visually inspected by the EVA crewmen. Interface connectors will be uncapped and mated by the EVA crewmen, and

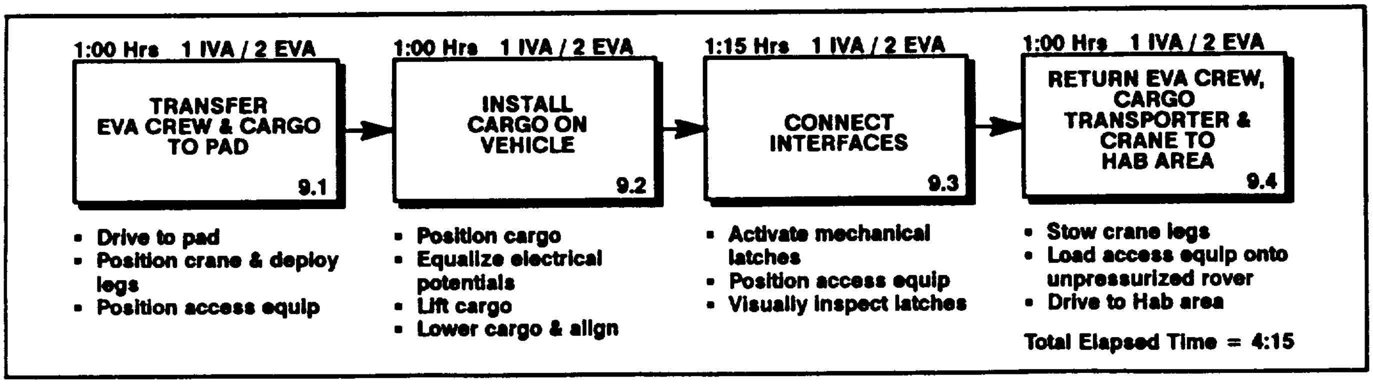


Figure 16. Cargo-vehicle integration.

the interfaces activated and tested by the IVA crewman.

The vehicle and lifting hardware will then be transported to the habitat area. Equipment will be stowed and stored, vehicles parked, and the EVA crewmen will return to the habitat. Total elapsed time is estimated to be 4 hours 15 minutes. Unmanned vehicles will now be ready for launch countdown and the LCSV will be ready for internal closeout.

10.0 INTERNAL PRELAUNCH CLOSEOUT

The FFD for the internal prelaunch closeout phase is shown in figure 17. This phase applies only to the manned LCSV, however, automated system checkout will be required to verify pre-countdown readiness of all vehicles. The LCSV Environmental Control and Life Support System (ECLSS) will be activated and will include a check of the SSCMS to ensure that all crew cabin maintenance items are closed. Critical crew module parameters (e.g., pressures, temperatures, oxygen level) will be monitored to ensure readiness for personnel ingress. Meanwhile, tools and equipment will be loaded into the PUV.

When the LCSV ECLSS parameters are acceptable, two IVA crewmen will detach the PUV from the habitat, drive to the pad, and position and dock the

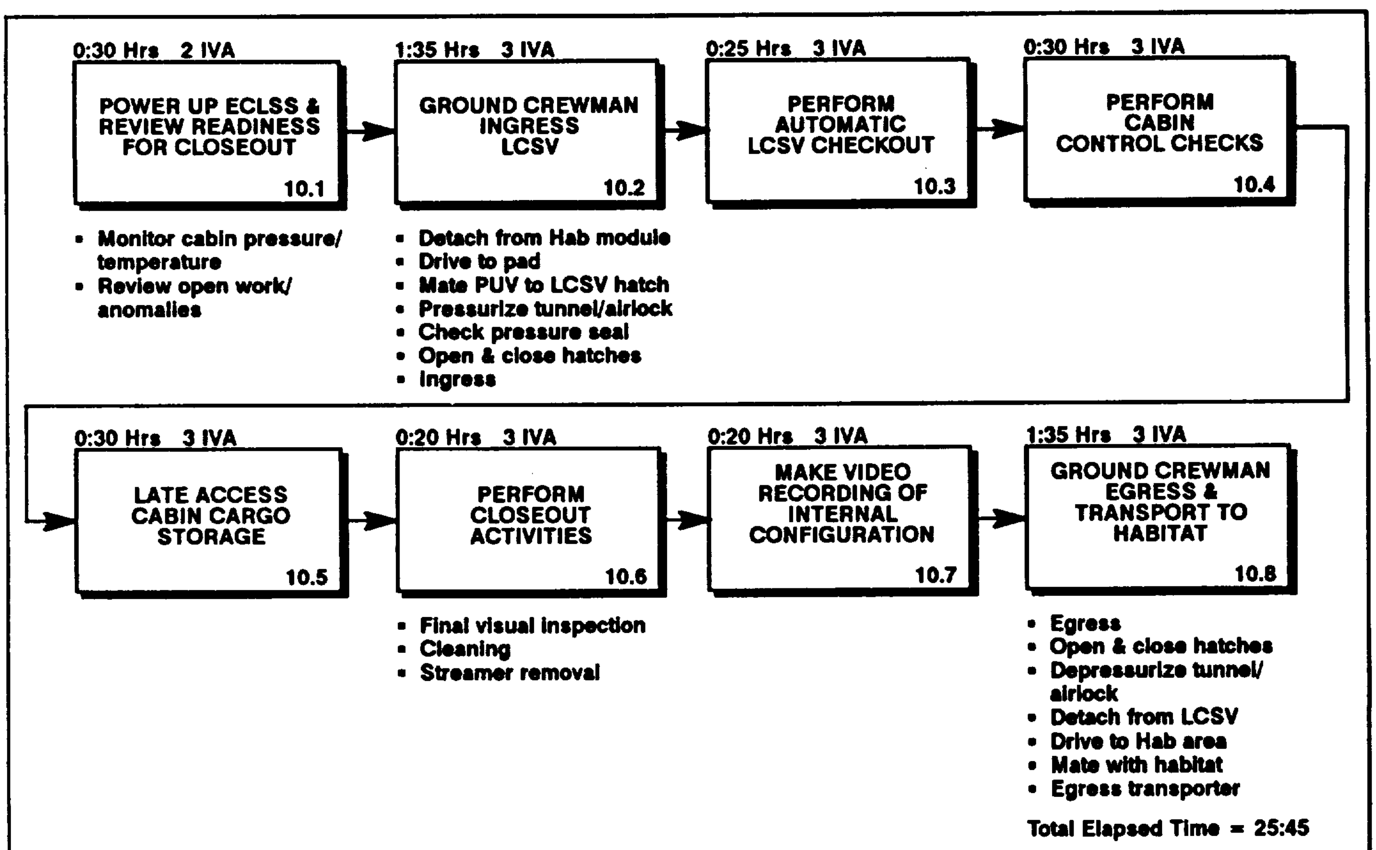


Figure 17. Internal prelaunch closeout.

PUV to the LCSV crew module hatch as described in Section 3.0. They will enter the LCSV crew module, and in conjunction with the SSCMS operator at the habitat, will perform an automated LCSV checkout to verify system readiness for launch countdown. This will be followed by module control checks, to configure the crew module for countdown. Stowage of late cabin cargo items will then be completed. These items will include anything excluded from the prepackaged bulk up-cargo and items subject to spoilage or deterioration in an uncontrolled environment. Final crew module closeout will include such things as a final visual inspection, final cleaning, and removal of "remove-before-flight" items. A video recording of the final internal configuration will be made as a closeout record.

The crew will then egress to the PUV, demate, and return to the habitat. Total elapsed time is estimated to require 5 hours 45 minutes.

11.0 LAUNCH COUNTDOWN/LAUNCH (LCSV)

The launch countdown FFD, shown in figure 18, is described for a manned LCSV, as it contains several tasks that will not be required for the unmanned vehicles. Before lunar LOX is available, LOX will be stored in the flight vehicle tanks. This scenario assumes the availability of lunar LOX and a requirement to load LOX before launch.

A two-man EVA crew will exit the habitat, load access and other equipment onto the Unpressurized Rover, pick up a loaded LOX tanker (assumed ready to travel), and drive to the launch and landing pad. Access equipment and the LOX tanker will be positioned, static electrical potentials equalized, LOX fill lines unstowed, protective covers removed, and interfaces to the flight vehicle mated. A single LOX fill connection is assumed. The fill lines will be slowly filled to each flight vehicle tank shutoff valve, with boiloff bled off at that point through the LOX vent system. When chilldown is complete, bleed valves will be closed, a main LOX tank shutoff valve will be opened, and the tank will be filled via a pressure feed. Once flow is established, the fill rate can be increased, and as the desired load is approached, the fill rate will be decreased until the desired load is attained. The process will then be repeated for the other main LOX tank and the fuel cell LOX tank. It may be possible to load all three tanks simultaneously. Fill time will depend upon the size of the transfer line and the pressures used. Smaller transfer lines will reduce the mass to be chilled and reduce boiloff but will increase the required fill time.

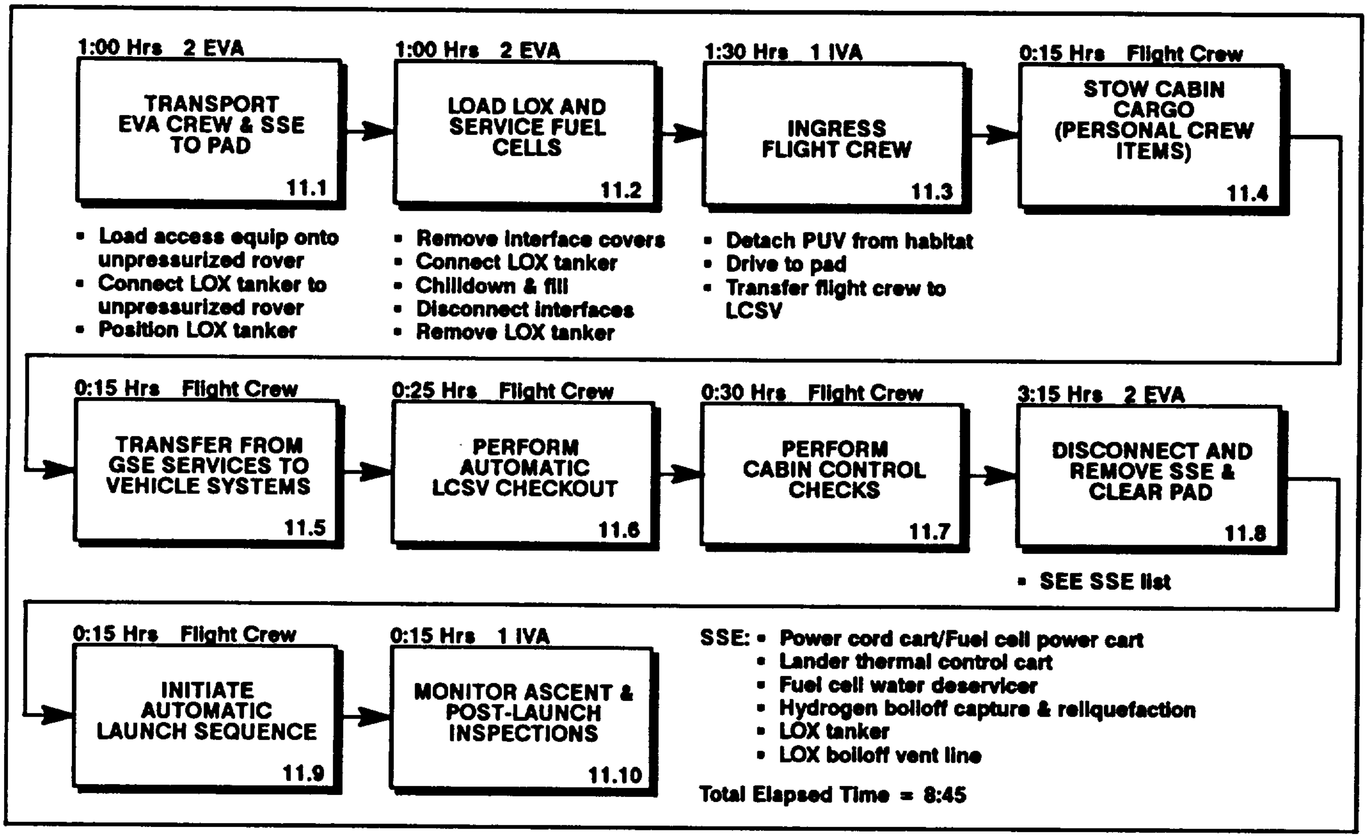


Figure 18. Launch countdown and launch.

When the fill is complete, the fill line will be drained back into the LOX tanker, and fill lines disconnected and stowed. EVA suits must protect personnel from injury due to contact with the cold LOX transfer line. EVA personnel will drive the Unpressurized Rover and the LOX tanker to the LOX storage area. The tanker will be connected to the storage tank and residual LOX returned to the tank in a manner similar to that described for the flight vehicle.

Once flight vehicle LOX boiloff has stabilized, the flight crew will enter the PUV with one base crewman. They will detach from the habitat, drive to the pad, and dock with the LCSV crew module hatch. The flight crew will ingress the LCSV crew module and stow personal items. The flight vehicle will then be transferred from surface systems services to flight vehicle services, and an automated test of vehicle systems conducted, followed by module control checks to ensure launch readiness.

Once the LOX tanker has been off-loaded of residuals and parked, the two-man EVA crew will return to the pad to disconnect and remove the equipment used to provide surface systems services to the flight vehicle. These items will be placed in storage and the two crewmen will return to the habitat.

The automatic launch sequence will be initiated with T-0 timed to achieve lunar orbit rendezvous with the Lunar Piloted Vehicle. The vehicle downlink data will be monitored during ascent, and after launch, the teleoperated assistant will be dispatched to the pad for a post-launch video inspection. Total elapsed time is expected to be 8 hours 45 minutes.

12.0 SURFACE SYSTEMS HARDWARE

Several items of SSE will be required to accomplish the turnaround scenario. Some of these items are unique to the launch and landing area; however, the majority will also be required by other areas.

12.1 HARDWARE UNIQUE TO LAUNCH AND LANDING.

Table I shows a cross reference to the phase of the turnaround scenario that requires the use of each item unique to Launch and Landing operations. The need for each of these items is described briefly below.

Lander Thermal Control Cart. Required to dissipate heat generated by flight vehicle systems while operating from surface systems services.

H2 and O2 Boiloff Capture and Reliquefaction Systems. Required to preclude loss of the launch propellant supply while the flight vehicle is stored on the lunar surface.

Oxygen Boiloff Vent Line. Required when lunar LOX becomes available to route residual LOX boiloff to lunar vacuum so as to preclude the possibility of GOX collection beneath the thermal blanket or other enclosed structures.

Landing/Navigation Aids. Required to ensure accurate landing at the proper Launch and Landing Pad.

Thermal Blanket. Required to protect each flight vehicle from direct and reflected solar thermal

TABLE I - HARDWARE UNIQUE TO LAUNCH AND LANDING

	Landing	Mate SSE I/F	Crew Egress	Cargo Removal	Install Blanket	Auto Maint Dump	Maint & Test	Remove Blanket	Cargo Integr	Internal Closeout	C/D & Launch
Lander Thermal Control Cart		■	■	■	■	■	■	■	■	■	■
H2 & O2 Reliquefaction Systems		■	■	■	■	■	■	■	■	■	■
Oxygen Boiloff Vent Line		■	■	■	■	■	■	■	■	■	■
Landing/Navigation Aids	■										
Thermal Blanket					■	■	■	■			
Electrical Grounding System		■	■	■	■	■	■	■	■	■	■
Range Safety System	■										■
Engine Handling Fixture							■				
Pad Electrical Power	■	■	■	■	■	■	■	■	■	■	■
Blast Protection	■										■

radiation during the lunar day. It will also provide some degree of micrometeoroid protection.

Electrical Grounding System. Required to ensure that the several pieces of flight and surface systems equipment involved in an operation are at equipotential, and thereby prevent equipment damage through static discharge.

Range Safety System. Required to protect the lunar base from errant unmanned flight vehicles during launch and landing.

Engine Handling Fixture. Required if engine removal and replacement is to be accomplished on the lunar surface.

Electrical Power at Launch and Landing Pad (options include: Power Cord Cart, Fuel Cell Power Cart, and Beamed Power Cart). Required to operate electrically driven surface systems, power tools, equipment, and the flight vehicles at the launch and landing pads.

Blast Protection. Required to protect surface systems equipment and facilities in the launch and landing area from surface ejecta generated during vehicle launch and landing.

12.2 HARDWARE SHARED WITH OTHER BASE OPERATIONS.

Those items needed for launch and landing operations that will also be required by other areas are listed in Table II. Also included is a cross reference to the phase of the turnaround scenario that requires its use. Some of the items are required infrequently during the scenario. These items, designated in the table by "Stays Connected," will be transported and connected at initial use. In order to save time and resources, they will be left connected until their final use before being removed from the pad area. However, they could be removed and used elsewhere should the need arise.

TABLE II - HARDWARE SHARED WITH OTHER BASE OPERATIONS

	Landing	Mate SSE I/F	Crew Egress	Cargo Removal	Install Blanket	Auto Maint Dump	Maint & Test	Remove Blanket	Cargo Integr	Internal Closeout	C/D & Launch
Unpressurized Rover		■		■	■		■	■	■		■
Pressurized Utility Vehicle	■		■				■			■	■
Mobile Crane				■					■		
Cargo Transport Vehicle				■					■		
Contamination Removal Sys.		■	■	■	■		■	■	■	■	
Cryogen Tanker(s)											■
Fuel Cell Maintenance Cart		■		Stays Connected			■	Stays Connected			■
ECLSS Maintenance Cart							■				
GN2 Handler				■			■				
Waste Mgmt. Sys. Deservicer		■		Stays Connected			■	Stays Connected			■
Surface Operations Lighting Sys.		■		■	■		■	■	■		■
Inspection Equipment						■					
Access Devices		■		■	■		■	■	■	■	■
LOX Cleaning Capability						■					
Communications Systems	■	■	■	■	■	■	■	■	■	■	■
Control & Monitor System	■	■	■	■	■	■	■	■	■	■	■
Auto. Performance Monitor	■	■	■	■		■	■		■	■	■
Mobile Teleoperated Assist.	■	■		■	■		■	■	■		
Remote Telerobotic Serv. Sys.				■	■		■	■	■		

Unpressurized Rover. Required to transport EVA personnel, tools, and equipment between the Habitat and the launch and landing pad.

Pressurized Utility Vehicle that will mate with the Habitat and the LCSV Crew Module. Required to effect IVA ingress/egress to the LCSV Crew Module for flight crew changeout and turnaround tasks.

Mobile Crane with Slings, etc. Required to remove down-cargo from or install up-cargo onto the LCL or LCSV. Also required to remove and replace major flight vehicle assemblies.

Cargo Transport Vehicle. Required to transport heavy cargos, up to 40 metric tons, between the launch and landing pads and the use area.

Lunar Dust Contamination Removal System. Required to remove dust from items whose performance may be degraded by lunar dust, to clean system interface connections, and to avoid introduction of dust into pressurized areas.

Cryogen Tanker(s). Required to transport lunar LOX and/or LH2 from the in situ production plant to the flight vehicle. Dedicated tankers will be required for each propellant.

Fuel Cell Maintenance Cart. Required to service and deservice flight vehicle, surface vehicle, and surface facility fuel cell oxygen, hydrogen, and water tanks.

ECLSS Maintenance Cart. Required to service, deservice, circulate, and filter fluids used in the various flight vehicle, surface vehicle and surface facility Environmental Control and Life Support Systems.

GN2 Handler. Required to transport GN2 and recharge GN2 tanks on pressurized buildings and vehicles.

Waste Management System Deservicer. Required to deservice the several WMS holding tanks, and clean and sanitize the systems.

Surface Operations Lighting System. Required to provide auxillary lighting for surface operations.

Inspection Equipment. Includes a range of inspection equipment required to determine the status of hardware and work environments.

Access Devices. Required to provide internal and external access to flight vehicles, surface vehicles, and surface facilities for inspection, maintenance, and servicing.

LOX Cleaning Capability. Required to LOX clean and double bag piece parts and components after maintenance (e.g., after replacement of valve soft goods), and to clean LOX systems after installation but before placing in service.

Communications System. Required for the simultaneous communication of command, data, video, and full-duplex voice among all locations involved in launch and landing operations.

Surface Systems Control and Monitor System. Required for control and monitor of flight vehicles and surface systems, vehicles, and personnel involved in multiple flight vehicle turnaround operations. Must include execution, monitoring, and evaluation of application test and monitoring software. Requires data exchange with and display of data from other base systems and similar systems on Earth.

Automated Performance Monitoring System. Required to automatically monitor systems performance and alert base personnel of out-of-tolerance conditions; thereby, relieving base personnel of this large and tedious task.

Mobile Teleoperated Assistant. Required to assist EVA personnel by performing such functions as holding, lifting, fetching, and stowing. Can also perform robotically simple tasks such as equipment, and launch and landing pad inspections to reduce crew size and EVA exposure.

Remote Telerobotic Servicer System. Required to perform robotically, tasks such as loading, unloading, positioning and other servicing operations, to reduce crew size and reduce EVA exposure. A derivative of the Space Station Freedom flight telerobotic servicer, modified for operation in the lunar surface environment, is envisioned. Could be a part of the Pressurized Utility Vehicle or a separate self-propelled vehicle.

SPECIAL ASSESSMENT AGENT (SAA)

2.1.1

Final Report

AV-FR-89/7011
AV Project Number: 70060

Final Report

MARS SOLAR ROVER

FEASIBILITY STUDY

Prepared for

NASA Lewis Research Center
21000 Brookpark Road
Cleveland, Ohio 44135

October 1989

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by

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ABSTRACT

The possibility of a solar powered rover to be used for the unmanned exploration of Mars is examined and analyzed. The amount of solar power available on Mars is found as a function of time, location and atmospheric dust, allowing the determination of the amount of power that can be collected by either photovoltaic or thermal collectors. It is found that the maximum available power in a normal Martian day varies from 220 to 140 watts per square meter depending on the amount of dust in the atmosphere, while the average annual power is 100 watts per square meter.

The energy requirements of the rover are determined and used to find the size of solar collector needed to power the vehicle. For the existing baseline vehicle, for which radioisotope thermoelectric generators are planned, is is expected that a 500 watt power supply will be required. The actual average power need of the baseline rover is 275 watts, and this is taken as the power supply requirements of a solar rover. About 66 percent of the energy consumed by the rover occurs when the rover is in an idle mode. For a solar powered vehicle which can be placed in a low power use dormant mode as needed (during dust storms, night, or in idle mode) 116 watts will be adequate. Power saving for a solar rover will be accomplished mainly by increasing the efficiency of the mechanical systems, and by significantly reducing the power demands of the computer systems from 75 watts to 25 watts by closing down parts of the computer system when they are not needed. For the baseline rover, energy storage is achieved by 72 kilograms of lithium titanium disulfide batteries with a capacity of about 7.2 kWh.

It is found that a solar powered Mars rover is possible, with a collector area of 16 to 23 square meters. The actual projected planform area of the existing vehicle is about 20 square meters. At any given time of year, the rover could operate successfully on over half of Mars. Near the equator, this solar rover could operate year-round. The introduction of a power saving "sleep" mode is found to be an effective method of reducing the collector area to about 7 square meters, as well as providing for improved rover survivability in the case of prolonged dust storms or failure of part of the solar panel or battery bank. Future work should include improved models for energy production and use, and design of rovers optimized for solar power.

TABLE OF CONTENTS

LIST OF FIGURES

Section 1
INTRODUCTION

The missions currently planned to place a unmanned rover on the surface of Mars all use radioisotope thermoelectric generators (RTGs) for the source of electric power. The only other source of power on Mars suitable to long term missions is solar power. In the past, solar power was not considered a useful source due to the distance from Mars to the Sun, the dust in the Martian atmosphere, and the day/night cycle.

Recently, several problems have surfaced that make the future use of RTGs difficult. The plutonium used to fuel them is in short supply, and the reactors that make this fuel are not operating. Making these reactors operational will require a large amount on capital, more than NASA may be willing to spend.

There are also safety issues associated with the use of plutonium. Although it is not very radioactive, plutonium is one of the most poisonous substances known. Containment of the plutonium in the event of a launch accident is mandatory. The current containment methods are believed adequate even for the case of the worst possible accident, and in fact some RTGs have survived launch failures intact. Because of this, the safety issue may be more of a perceived problem than a real one. However, even perceived problems can slow or stop a space mission, so this issue must be addressed.

Energy sources other than solar and nuclear do not appear to be useful on Mars. Stored chemical energy could not supply sufficient energy for the duration anticipated for the Mars rover sample return (MRSR) mission. The needed energy must be collected from the environment. Possible sources include temperature differences, geothermal heat, wind, and solar energy. Temperature differences, whether between different times of day, or different positions (for example between the air and the ground) are hard to use because they require large heat exchangers and a heat engine. Geothermal energy cannot be used by a rover, although it could one day prove useful for large fixed installations. Wind energy is probably insufficient due to the low air density on Mars. In addition, wind energy tends to be highly variable in both time and position, making it a difficult resource for a rover to rely on. Thus, direct solar power appears to be the only viable alternative to RTGs for a Mars rover. This report gives the results of an initial investigation into the feasibility of a Mars solar rover.

The first subject that must be addressed is the availability of solar energy on Mars. Factors that affect the amount of solar radiation received on a collector on the surface of Mars are the eccentricity of Mars' orbit, the Martian seasons, the time of day, the panel orientation, and the

amount of dust in the atmosphere. Of these, the effects of the orbit, season, time, and panel orientation can be calculated. The effects of dust can be based on Viking lander data and theoretical models.

Once the resource has been quantified, the amount of energy that can be converted to a useful form can be determined. Two methods of conversion are investigated: photovoltaic conversion for electrical energy production, and thermal collectors to gather heat for the thermal control system. For the photovoltaics, both silicon and aluminum-gallium-arsenide gallium-arsenide heterojunction (abbreviated GaAs) types of cells are considered, along with two collector orientations, horizontal and tracking. For the thermal collectors, only simple flat-plate collectors are considered.

Sizing the solar panel requires quantification of the power needs of the rover. Power is needed for mobility, computation, data storage, science, communications, vehicle control, and thermal control. Each of these systems requires a varying amount of power depending on the operational mode of the rover. An average power requirement can be found by examining a possible operating scenario that defines the baseline case. When this is done, the average power use is found to be only about 10 percent more than the power used by the rover when it is in idle mode. The rover spends 72 percent of the time in idle mode. Because of this, the possibility of a rover with reduced energy needs is examined, where the reduction is to be achieved by reducing the power needs of the idle mode.

Use of solar power requires energy storage. For the electrical energy, batteries are used. The driver for the battery store size is the need to survive the night. The batteries also influence the size of the solar panel, as a portion of the energy stored and later retrieved from them is lost. For the thermal energy store, a phase-change material is assumed. Due to the temperatures involved, water is an acceptable material.

With the output of the various types of collectors determined, and the energy needs of the rover specified, the size of the required collector can be found. This is done for three cases: a panel that can provide an average of 500 watts, the same power as the RTG of the current MRSR rover; a combined electric and thermal collection panel to handle the baseline case; and a combined panel to handle the case with the reduced power idle mode.

The conclusions of this program are as follows:

- The available solar power on a horizontal panel averages over 100 watts per square meter for most of Mars for most of the year.

- The amount of power that can be collected averages 17 to 20 watts per square meter for photovoltaics, and 60 watts per square meter for thermal collectors, for areas of Mars with a 100 watt per square meter resource.

- The power needs of the baseline rover average 256 watts electrical and 50 watts thermal, including the storage losses. If the idle mode, which currently consumes 240 watts, is replaced by a sleep mode, which consumes 80 watts (and in both cases 50 of those watts are thermal), then the average power requirements drop to 116 watts electrical and 50 watts thermal.

- The required panel size is 25 square meters, if an average power output of 500 watts is required, corresponding to the current design using an RTG. For the baseline rover using combined electric and thermal collection, the panel size is 13.63 square meters. For the case with the sleep mode, the panel size is 6.63 square meters.

The recommendations from this study are the following:

- Improved models for the solar resource on Mars, as well as the amount of the resource that can be converted to useful forms of energy, should be developed.

- Methods for reducing the power needs of the rover should be investigated. These should include both component efficiency improvements and energy management improvements, such as introduction of a sleep mode.

- Rover configurations more conducive to solar power should be developed. Such configurations would have large areas suitable for mounting solar collectors with little or no need for deployable structures and would make allowances for camera and antenna placement so that the solar collectors would be shadowed as little as possible.

- Experimental data are needed on the long term effects of the Martian environment on the efficiency of solar collectors on Mars. This may require that a small, simple probe be sent to Mars for this purpose. A somewhat more complex probe could also be sent, for example a small rover. Such a rover could be used to test the Martian surface in order to determine whether there are any problems with mobility, before a larger, more expensive rover is sent.

Section 2
SOLAR POWER PRODUCTION

2.1 Solar Availability

To examine the feasibility of a photovoltaic power system for an unmanned Mars rover, the solar radiation levels on the Mars surface must first be determined. The total radiation reaching a Martian surface is the sum of the direct solar radiation and a diffuse component resulting from scattering in the atmosphere and reflection from surrounding surfaces. Estimates for these quantities are based on information provided by Appelbaum (1989).

2.2 Direct Solar Radiation

The following equations are used to estimate the direct solar radiation on a horizontal surface as a function of season, latitude, time of day and optical depth of the atmosphere. The direct radiation, I_b, is:

$$I_b = I_o \cos(\beta)\, e^{-(\tau/\cos(\beta))}$$

where the zenith angle β is given by:

$$\cos(\beta) = \sin(\phi)\sin(\delta) + \cos(\phi)\cos(\delta)\cos(h)$$

ϕ = latitude

δ = solar declination.

L_s is the Areocentric longitude defined as the position of Mars in its orbit measured from the Martian vernal equinox. Thus:

At $L_s = 270°$ (N. Hemisphere winter), $\delta = -24.8°$

At $L_s = 90°$ (N. Hemisphere summer), $\delta = 24.8°$

h = hour angle (0 at zenith; + to the west).

I_o is the solar radiation on a surface normal to the sun's rays beyond the Martian atmosphere and is given by:

$$I_o = 590(1 + ecc(\cos(L_s - 245)))^2/(1 - ecc^2)^2 \quad (W/m^2)$$

where the eccentricity, ecc = 0.093377.

Here, τ is the optical depth, a dimensionless quantity which determines the reduction in the direct radiation due to scattering in the atmosphere. A value for τ of 0.5 has been assumed for clear sky conditions, and 2.0 for dust storm conditions. Higher values for τ occur, but they are rare and do not last long.

Because the Martian orbit is elliptical, I_o varies from a maximum of 718 W/m^2 to a minimum of 493 W/m^2. The variation in I_o is shown in Figure 2-1. Aphelion occurs at $L_s = 69°$ and perihelion at $L_s = 249°$.

To begin the examination of the insolation levels, the diurnal variation at the equator was estimated for $L_s = 90°$ (summer in north hemisphere) and $L_s = 270°$ (winter in northern hemisphere) for an assumed optical depth of zero. These estimates are shown in Figure 2–2. The average for the daylight portion of a sol (one Martian day) at $L_s = 90°$ is 321 W/m^2, and for $L_s = 270°$, the daily average is 451 W/m^2. An optical depth of zero is not realistic, so in Figure 2-3 the estimates of the direct insolation for an optical depth of 0.5 are summarized. With this value for the optical depth, the attenuation in the direct insolation reduces the daylight average values to 155 and 217 W/m^2 for $L_s = 90°$ and $L_s = 270°$, respectively. That is, the direct radiation is reduced by approximately half by scattering in the atmosphere (assuming an optical depth of 0.5). The total insolation will not, however, be reduced by this amount, because there will be an increase in the diffuse component.

Estimates of solar insolation have also been made for a location with a latitude of 45° N. These results are shown in Figure 2-4. During the summer, the average daylight insolation is 195 W/m^2, which is comparable to values at the equator. However, during the winter, the daylight average falls dramatically to only 20 W/m^2, making operation of a solar-powered vehicle this far north possible only during the summer.

The above estimates have been calculated assuming the solar array is a horizontal surface. If the array tracks the sun, so that its surface normal is parallel to the incident solar rays, the insolation value can be increased. Results for a tracking array are shown in Figure 2-5. By tracking, the average daylight insolation can be increased from 217 to 301 W/m^2 for $L_s = 270°$ and from 155 to 214 W/m^2 for $L_s = 90°$. There is a penalty for tracking in that power must be used to provide the sensing and tracking and additional mass is required for

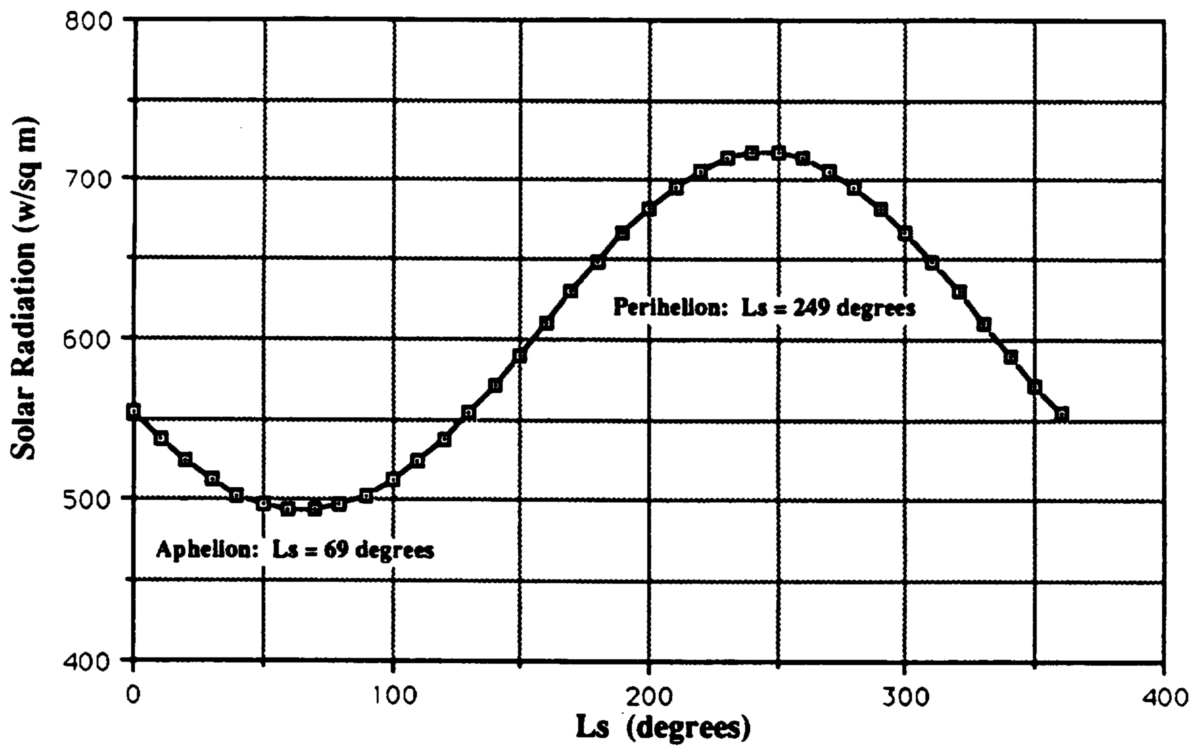

Figure 2-1. Solar radiation intensity at Mars, normal incidence, zero optical depth

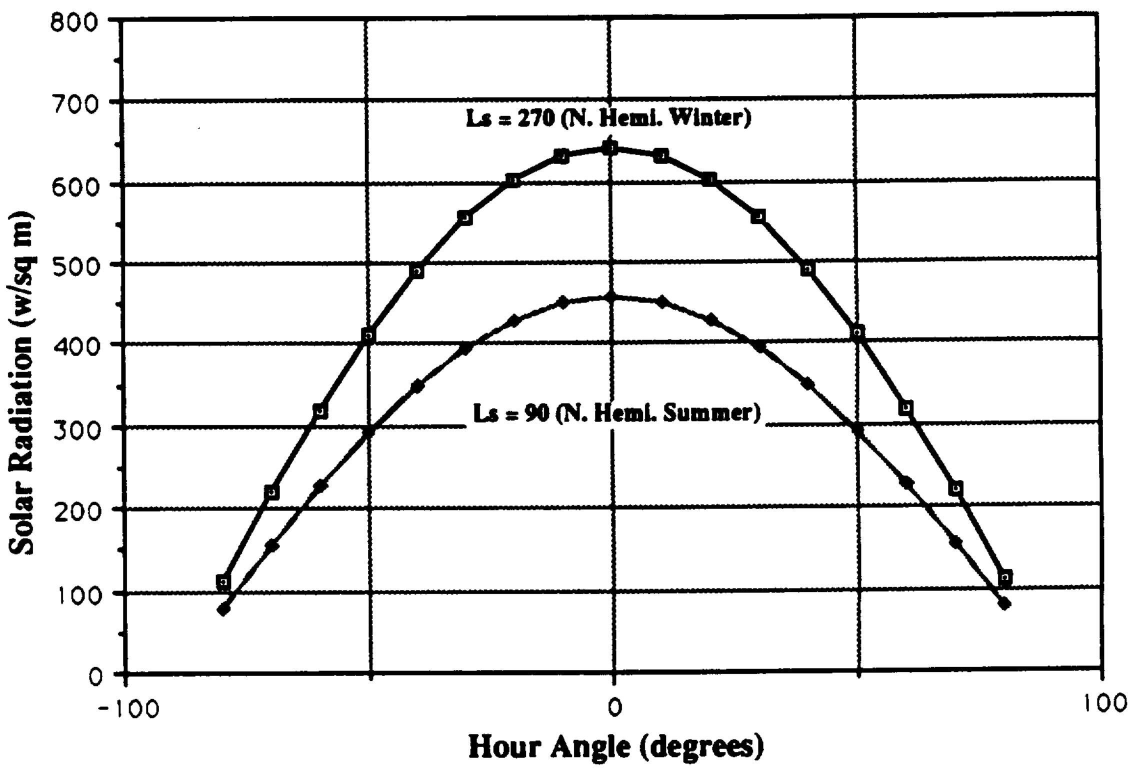

Figure 2-2. Diurnal variation of direct radiation at equator (horizontal surface), optical depth = 0.

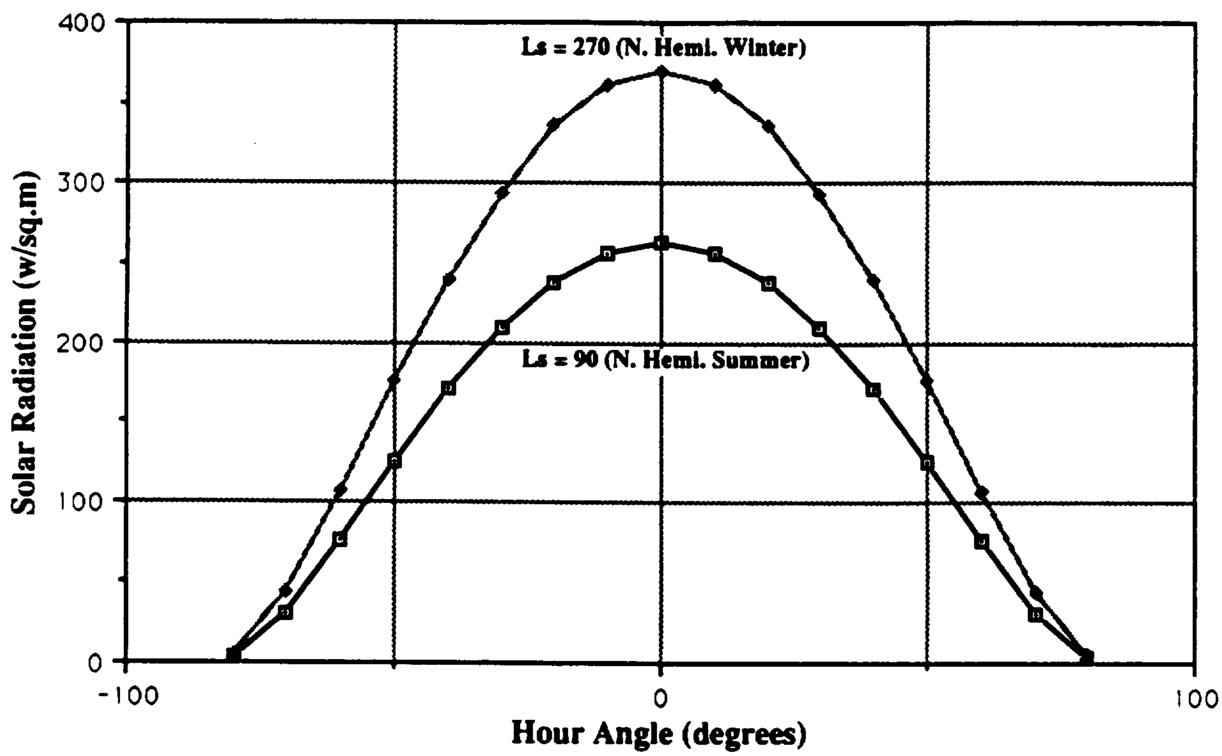

Figure 2-3. Diurnal variation of direct radiation at equator (horizontal surface), optical depth = 0.5.

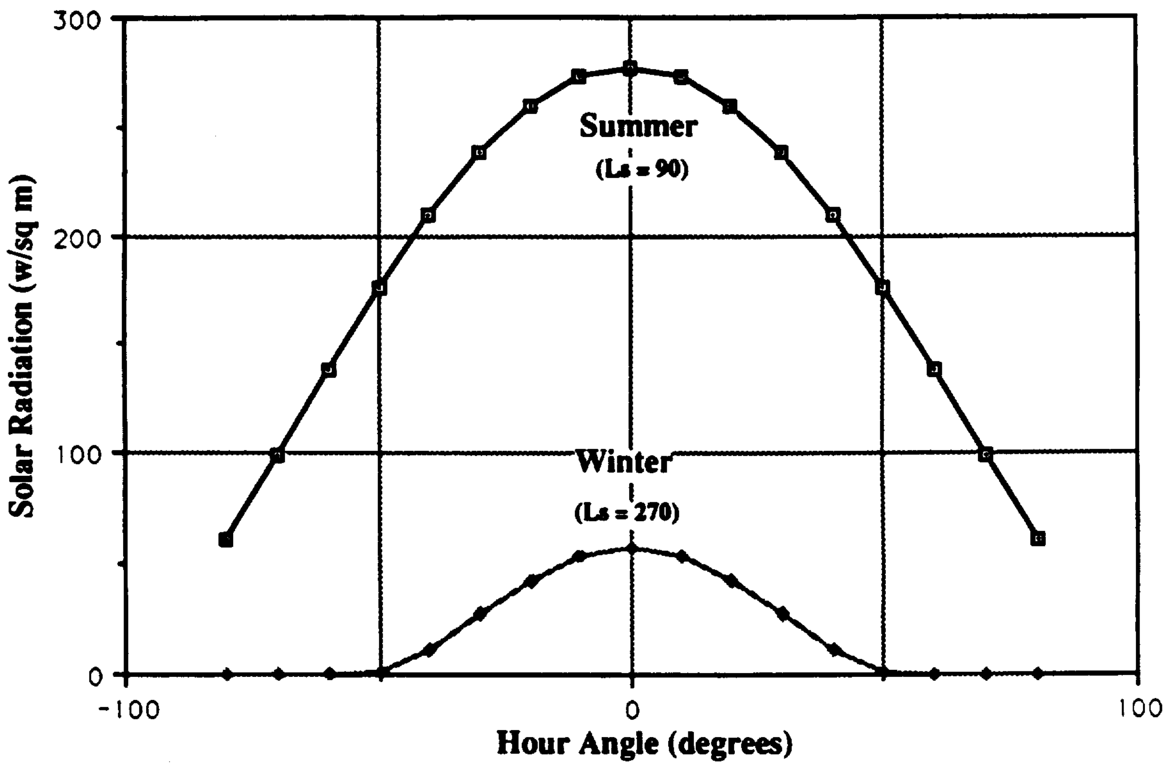

Figure 2-4. Diurnal variation of direct radiation at 45 degrees north (horizontal surface), optical depth = 0.5

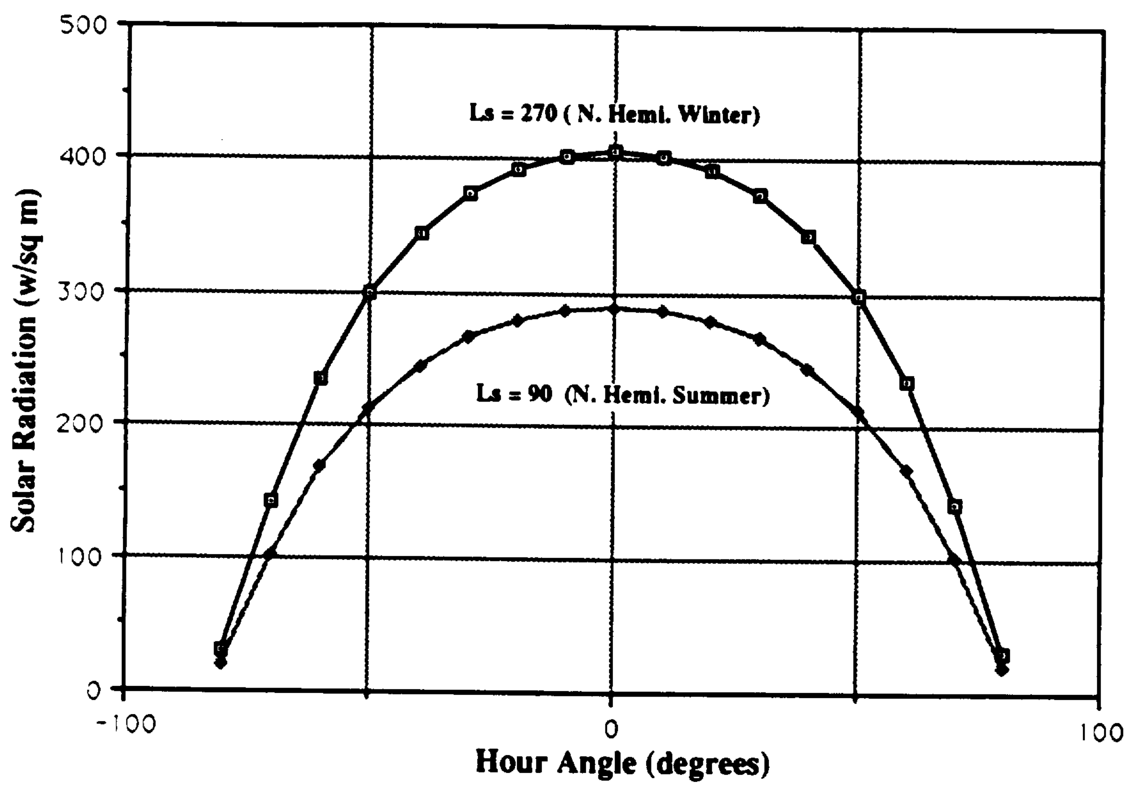

Figure 2-5. Diurnal variation of direct radiation at equator (tracking surface), optical depth=0.5.

motors, sensors and linkages. This additional power requirement can be estimated by referring to the power required to track and move the communications antenna, as shown in the Phase 1 design data book (Muirhead, 1988). To move a 1-kg antenna dish requires 15 watts. So if, for example, if a 10 m^2 panel at 2 kg/m^2 is required to provide the necessary power, then 300 watts is necessary for tracking. This is not, of course, a continuous requirement, but would made at discrete time intervals.

2.3 Diffuse Radiation

In the previous section, it was shown that the atmospheric dust reduces the direct radiation by 40 percent when the optical depth is 0.5, which is considered clear conditions. In dusty conditions when the optical depth is 2.0, the direct radiation can be reduced to 14 percent of the level in space. Fortunately for the use of solar power on Mars, much of the loss in direct radiation is still available as scattered radiation. Appelbaum gives predictions for the total radiation, the sum of the direct and scattered components, for a wide range of optical depths and solar elevation angles. Using these, the total available solar intensity on Mars can be found.

The optical depths that occur most of the time can be determined from the Viking lander data. A summary of this data is shown in Figure 2-6 for Lander number one and in Figure 2-7 for Lander number 2. For a large portion of the year the optical depth is about 0.5. In the later part of the year the global dust storms occur, raising the optical depth. For lander number one, the dust storm causes a peak optical depth of about three. For the other lander, the maximum is about two. For both landers, the optical depth is rarely greater than two. The higher optical depths tend to only occur when Mars is near perihelion, so the reduced insolation due to dust is compensated for by being closer to the sun. For design purposes, a optical depth range of 0.5 to 2.0 was selected as typical operating conditions. When higher optical depths occur, rover operations will have to be modified to accommodate the reduced power production.

Figure 2-8 shows G_h, the available solar power on a horizontal plate for several solar zenith angles as a function of atmospheric optical depth. The region of the plot that covers those optical depths that occur most of the time is $\tau = 0.5$ to 2.0. For this plot, the in-space intensity of the solar energy is 590 watts per square meter, the average value for Mars over the course of a orbit. As shown in the plot, the effect of dust on the total radiation level is not great, about 10 percent loss for a optical depth of 0.5, to 40 percent loss for an optical depth of 2.0.

Integration over one Martian day, a sol, gives the solar energy available. The results of this

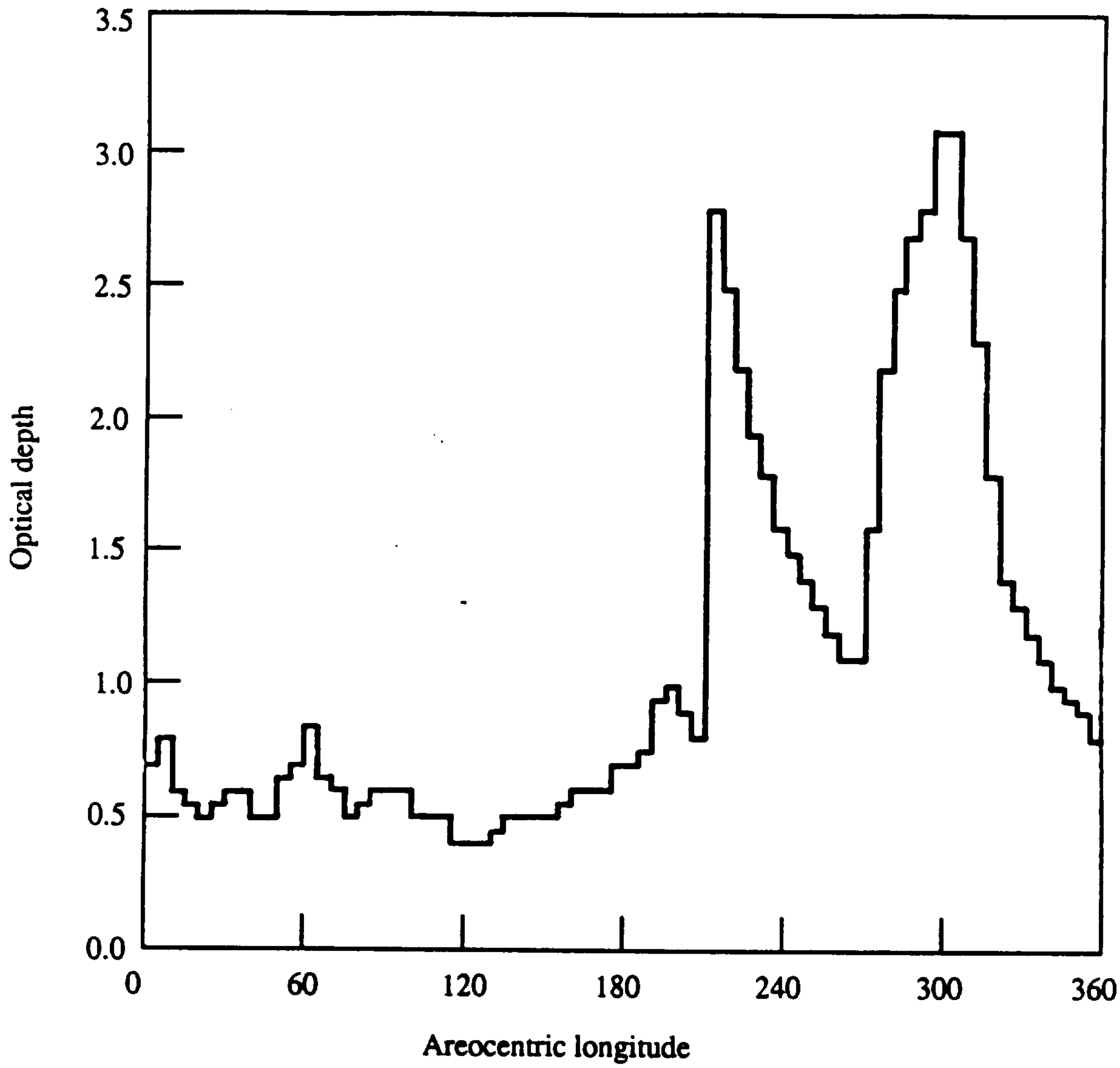

Figure 2-6. Optical depth history at Viking lander one.

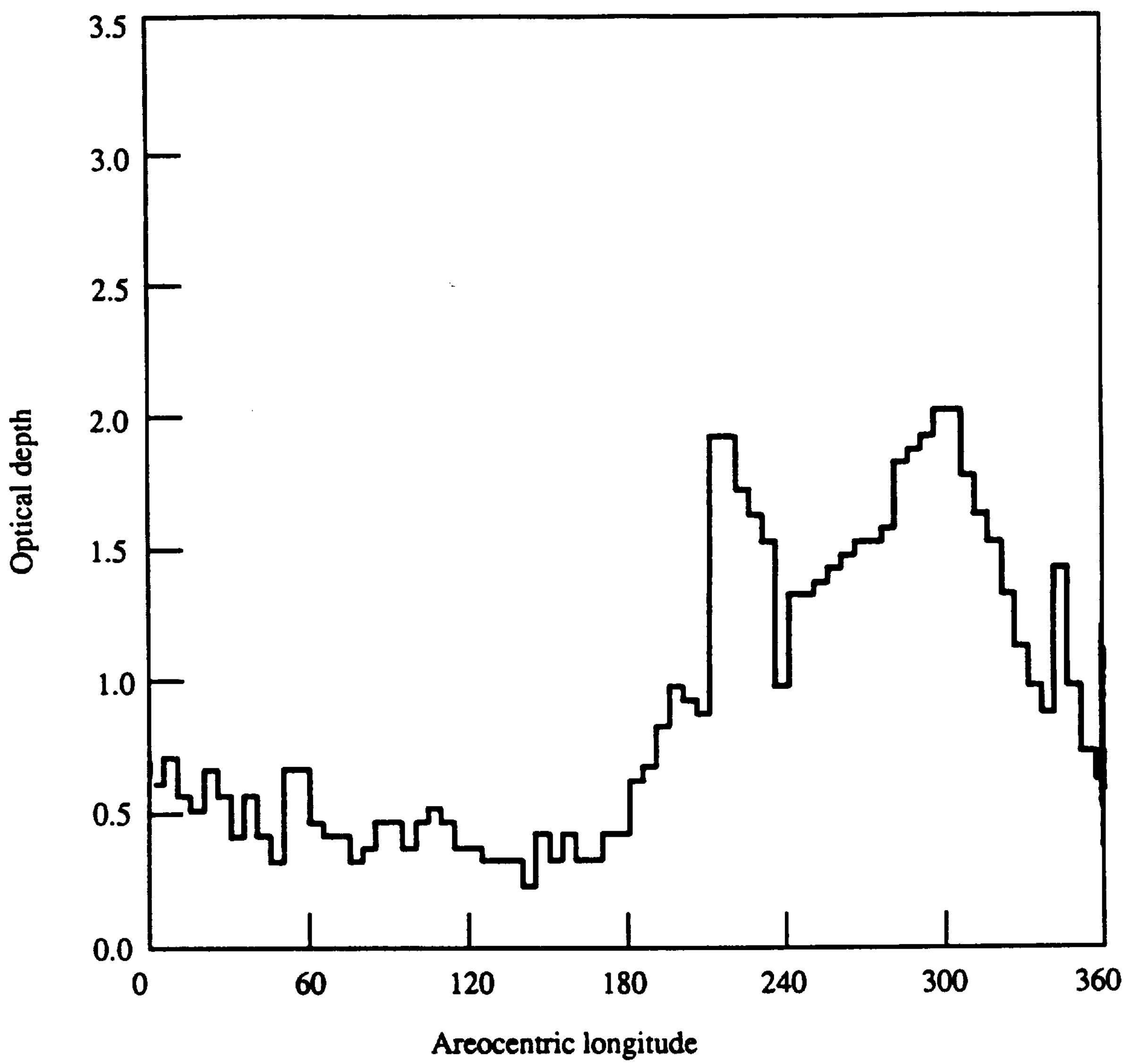

Figure 2-7. Optical depth history at Viking lander two.

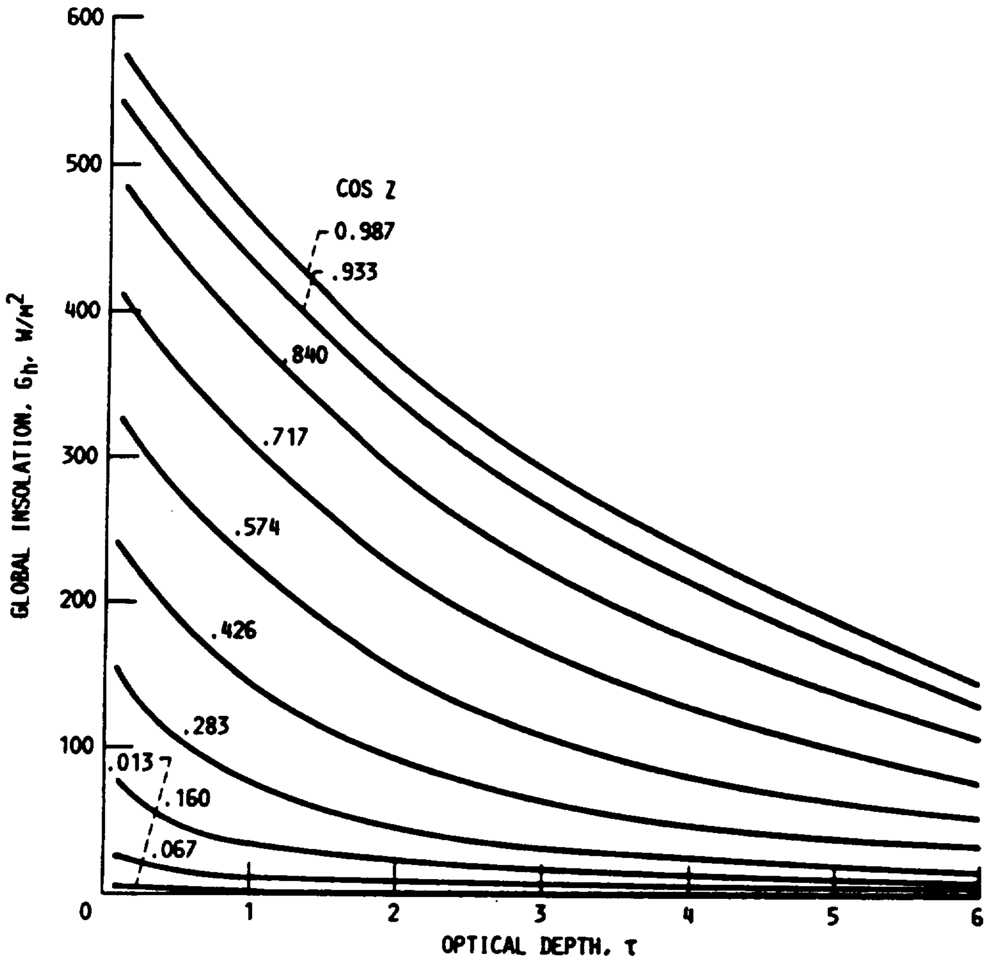

Figure 2-8. Variation of total insolation (direct plus scattered) on a horizontal surface with optical depth.

a-4

integration are shown in Figure 2-9 for the case of an optical depth of 0.5, and in Figure 2-10 for the case of an optical depth of 2.0. In both figures, the available energy averaged over a entire sol is shown for several latitudes and areocentric longitudes.

These figures indicate that 100 watts per square meter are available in most locations and in most seasons on Mars. The maximum availability occurs in the southern summer, at over 220 watts per square meter. At an optical depth of 2.0 due to a moderately severe dust storm, the availability is reduced to about 66 percent of the clear condition.

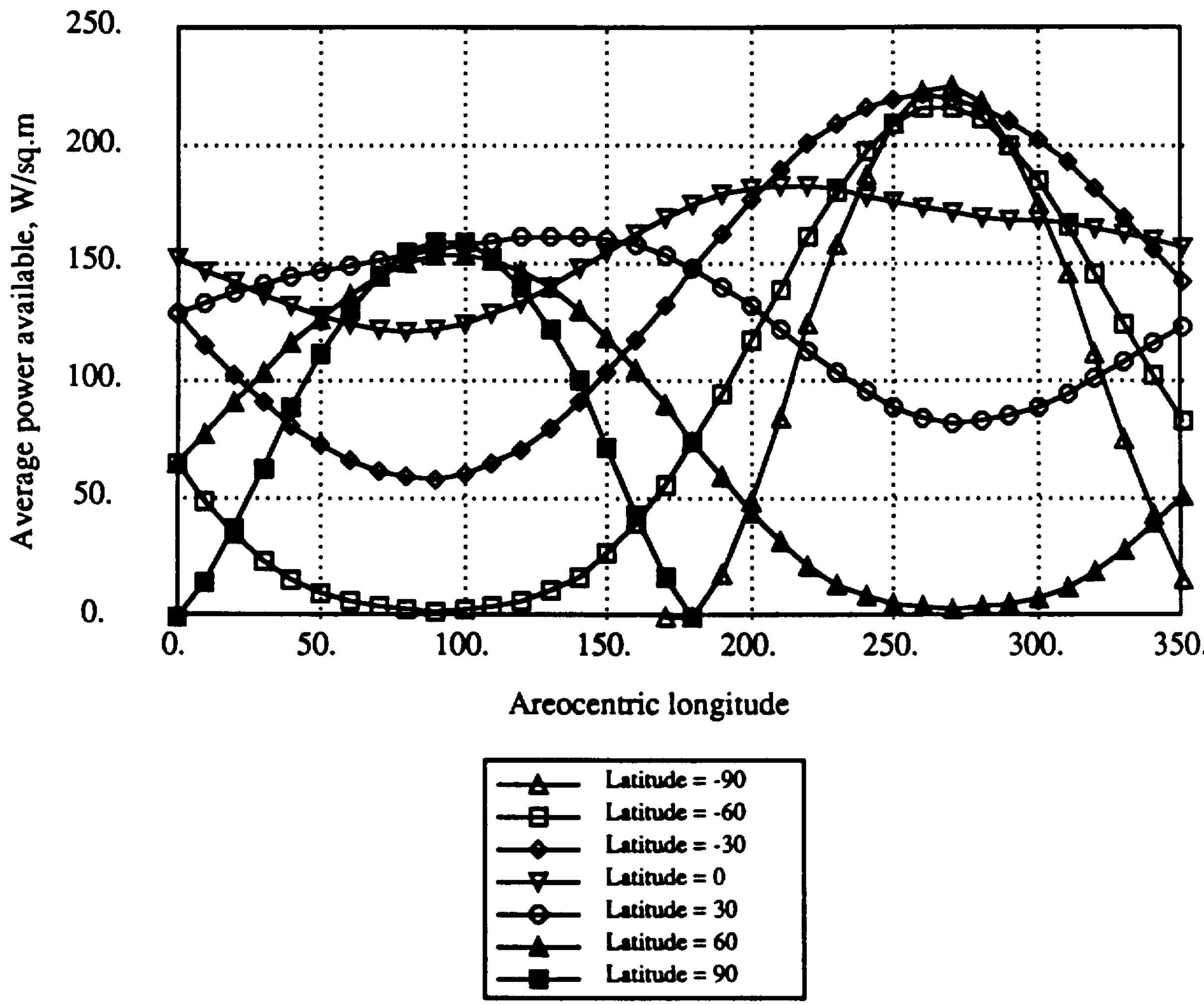

Figure 2-9. Solar energy available for a horizontal surface on Mars, optical depth = 0.5

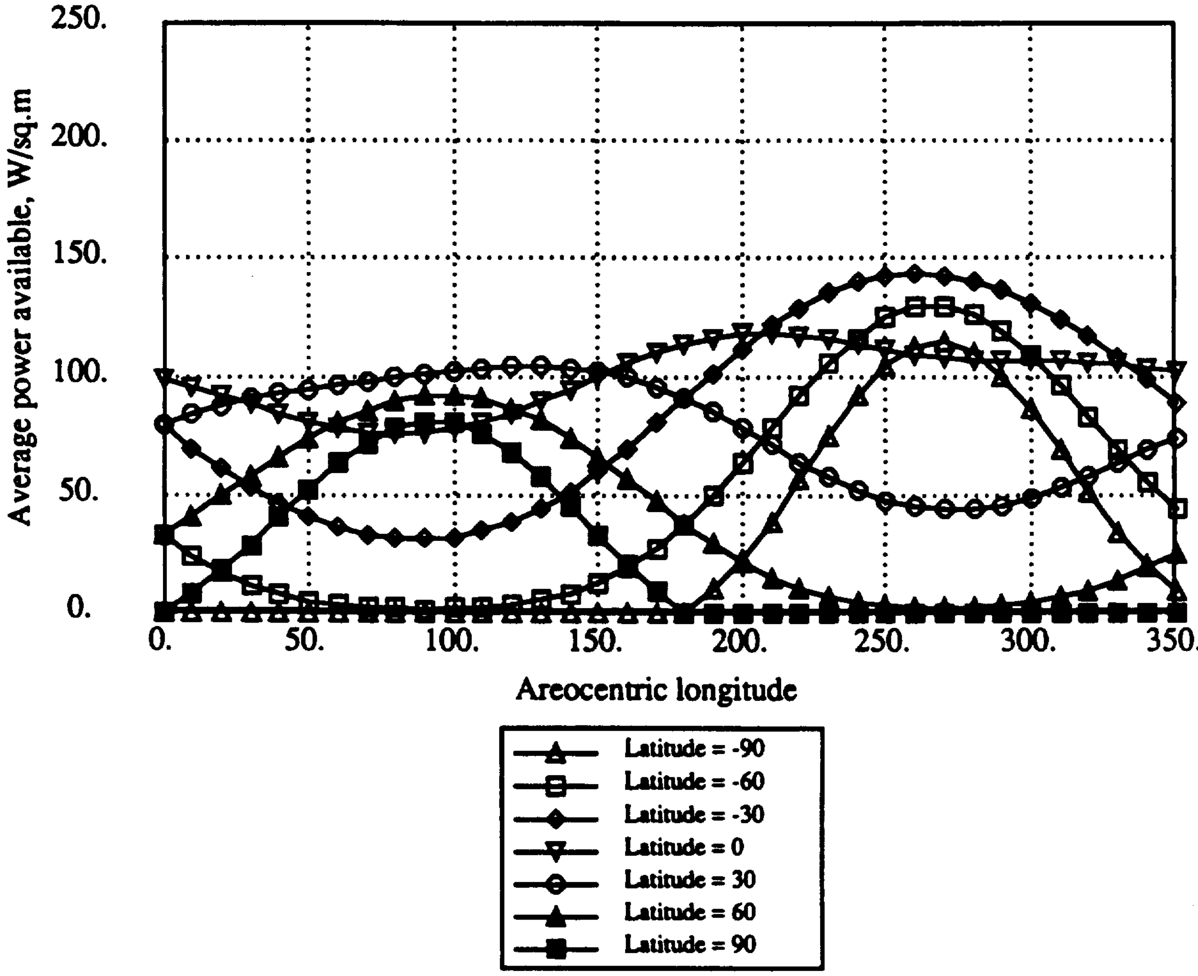

Figure 2-10. Solar energy available for a horizontal surface on Mars, optical depth = 2.0

Section 3
SOLAR ELECTRIC ENERGY PRODUCTION

3.1 Model Description

In this study, the level of analysis used for modeling the production of electrical power from solar cells is kept fairly simple. The effects taken into account are the solar intensity, both direct and scattered, the cell material, and the cell temperature. The effects not taken into account are the radiation scattered off the ground, the change in the panel's efficiency due to radiation level, except as it affects temperature, and the second order effects of heating and cooling on cell temperature. In determining the cell temperature, only the effects of the incoming solar radiation and black body radiation from the front of the panel are included. The effects of cooling from the Martian atmosphere, heating from scattered and emitted radiation of the ground, and infrared radiation emitted from atmospheric dust were ignored.

3.2 Cell Temperature and Efficiency

The solar panel has three main paths for gaining and losing energy: incoming solar radiation, electrical energy production, and black body radiation from its front surface. Other paths also exist. Thermal exchange with the ground is possible; however, the ground can be expected to have a temperature close to that of the panel, as both are exposed to the same sun, and the back of the cells is somewhat insulated by their supporting structure. Thus, the effects of the ground will be ignored. The Martian atmosphere can be expected to cool the panel to some extent. On Earth, a solar panel will lose about half its heat to the air by convective cooling, the rest by black body radiation. On Mars, with its lower atmospheric density, convective cooling can be expected to be a small effect, so it will be ignored here. Atmospheric dust will also radiate black body radiation to the cells. However, the dust is cold and fine, resulting in low levels of infrared radiation. With these simplifying assumptions, the only terms that remain are the solar input, electrical production, and the cell's black body radiation. The temperature of the cell can now be found.

The incoming energy to the cell that is not converted into electricity, and will be reradiated as black body radiation is

$$\text{energy input} = I\,(1 - \varepsilon)$$

where I is the incoming solar radiation and ε is the efficiency of the solar cells. The efficiency is modeled as a simple function of temperature:

$$\varepsilon = \varepsilon_0 (1 - \alpha T)$$

where ε_0 is the extrapolated efficiency of the cell at absolute zero, α is the reduction in cell output per degree, and T is the absolute temperature. The values used for the constants ε_0 and α for typical GaAs (Flood, 1989) and silicon cells (Sturtevant, 1989) are given in Table 3-1.

Table 3-1

Cell type	Efficiency at 25 C	ε_0	α
GaAs	21.5%	0.3095	1.0243×10^{-3}
Silicon	15.0%	0.350	1.917×10^{-3}

The energy lost in the form of black body radiation is

$$\text{Energy lost} = 5.67 \times 10^{-8}\, T^4\ \text{W/m}^2.$$

The equations for energy input and energy lost can be equated and solved for temperature. This assumes that the solar cells are in a state of thermal equilibrium at all times, i.e., their thermal mass is low so there is no appreciable lag in temperature when they are warming up or cooling down. This procedure also assumes that the cell's absorptance and emittance are about equal to one.

The resulting equation is

$$I (1 - \varepsilon_0 (1 - \alpha T)) = 5.67 \times 10^{-8}\, T^4$$

which can be solved for T. The temperature is then used to find the cell efficiency.

3.3 Solar Input

The solar input to the cells was calculated for two panel geometries: a horizontal panel and a tracking panel. In the case of the tracking panel, the added solar input due to scattered light from the ground was ignored. The effects of the eccentricity of the Martian orbit, the latitude of the

panel, and the atmospheric dust were taken into account as described in Section 2. Calculations were carried out in Section 2 for optical depths of 0.5 (clear condition) and 2.0 (dusty conditions).

3.4 Energy Calculations

Given the above equations, the energy production of a solar panel can be found. For a particular season and time of day, the incoming solar energy can be found from the equations in Section 2.2. The effects of the atmospheric dust on the incoming radiation can be found from Figure 2-6. The equations in Section 3.2 can then be used to find the cell temperature, and hence the efficiency and power output.

The average energy production of a solar panel was found using the above method. The panel output was averaged over an entire Martian day, one sol. Calculations were carried out for seven latitudes on Mars (-90, -60, -30, 0, 30, 60, and 90 degrees), the complete range of areocentric longitudes, the two types of cells (GaAs and silicon), two optical depths (0.5 and 2.0), and two panel geometries (horizontal and tracking). The results are shown in Figures 3-1 to 3-8. If the average power for the daylight portion of the day is desired, then the Figure 3-1 to 3-8 results can be divided by the fraction of the sol that has daylight, given in Figure 3-9.

The effects of dust storms can be seen by comparing the results for the low and high optical depth cases. To a first approximation, the increased optical depth reduces the energy production by 30 to 40 percent. This reduction only applies for the latter half of the year, when Mars is near perihelion, and is compensated for by the distance reduction to the sun.

The results indicate that the advantage of the GaAs cells over the silicon cells is small, because the low temperatures at which the cells are operating is more advantageous to silicon than GaAs. An example time history of cell temperature is shown in Figure 3-10. Comparing this predicted temperature with measured air temperatures, it is found that the cell temperature is within 50 degrees of the air temperature.

The effect of using a tracking collector is to increase the energy production by about 50 percent over a non–tracking collector. This estimate is somewhat high, however. In making it, an assumption was made that all of the incoming light came from the direction of the sun. Due to the effects of the dust, this is not a correct assumption. If most of the scattering is forward scattering, then this assumption is close to correct, and if the scattering is more isotropic, then it is a poor one. Thus, these results for a tracking collector should be viewed as a upper limit. A further complication to this issue is that the both the scattering and the photovoltaic cell efficiency are

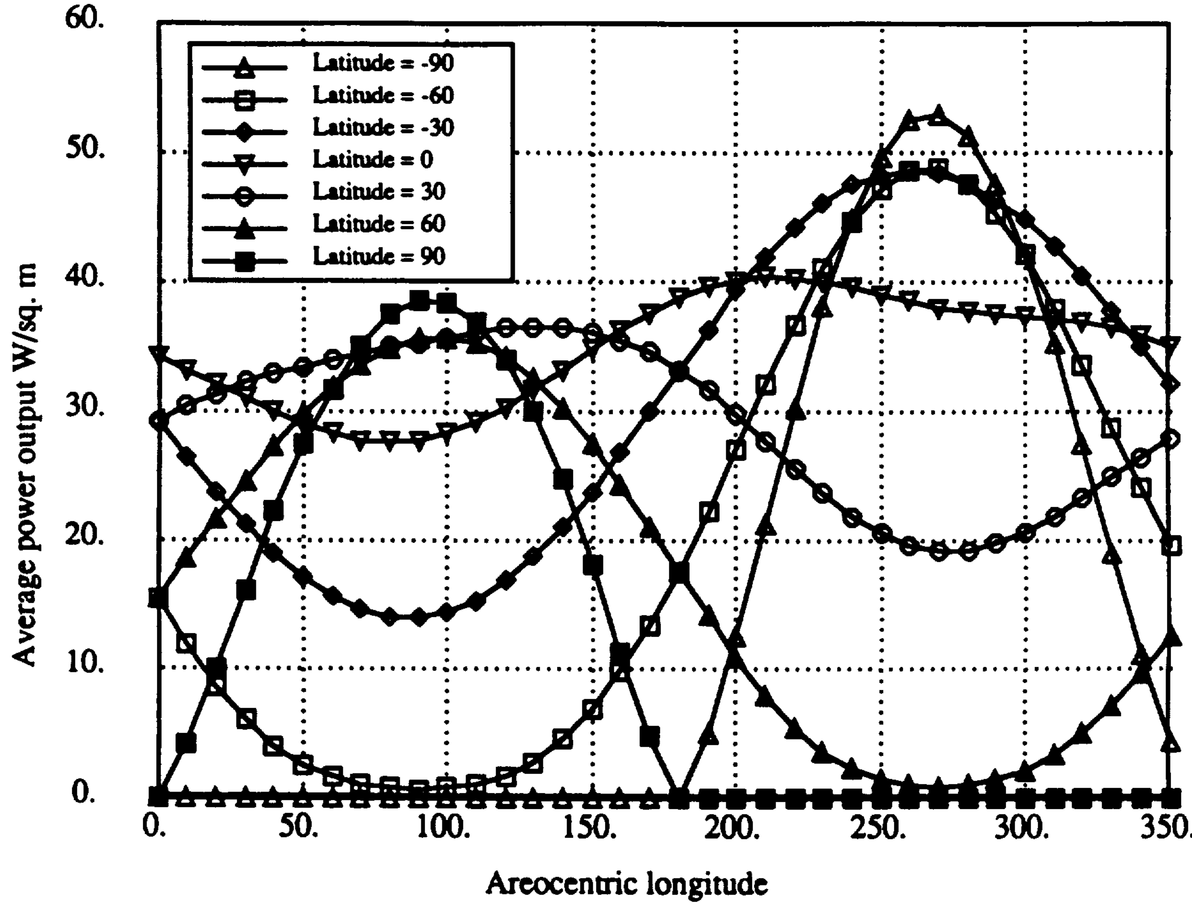

Figure 3-1. Solar panel average power output for a nontracking, horizontal Ga As panel with a optical depth of 0.5.

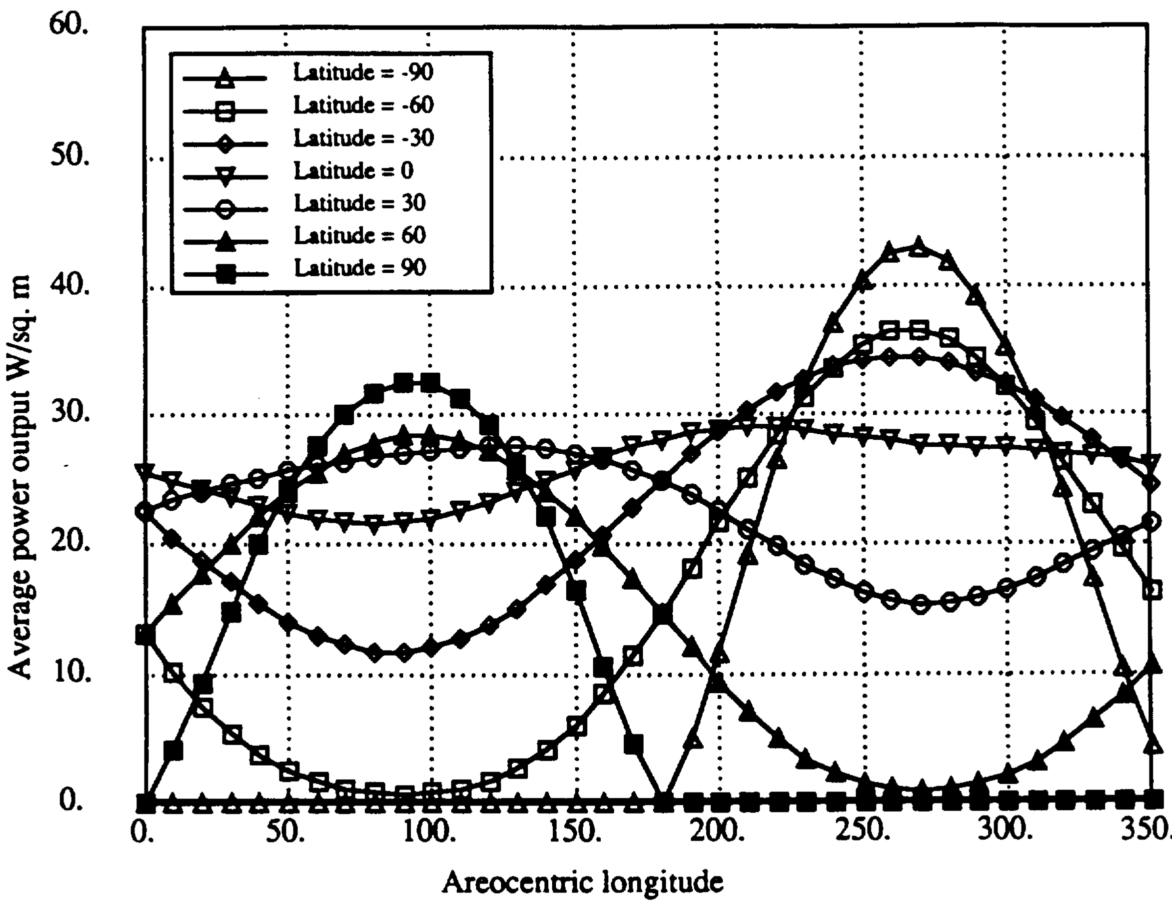

Figure 3-2. Solar panel average power output for a nontracking, horizontal Silicon panel with a optical depth of 0.5.

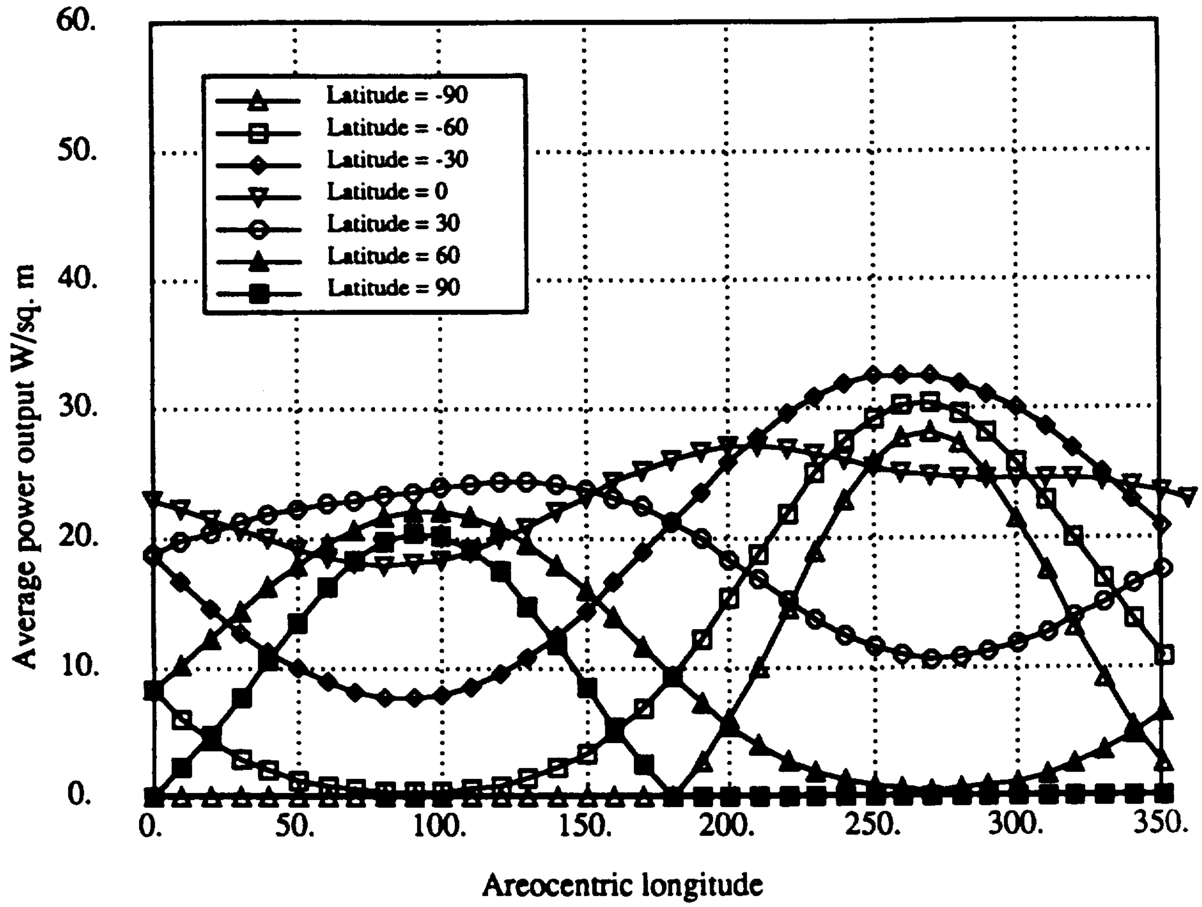

Figure 3-3. Solar panel average power output for a nontracking, horizontal Ga As panel with a optical depth of 2.0.

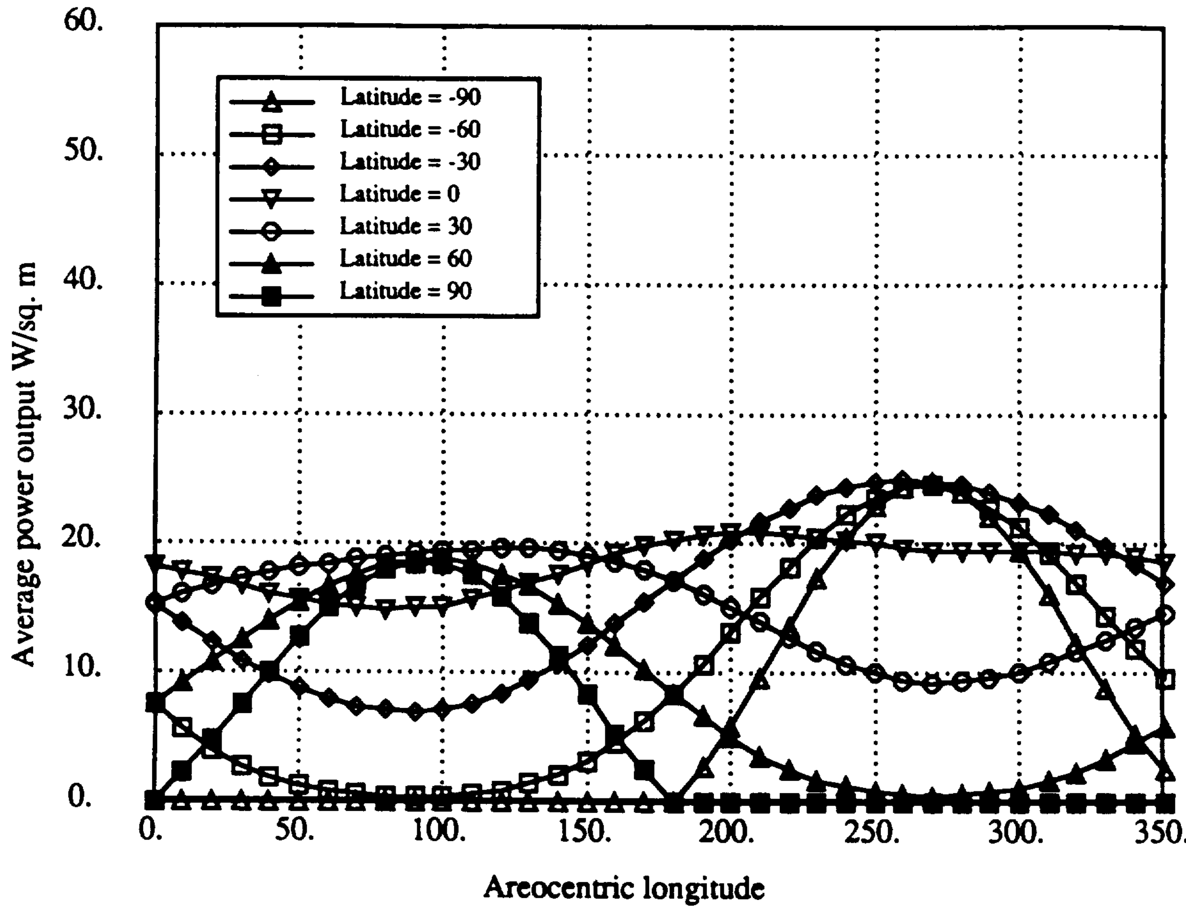

Figure 3-4. Solar panel average power output for a nontracking, horizontal Silicon panel with a optical depth of 2.0.

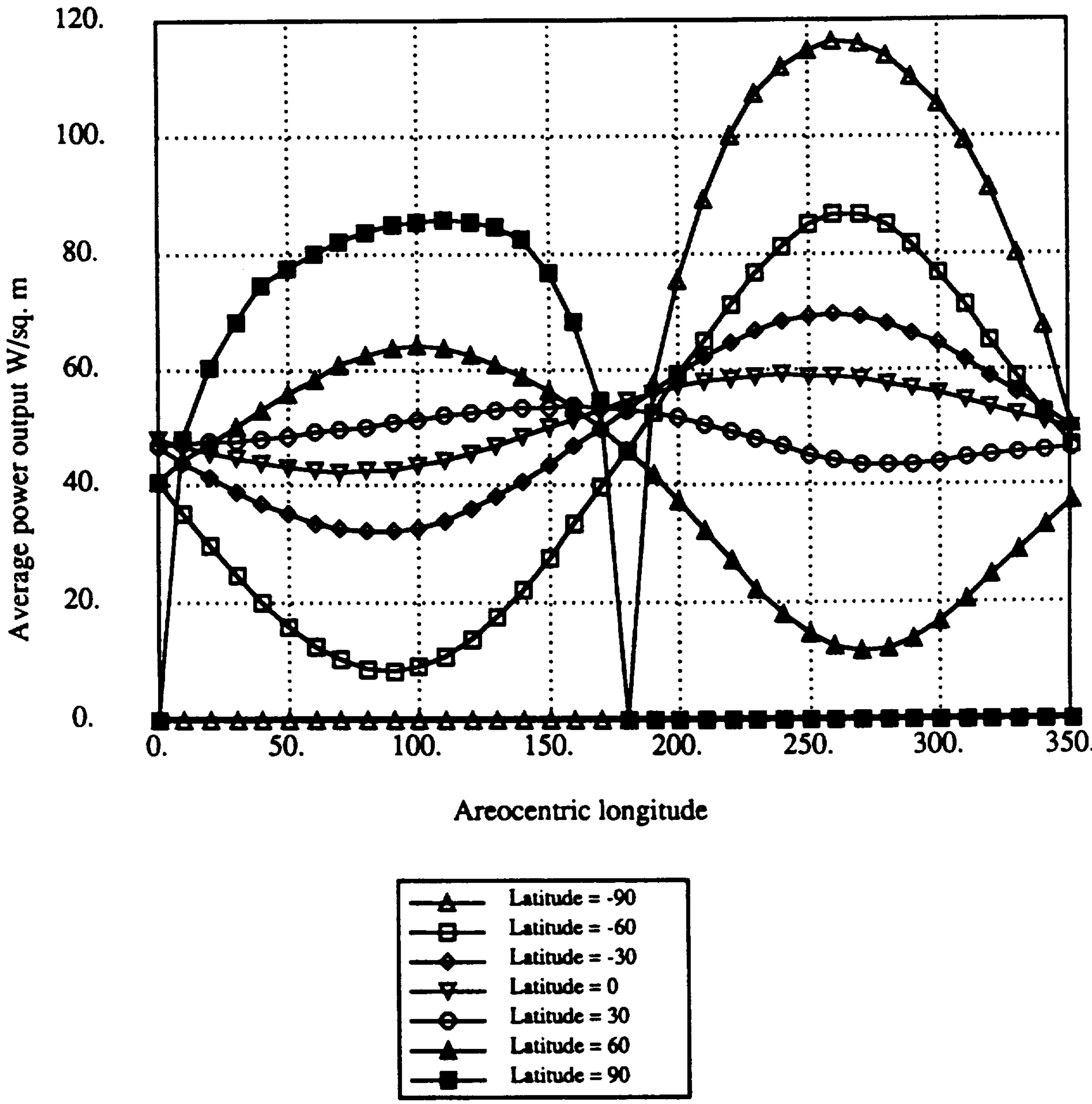

Figure 3-5. Solar panel average power output for a tracking, horizontal GaAs panel with a optical depth of 0.5.

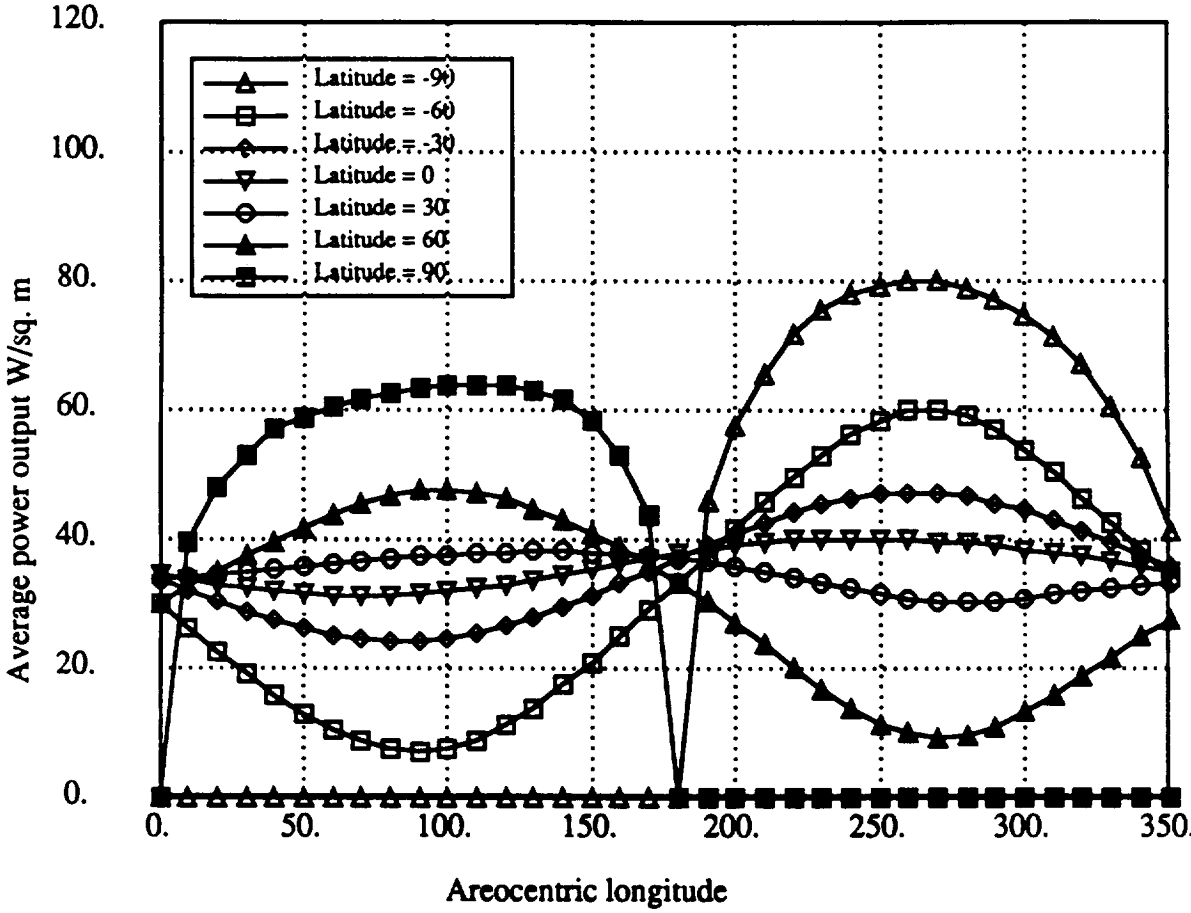

Figure 3-6. Solar panel average power output for a tracking Silicon panel with a optical depth of 0.5.

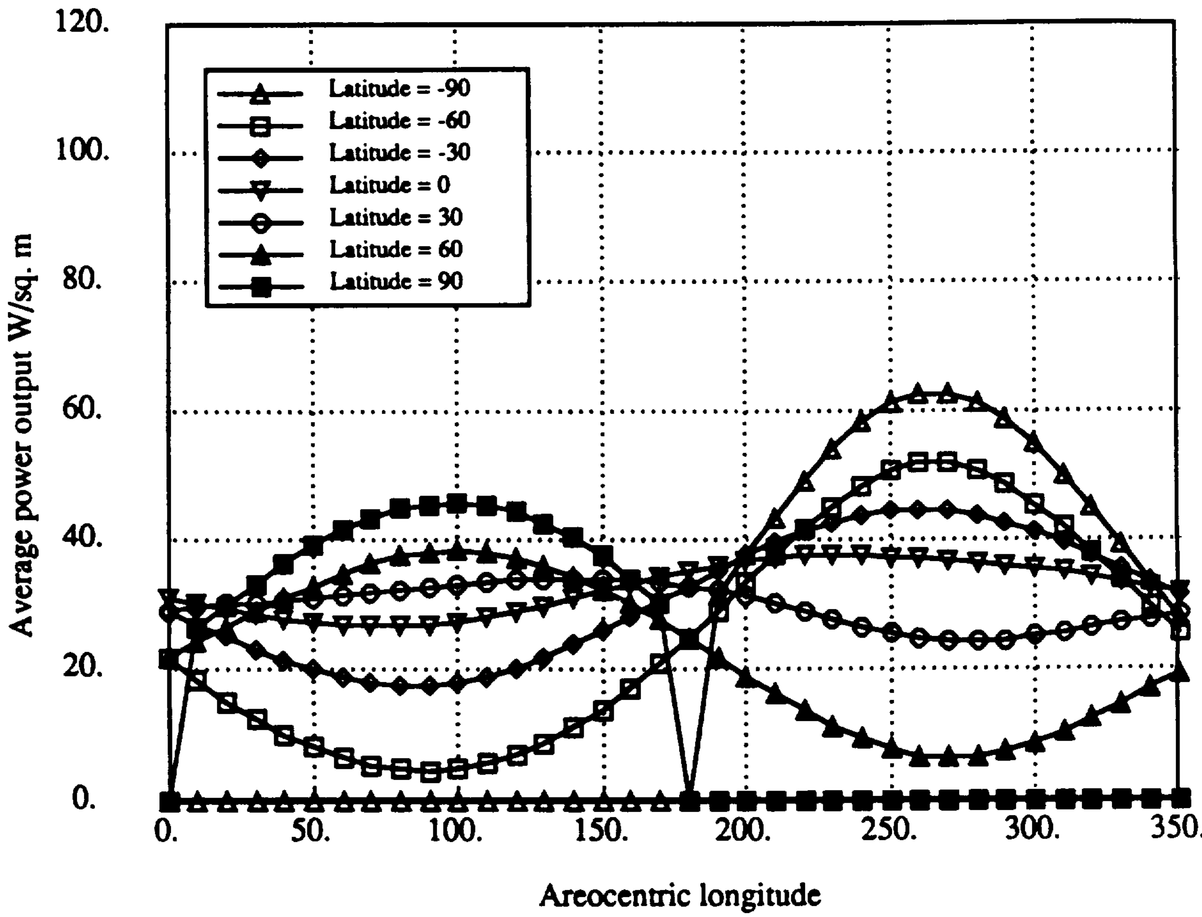

Figure 3-7. Solar panel average power output for a tracking GaAs panel with a optical depth of 2.0.

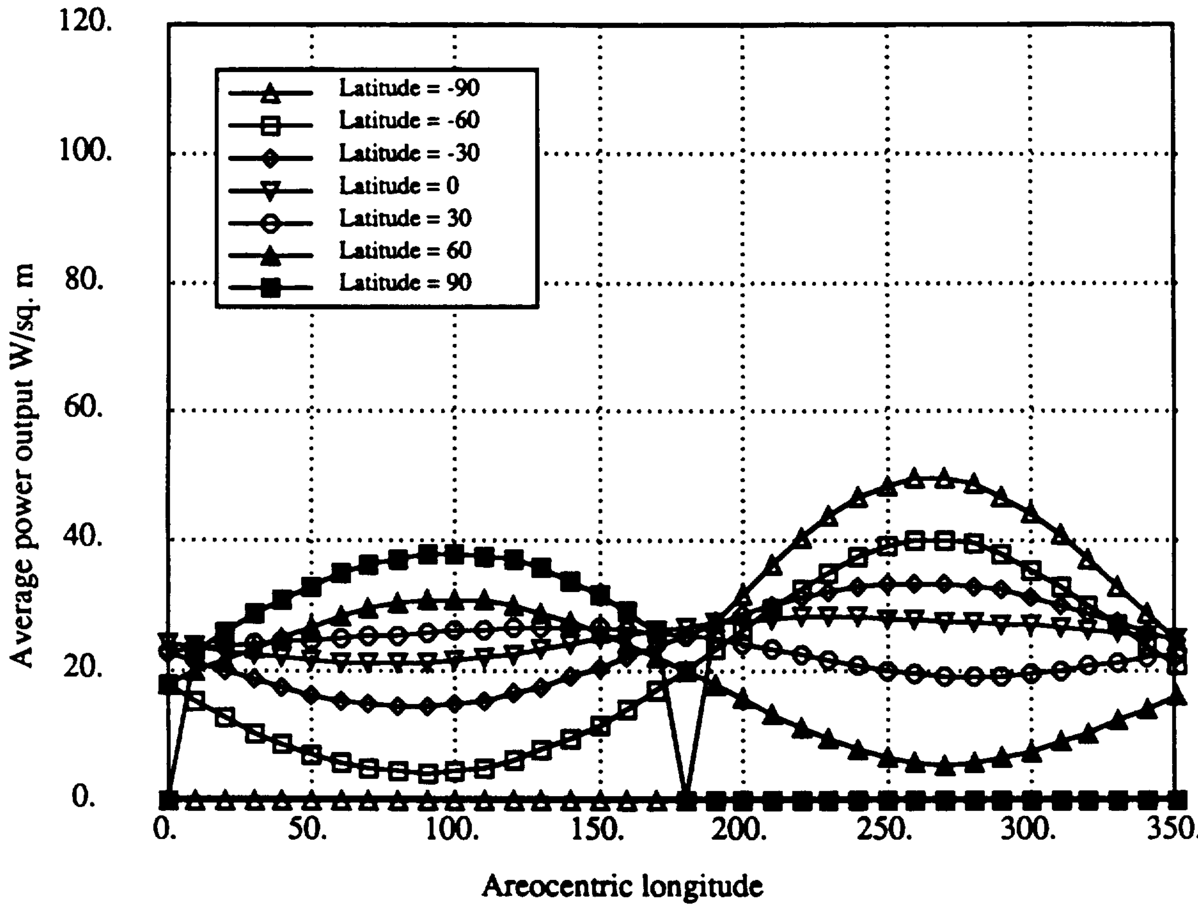

Figure 3-8. Solar panel average power output for a tracking Silicon panel with a optical depth of 2.0.

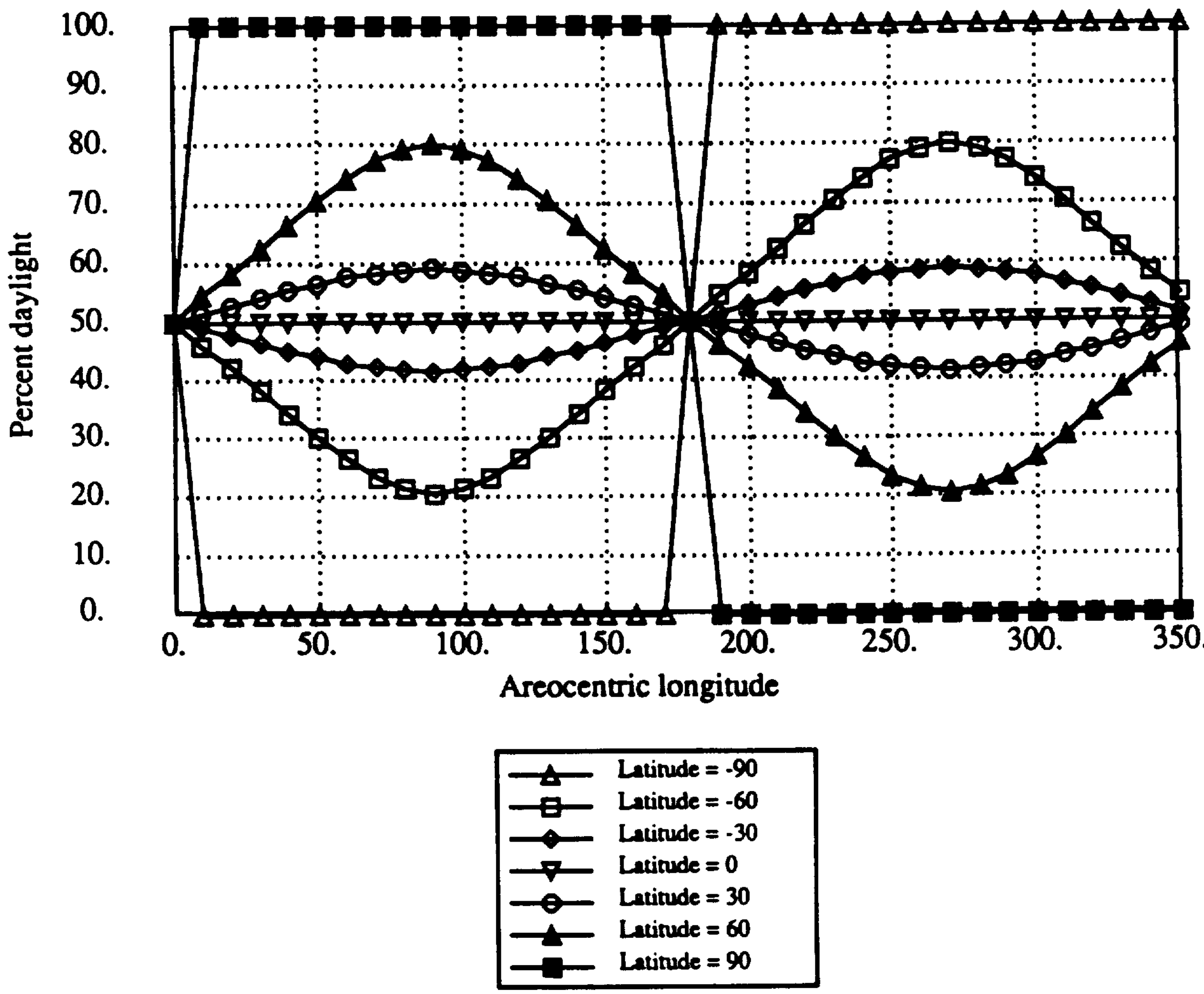

Figure 3-9. Percent of daylight per sol

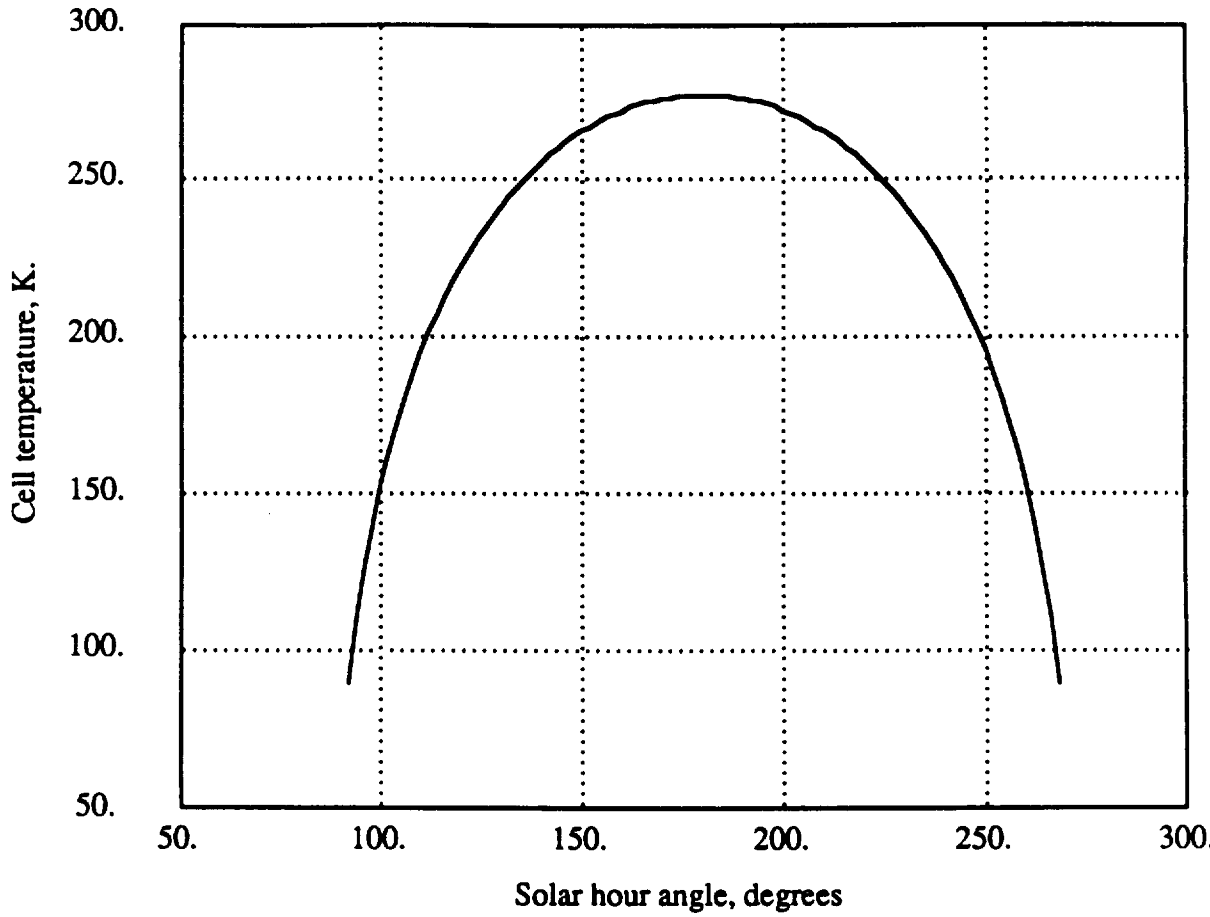

Figure 3-10. Cell temperature for the nontracking Gallium panel, areocentric longitude of 0 deg., latitude of 30 deg., optical depth of 0.5.

functions of the wavelength of the light. This results in errors in both the tracking and horizontal collector results. The degree that this effects the energy calculations is not known, of even if it results in a increase of reduction in energy.

At present, the location on Mars and the time of year when the rover will be operating is not certain. Thus there is a wide range of average power output per square meter values that be used in the design process. Consideration of all possible power levels will complicate the design process, which at the level of analysis of this study is undesirable. Thus, a single representative value will be used for each cell type. The average power output of a GaAs panel is 22 watts per square meter or more for a wide range of latitudes and areocentric longitudes. If the rover is designed for this power level, then most of Mars will be accessible to it. The equatorial region between about -20° to 20° will be accessible the entire year, making this area appropriate for extended missions. Regions further north or south are suitable for missions of limited duration. For silicon cells, a value of 17 w/m^2 will be used.

Section 4
SOLAR THERMAL ENERGY COLLECTION

4.1 Problem Description

The Mars environment is cold, with average temperatures of 210 K, dropping as low as 150 K. Any Mars rover must be able to cope with this environment. This generally requires that the rover have a thermal control system to provide heat to various systems as needed. For the designs with the primary electrical power coming from a radioisotope thermoelectric generator (RTG), the thermal control system would rely on electric power, or from small radioisotope thermoelectric heaters (RTHs).

Both RTGs and RTHs require the use of plutonium. The main reason for examining the possibility of a solar-powered rover is to eliminate plutonium. The use of solar cells for electric power eliminates the need for the RTG; however, the thermal control task remains. The most obvious way to handle it is to use electrical power from the solar panel. The power needed is quite large: for example, the current MRSR design uses 50 watts for thermal control. If this is to be supplied by solar cells, a panel with an area of 2.5 to 3 square meters will be needed for this task alone. This is based on the average power output value given in Section 3, making some allowance for storage losses.

Another way to get the energy needed is with solar thermal collectors. This has the advantage that the efficiency of solar thermal collectors is potentially much higher than the 20 percent or so that can be achieved with photovoltaics. Also, the energy can be stored as either sensible heat or latent heat. In both cases, water would make a good storage medium. Water can store 64 watt hours per kilogram in the phase change from solid to liquid, and an additional 1.16 watt hours per kilogram per degree of temperature change.

4.2 Collector Design and Analysis

Solar thermal collectors need to maximize heat absorption and minimize heat loss. One common method for doing so uses a special coating, called a selective surface, on the absorber that enhances absorption of solar radiation while reducing the radiation of long wave infrared. Also used are cover windows to reduce the loss due to conduction and convection to the air, and to some extent, the loss of heat due to infrared radiation. The windows can also be coated with a selective surface on the inside that is transparent to solar radiation but reflects thermal infrared emitted by the collector back to the absorber. The shape of the collector is usually either flat or cylindrical.

In designing a solar thermal collector for use on a Mars rover there are several factors to consider. As the collectors are supplying heat to the thermal control system, the temperature that must be maintained by that system will have a large effect on the performance of the collector system. Generally, the lower the temperature that must be maintained, the higher the efficiency of the collector. The complexity of the system also needs to be minimized so as to reduce the chance of failure. In addition, there are constraints of size and shape imposed on the collector by the design and operation of the rover.

There are two main loss mechanisms for the solar collector. The first is radiation loss. This is proportional to the temperature to the fourth power and is also a function of the surface emittance. Some materials have a very low emittance, 10 percent or less of that of the ideal black body. The other main loss term is loss to the air by conduction and convection. This loss can be estimated using the Welty equation as given in Meinel (1976):

$$H = 0.062\ k\ Re^{0.62} / L$$

where k is the thermal conductivity of the air, L is the length of the surface losing the heat, and Re is the Reynolds number of the surface:

$$Re = \rho\ V\ L / \upsilon$$

based on its length, L, the wind speed (V), the density of the air (ρ) and the viscosity (υ).
For typical Martian conditions and a carbon dioxide atmosphere we have;

$\rho = 0.020$ kg/m^3
$k = 0.0226$ W/m/K
$\upsilon = 1.07 \cdot 10^{-5}$ kg/m sec
$V = 3$ m/sec
$L = 2.5$ m

which gives $H = 0.21$ W/m^2/K. This is the combined loss due to both conduction and convection to the air. Note that it is a small loss, even if the temperature of the solar collector is 100 degrees greater than that of the air, the loss of heat is only 21 watts per square meter. By comparison black body radiation from a surface at 0°C is 315 watts per square meter.

In order to determine the amount of heat lost by convection and conduction, the temperature

difference between the air and the collector must be known. The temperature of the collector will be assumed to be equal to that of the thermal storage module, as would be expected with good thermal contact between the two. Data from the Viking landers were used to find the daily maximum and minimum air temperature as a function of season. The diurnal temperature variation between the maximum and minimum was assumed to vary sinusoidally. This model was used for all calculations presented here, at all Martian latitudes. This is not very realistic for latitudes other than that of the Viking lander, but the resulting errors are expected to be small, as most of the heat loss is by radiation.

The large difference between the convective and radiative loss terms indicates that controlling the convective loss by use of cover glasses may not be necessary, a bare collector can be used. Controlling the radiative loss can be done effectively with selective surfaces.

The relative merit of a selective surface is best described by its selectivity ratio, the ratio of its absorption of solar energy to its emission of infrared in the frequencies appropriate to the temperature at which the surface is operating. Selectivities of 10 can be achieved with coatings that are sufficiently low in cost that they are used for commercial solar collectors. Selectivities of 50 are possible with higher cost coatings. For the Mars rover, the coating must not only have a high selectivity, but must be able to withstand the Martian environment. This includes dust, low temperatures, high levels of ultraviolet, and weathering from the atmosphere. Due to the need for surviving the Martian environment, a selectivity of 10 will be assumed to be the best that can be obtained.

A thermal control system based on solar energy requires a thermal storage module. This can be connected to the collectors in one of two ways. It can be directly connected to the collector in such a manner that heat can flow either from or to storage. This results in stored heat being lost during periods of darkness. The other option is to place a "thermal diode" between the collector and the heat storage module. Such a thermal diode could be simply a pair of temperature sensors and a pump that circulates a heat transfer fluid through the collector and the store, or something more advanced. This will result in reduced heat loss at the cost of increased complexity.

4.3 Results

The average thermal power that can be collected on Mars was found for several cases. For all cases, the collector configuration was a flat plate oriented horizontally with no cover glasses. A selective surface as described above was assumed. Water was used for the thermal storage medium, with the phase change from solid to liquid at 0° C being used to store the heat. This should be warm enough for the equipment that must heated.

The four cases analyzed were with and without the thermal diode and for atmospheric optical depth of 0.5 and 2.0. For each case, results were found for several latitudes and areocentric longitudes. The results, in terms of thermal power collected averaged over a sol are shown in figures 4-1 to 4–4. Several interesting trends are shown. The average thermal power that can be collected is 60 to 80 w/m^2 for a large portion of the surface and seasons. The benefit of the thermal diode is to increase the energy collected by 25 to 30 w/m^2, depending on the season and location. Comparing the actual energy collected to the energy available shows that the collector operates at an efficiency of about 80 percent for the case with the thermal diode, dropping to 60 percent for the case without the thermal diode. This compares favorably with the 20 percent efficiency available with solar cells. The high efficiency also indicates that there is little to be gained by using a more complex collector design with cover glasses, selective windows, and so on. A bare collector is sufficient.

The effectiveness of such a simple collector design suggests an interesting possibility. The entire rover could be covered with a selective surface. This would make the rover a large solar collector, and the rover mass would become the thermal storage module. This could greatly simplify the thermal control of the rover.

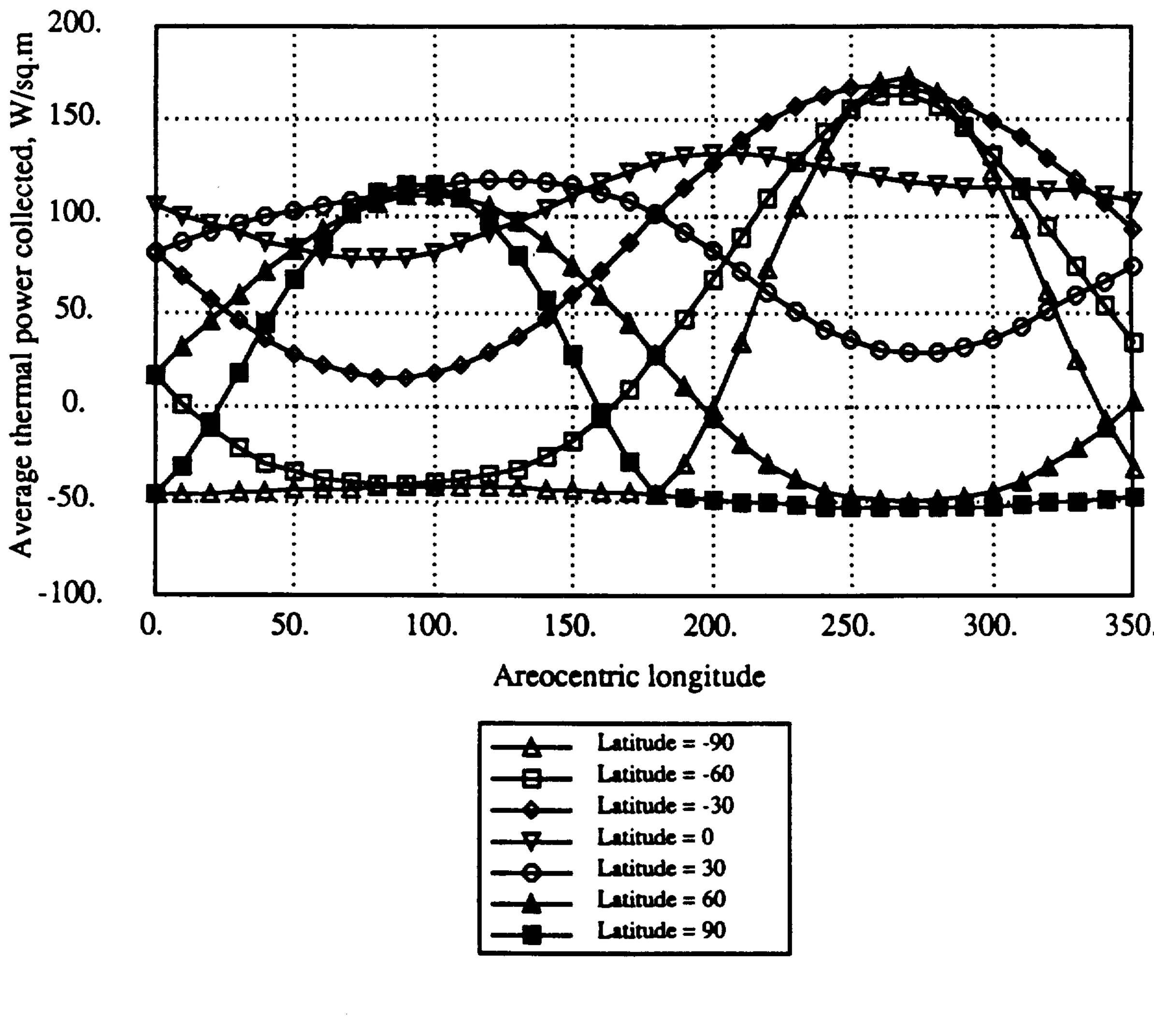

Figure 4-1. Solar thermal power collected for a horizontal absorber, selectivity = 10, wind speed = 3 m/sec, optical depth = 0.5, no thermal diode. Absorber tempreature = 273 K.

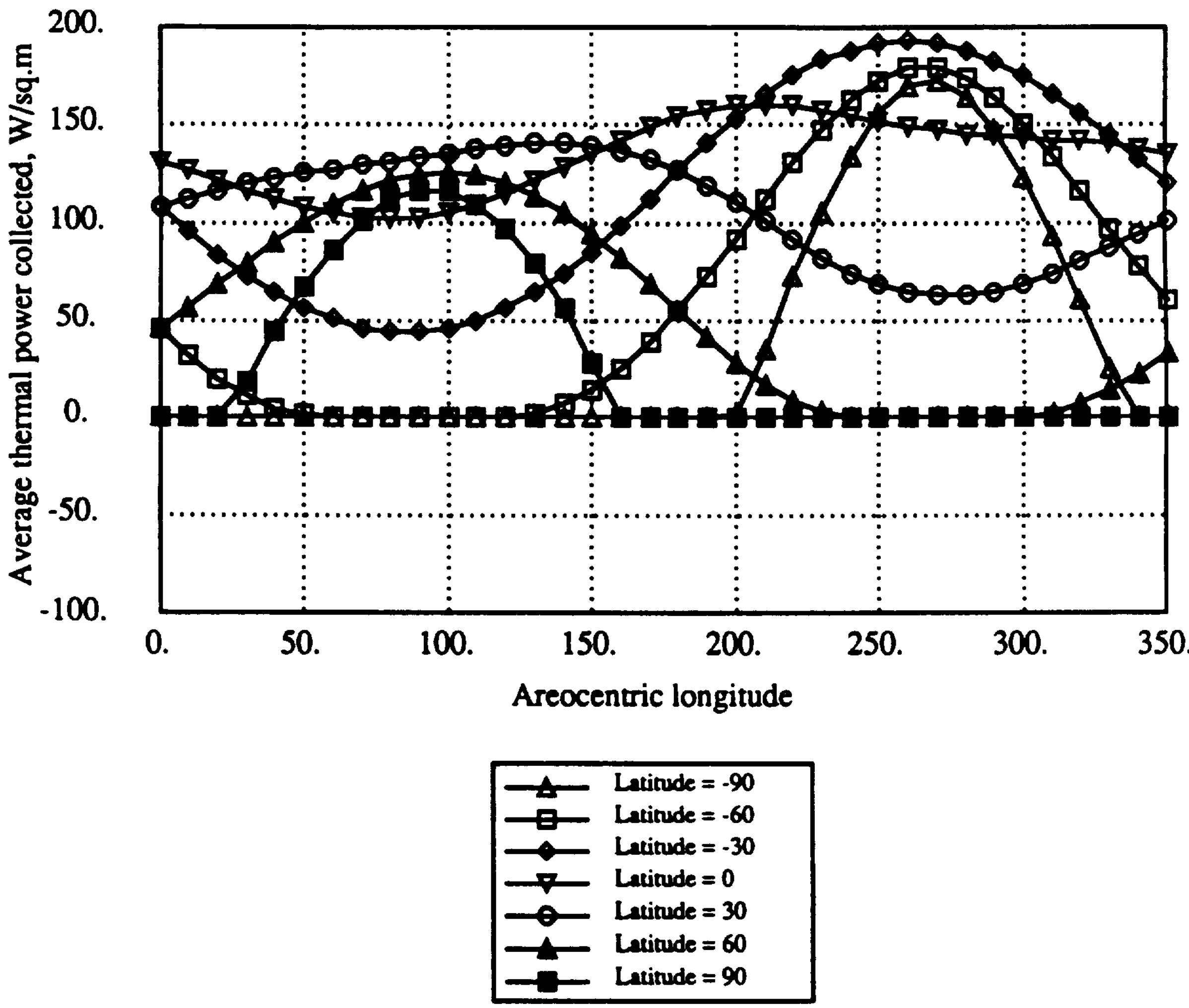

Figure 4-2. Solar thermal power collected for a horizontal absorber, selectivity = 10, wind speed = 3 m/sec, optical depth = 0.5, with thermal diode. Absorber tempreature = 273 K.

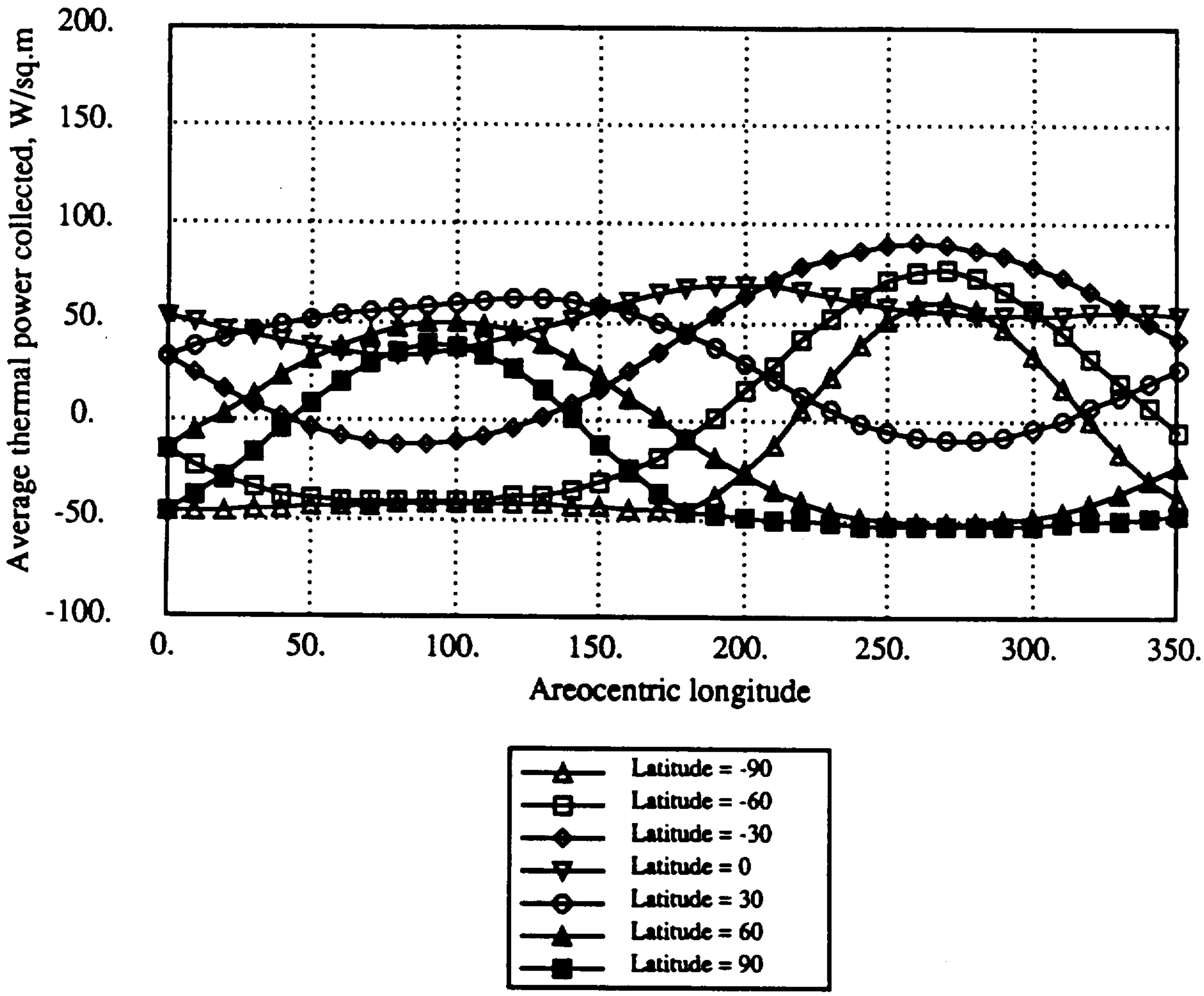

Figure 4-3. Solar thermal power collected for a horizontal absorber, selectivity = 10, wind speed = 3 m/sec, optical depth = 2.0, no thermal diode. Absorber tempreature = 273 K.

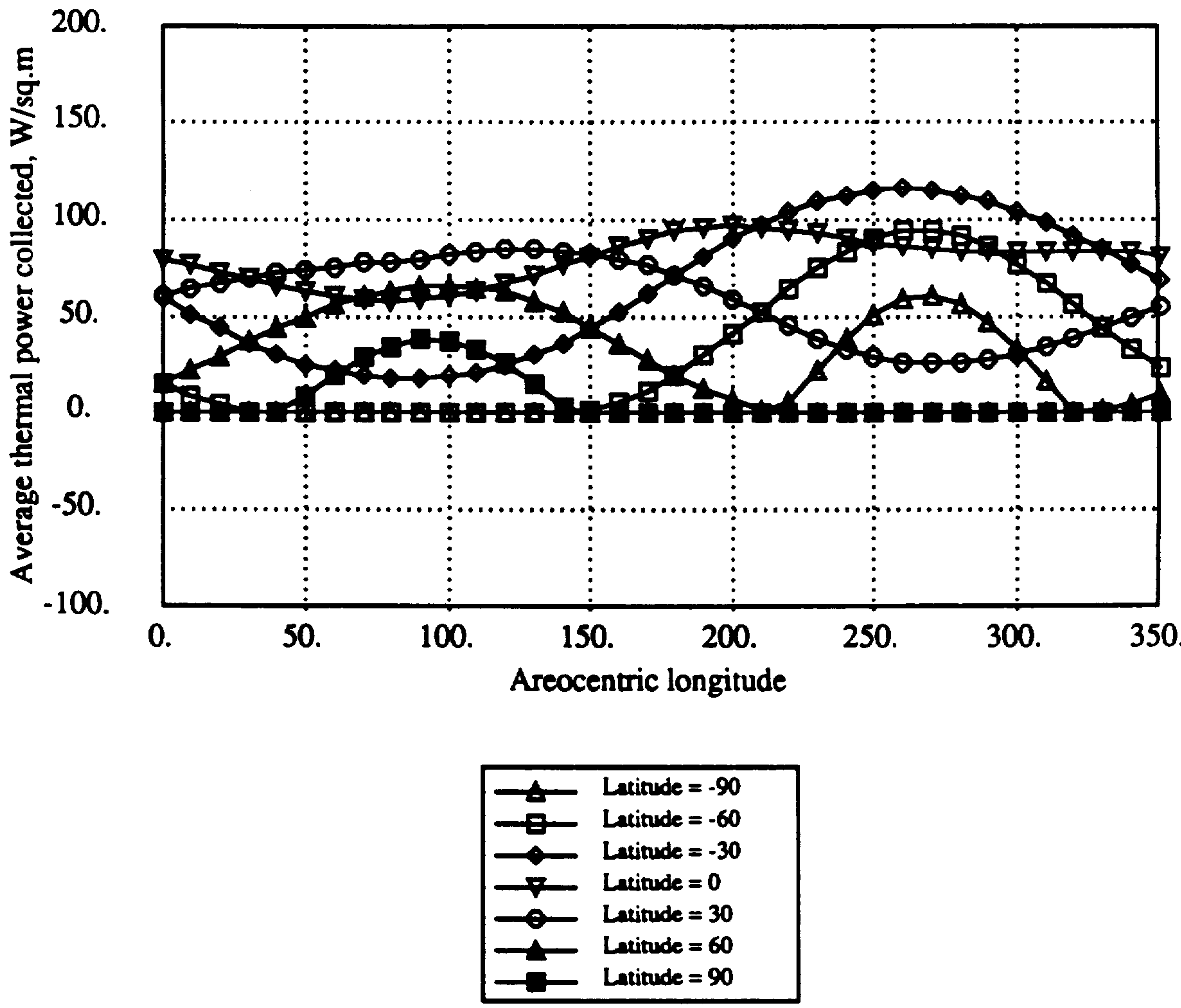

Figure 4-4. Solar thermal power collected for a horizontal absorber, selectivity = 10, wind speed = 3 m/sec, optical depth = 2.0, with thermal diode. Absorber tempreature = 273 K.

Section 5
POWER REQUIREMENTS

5.1 Assumptions

The power needs of the rover have been assumed to be similar to those of the MRSR vehicle. This vehicle has several operating modes during which each of its systems requires a particular amount of power. If each of these power requirements is known, and an operational scenario is known, then the average power needed by the vehicle can be determined.

In the following sections, the power needs of the MRSR vehicle systems will be examined, starting with the mobility system, then the thermal control system and, finally, the remaining systems. Next, the impact of these needs on the average power use by the vehicle will be determined.

5.2 Mobility Power Needs

Normally, the resistance to vehicle motion is due to three components, namely: air, traction and gradients. For a Mars rover, the air resistance is negligible due to the low rover speeds (< 1 m/s) and low atmospheric density (~1 percent of Earth's). The traction resistance is the energy lost due to deformations of the wheels and the surface on which the wheels are rolling and also energy lost in wheel bearings and seals. Values for the rolling resistance are normally given as some percentage of the vehicle weight, but reliable values are difficult to obtain. In the case of a Mars rover, where the construction details as well as the Martian surface characteristics are unknown, it is impossible to estimate with confidence a value for the rolling resistance. The rover will probably have to operate on surfaces ranging from loose sand to rocky terrain. Common sources on rolling resistance (Mark's Mechanical Engineering Handbook, eighth edition, 1979) give a range of 0.15 to 0.30 for a pneumatic tire on loose sand and 0.1 for badly cobbled roads. Because of the uncertainty in the expected value of the rolling resistance, in this report we shall use the range of 0.15 to 0.30 as representative of what may be expected for the rover operation on the Martian surface.

The power to overcome rolling resistance is then estimated from:

$$P_T = C_T W V/\eta_D$$

where: P_r = power to overcome rolling resistance (watts)
C_r = coefficient of rolling resistance (0.15 - 0.30)
W = vehicle weight (newtons)
V = vehicle speed (m/s)
η_D = drive efficiency (0.5 - 0.8).

The drive efficiency as used here is the ratio of the power available at the wheel axle to the output power at the array. This efficiency then includes the efficiencies of the electronics between the array and the motor, of the motor itself, and of the gearbox. The GM Sunraycer, a solar powered electrically driven terrestrial four wheeled vehicle developed by AeroVironment for General motors achieved drive efficiencies of about 85 percent (Sturtevant, 1989). This efficiency will be unlikely on Mars because of the large gear reduction required between the motor and the wheels, and the low temperatures. The vehicle will be operating at very low temperatures (~ 200 K), so seals and lubrication will make an unknown contribution to the total rolling resistance.

Harmonic and planetary gear systems are under consideration for the final drive. The planetary gearbox is heavier than the harmonic gear system, but has potentially high efficiency (>90 percent) instead of the lower efficiency harmonic drive (efficiency ~ 50 percent or less under partial load conditions). How these would be lubricated and sealed for the lower Mars temperatures is undecided. Because of these unknowns, we have again assumed a range for the value of the drive efficiency. As an upper bound, we have chosen 80 percent, in keeping with Sunraycer experience, and as a lower bound 50 percent.

Slope or gradient resistance is usually given as the product of the grade in percent times the vehicle weight. If a 30° slope is chosen as the maximum, then the additional resistance is one half the vehicle weight. This is somewhat misleading because there will be a weight transfer to the rear wheels when climbing a slope that may influence the rolling resistance on those two wheels, particularly on soft surfaces. Another consideration in estimating the required drive power is the necessity of overcoming a large obstacle that cannot be avoided. The power needed to climb a slope is recovered when the rover descends. If the rover always returns to its starting point, the net energy needed to climb slopes is to the first approximation zero. However, the different operating conditions caused by slopes will result in added inefficiencies in the drive system, increasing the average power needs.

Estimates of the power required to overcome slopes and obstacles depend on the occurrence of these in the terrain and also on the geometry of the rover. The emphasis in this report will be on estimating the average power requirements for sizing panel arrays. Further information is required

to estimate the additional power necessary to overcome slopes and obstacles.

- **Drive power estimates**

In Figure 5-1, estimates of the power required for the rover to operate on level ground with a rolling resistance coefficient of 0.15 and a drive efficiency of 0.5 are shown for various rover masses. It is clear from the figure that a light rover, or a slow one, will require less power. The power values in this figure are those required when the rover is in motion. For sizing of photovoltaics panel, it is more useful to have this information in terms of power averaged over the entire day. This has been done in Figure 5-2, which shows that the power requirements are quite modest. For example, at an average speed of 1 km/sol, the power required for a 500 kg rover is about 6 watts.

Based on average power required, it might be suggested that the rover mass is not a significant factor. However, Figure 5-1 shows that the heavier rovers require large instantaneous power inputs to move and particularly to overcome obstacles. This would require larger, heavier motors. For a photovoltaic power system, it is essential to maintain low power requirements and hence the emphasis in rover design should be to produce as light a rover as possible. This will assist in launch vehicle payload constraints as well.

Figures 5-3 and 5-4 show the influence of the value of the rolling resistance on the drive power requirement. To keep the power requirement low, it is desirable to have as low a rolling resistance as possible. The coefficient of rolling resistance depends primarily on the deformations of the wheels and ground. To keep the rolling resistance low in soft materials, the contact pressures must be low. This is normally done by using wheels that have a large contact patch. By reducing the weight of the rover for a given wheel size, the contact pressure can be reduced. In this case, reducing the rover weight has a two-fold effect on the rolling resistance. By decreasing the weight, the rolling resistance decreases and, if the contact patch area is maintained, the coefficient of rolling resistance will also decrease.

Figures 5-5 and 5-6 show the instantaneous and averaged power requirements for the range of values of the drive efficiency. For a low required power, the drive efficiency must be high. As demonstrated by the GM Sunraycer, the drive system can be made to have an efficiency of about 85 percent. The gearbox can be made efficient and light by choosing a planetary gear train and tailoring each stage.

Although the above examination of the rover drive power requirements is incomplete because power for slopes and obstacles has not been included, a major conclusion that can be drawn from

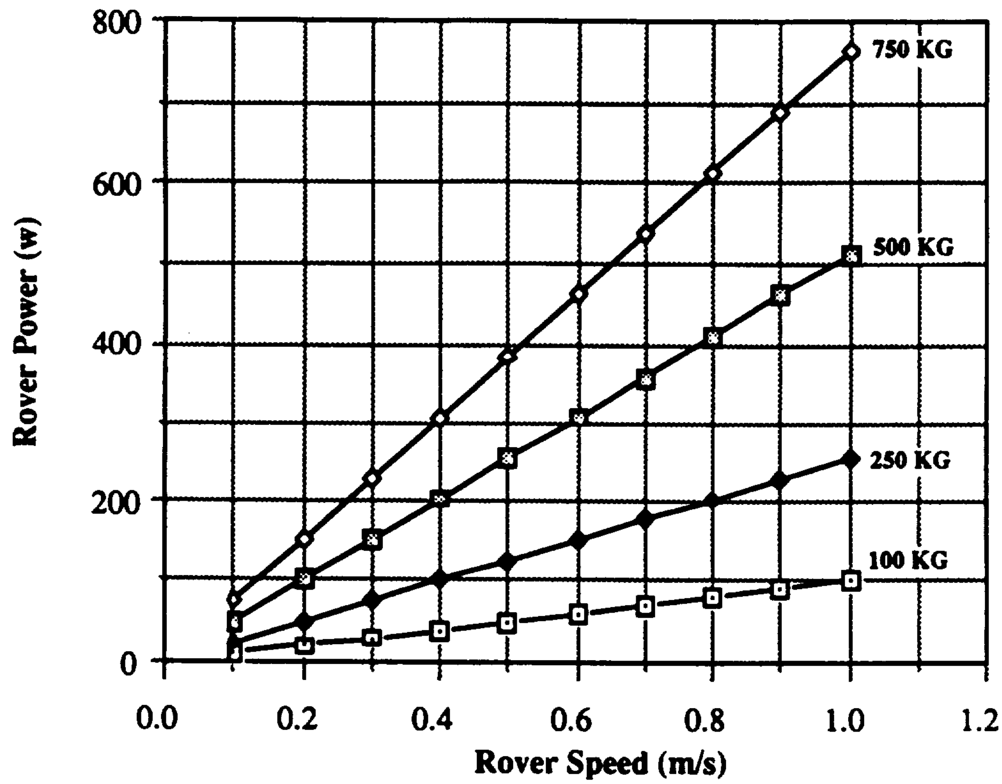

Figure 5-1. The estimated rover drive power for various speeds and masses. The coefficient of rolling resistance is 0.15 and the drive efficiency is 0.5.

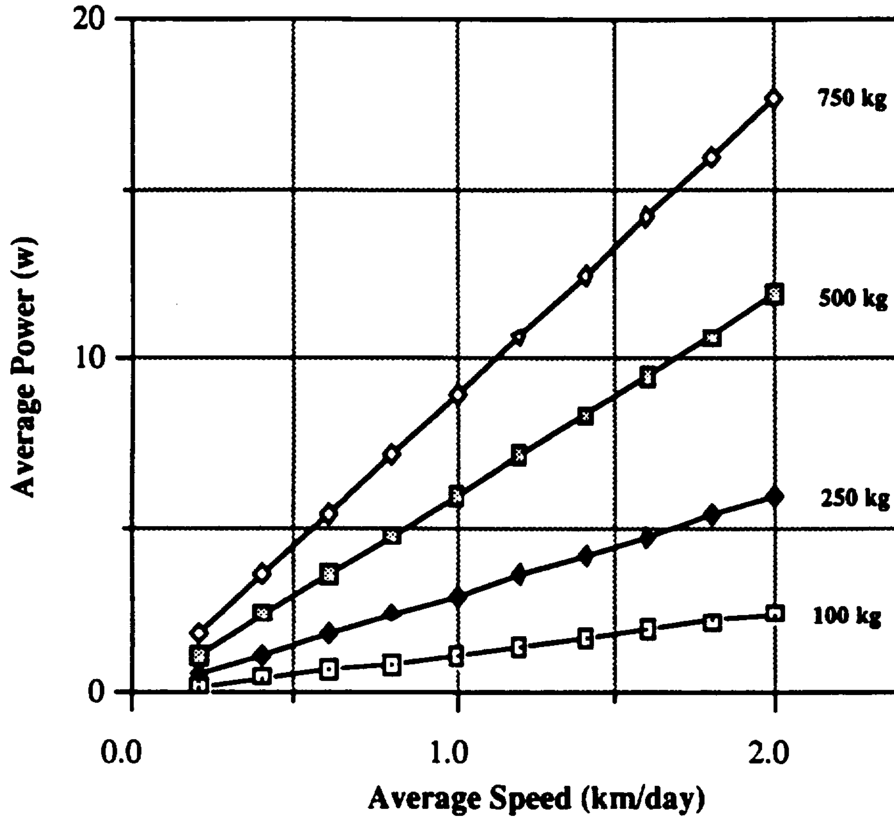

Figure 5-2. The estimated drive power averaged over the number of kilometers traveled in one Martian day.

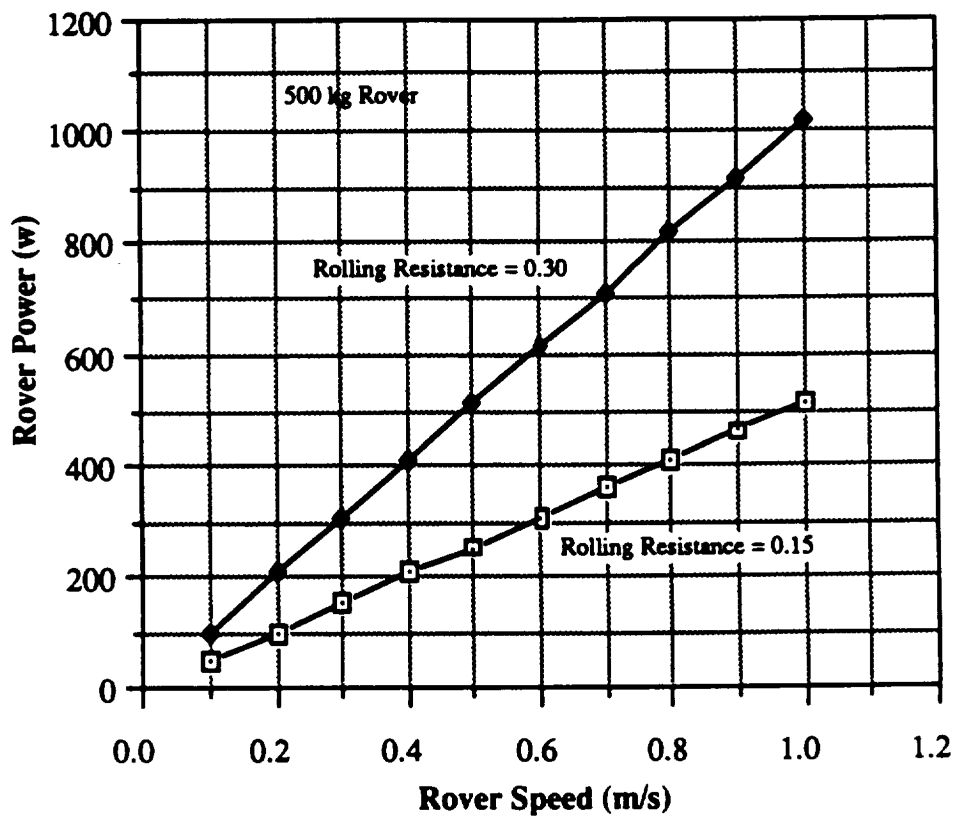

Figure 5-3. The estimated rover drive power for a 500 kg rover with a drive efficiency of 0.5 for two values of the rolling resistance coefficient.

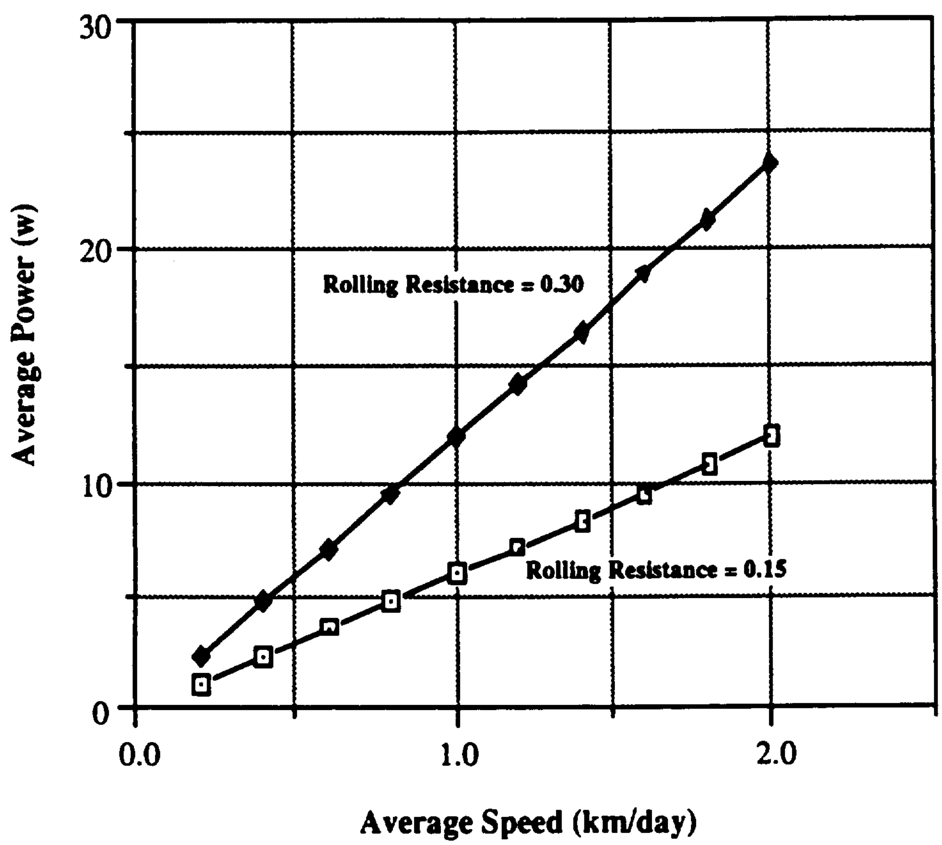

Figure 5-4. The estimated drive power averaged over the number of kilometers traveled in one Martian day. Rover mass is 500 kg and the drive efficiency is 0.5.

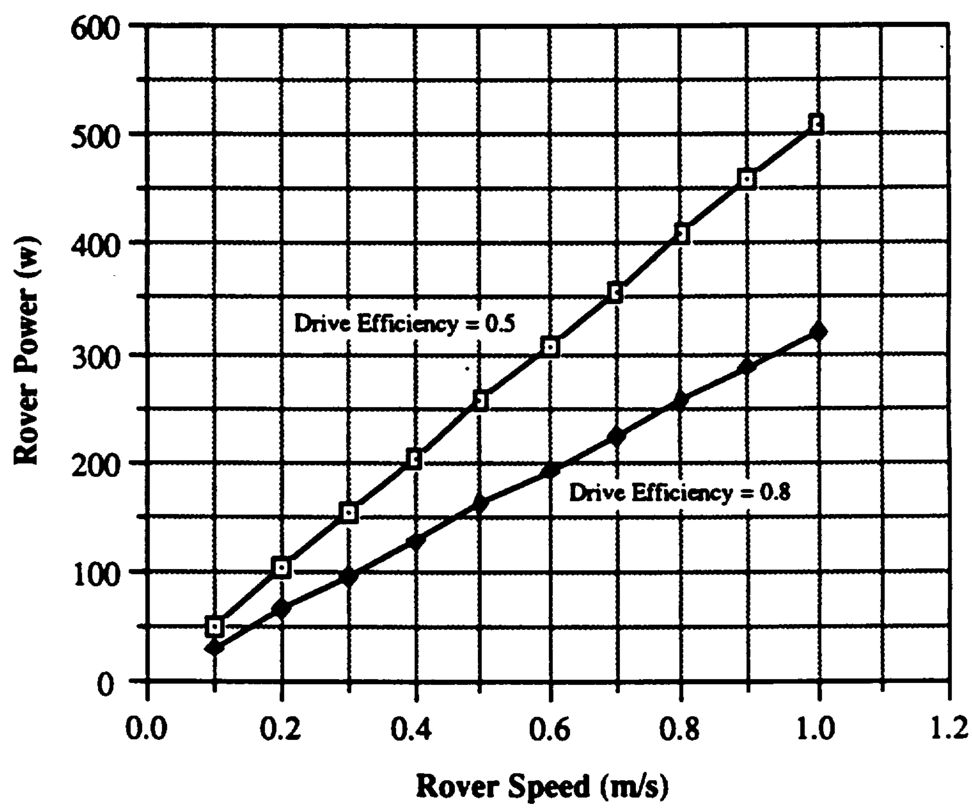

Figure 5.5. The influence of drive efficiency of the estimated rover power for a 500 kg rover and a rolling resistance of 0.15.

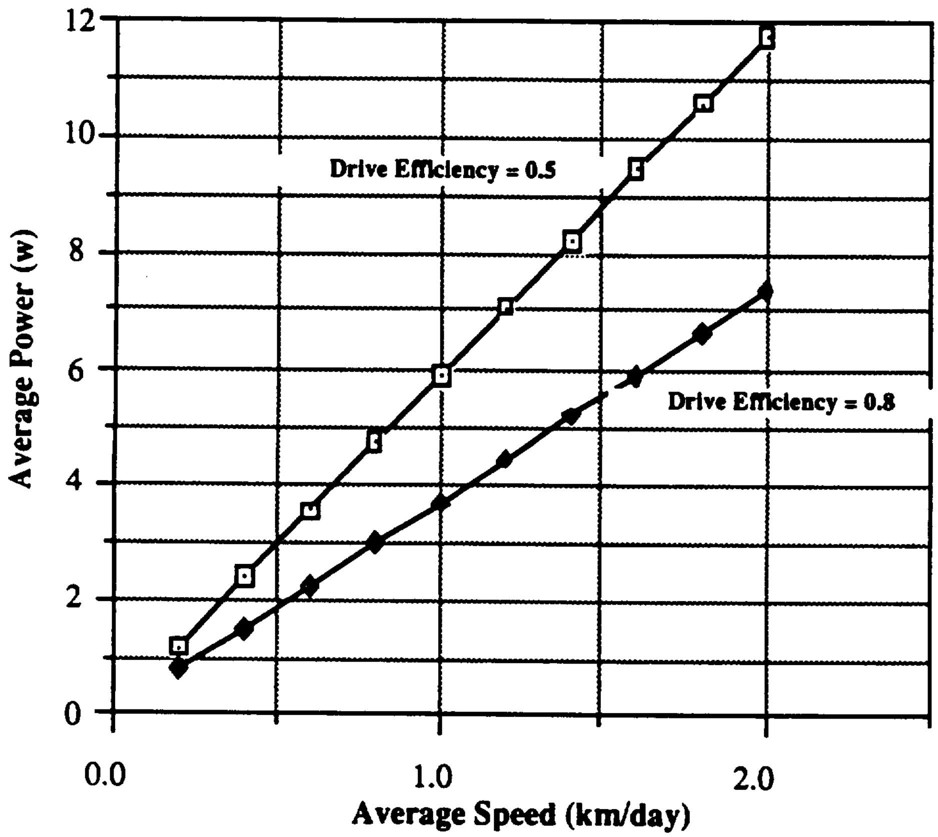

Figure 5-6. The influence of drive efficiency on average rover power. Rover mass is 500 kg and the assumed coefficient of rolling efficiency is 0.15

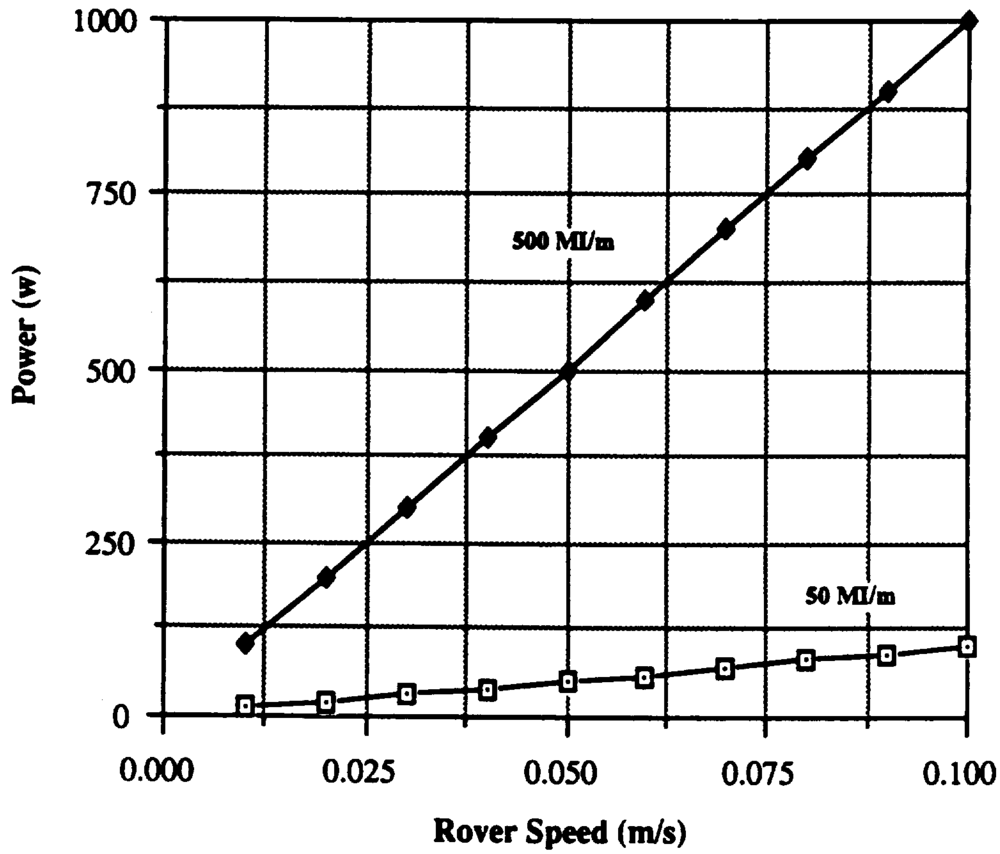

Figure 5-7. The estimated power for navigation at 20 watts/MIPS.

the study is that it is necessary to keep the rover as light as possible.

• <u>Rover Speed</u>

In the previous section it was noted that the slower the rover moved the better from the point of view of the drive power. It is worthwhile examining the rover speed for a few moments to determine what the constraints on this variable are.

During the Mars mission it is desirable to have the rover visit as many geologically diverse regions as possible. Since the mission has a limited duration, this means the rover's average speed should be high. The average speed is controlled primarily by the navigation method. There are two possible methods for the local navigation: computer-aided remote driving (CARD) and semi-autonomous navigation (SAN). With the CARD approach, stereo pictures from the rover are uplinked to Earth, where a human operator views the image on a three-dimensional display. The operator then plots a path and downlinks instructions to the rover. With this arrangement, rover steps of 10 - 20 meters are possible. However, because of the long time delay (~30 minutes) due to signal transmission at speed the of light, and the limited Earth view time (~10 hours), only about 20 steps or 200 - 400 meters would be possible per Martian day. Since the rover spends most of its time waiting for instructions, there is no advantage for the rover in moving fast and a slow speed is acceptable. The CARD approach is suitable for a mission with a limited range or a long duration.

In the semi-autonomous approach, the rover moves by comparing images and/or range information with a map uplinked from Earth that has been prepared from orbiter pictures. In this scenario, the rover may navigate several kilometers without intervention from a human operator. Because real time Earth control is not needed, the signal delay is not of concern. In this case, speed is limited by the computational power on board the rover. Current SAN software require 50 to 500 million instructions per meter of travel (Wilcox et al., 1988), although reduction to 10 million is possible. To travel at 10 cm/sec then requires a computational capability of 5 to 50 million instructions per second (MIPS). In this case, the only limit on the speed is imposed by the on-board computational capacity.

If the rover used SAN and sufficient computational capacity was available, would there be any advantage to traveling fast? If the geological areas of interest are separated by a large distance, and they are all to be visited in a short time, then the answer to this question is yes. The rover would travel quickly between the areas and then spend time studying the areas of interest. There would then be a definite need for the development of space-certified computers capable of carrying out many operations per second.

In addition, there is some advantage for increased speed during the move itself even with a fixed computational capability. The rover operates in a cyclic manner with the first part of the cycle being computational, planning the move, the later part being mechanically executing the move. During the computation part of the cycle in which it finds how to do the next move, the rover does not move. During the movement part, it does not compute for the subsequent move. The two parts of the cycle do not overlap. Thus, increasing the speed of either will increase the distance traveled per unit time. While in transit mode, the rover may spend 70 percent of its time computing and the rest actually moving. If the movement speed was increased by a factor of two, then the distance traveled per unit time would be increased by 18 percent. On a mission where 200 kilometers of movement is to be accomplished, this would add 36 kilometers to the rover's capability.

• **Navigation computation requirements**

At the present level of development, SAN requires 50 - 500 million instructions per meter of travel. The estimate for the power required to perform this number of instructions is currently not well determined. For example, Wilcox provides the following power requirements for on-board computer performance for a number of missions:

Estimated Power/MIPS

Mission	Power, watts
Galileo	200
CRAF*	20
MAX**	5
MAX with Image processing^{+}	3

*Comet Rendezvous Asteroid Flyby

**A multiprocessor, data flow computer, assumed for the MRSR Phase 1 design

$^{+}$MAX with VLSI-based image processor.

The power required for navigation is given by:

$$P_N = V\ N\ W_m$$

where: V = rover speed (m/s)

N = number of instructions required per meter of travel

W_m = power required per million instructions per second (watts).

This power requirement has been plotted in Figures 5-7 and 5-8 for instantaneous power and average power. It has been assumed in these calculations that the computational requirement is 20 W/MIPS. In Figure 5-7 the navigation power is plotted versus speed in m/s, and in Figure 5-8 it is plotted versus the number of kilometers traveled per sol. For a given distance, the power required for navigation is independent of the rover speed. It is interesting to compare the power required for navigation with that required for motion. For example, a 500 kg rover covering 1 km/sol requires between 4 and 12 watts, depending upon the drive efficiency and rolling resistance. The navigational power for 1 km/sol varies between about 10 and 100 watts, depending upon the number of instructions per meter. That is, the power for drive in the worst case is equal to the power required for navigation in the best case. Consequently, if the rover's speed is to be increased and additional power is available, the power should be put into computation.

- **Combined power requirements**

In Figure 5-9, the combined drive and navigation power have been plotted versus the distance traveled per sol. For the navigation power, it has been assumed that a nominal 150 million instructions are required per meter of travel and 20 watts are required per million instructions per second. For the rover drive power, a 500-kg rover with a drive efficiency of 0.5 and a rolling coefficient of 0.3 (i.e., the worst case) has been assumed. Again, as indicated in the Figure, about two thirds of the power are required for navigation and about one third for the drive. For the particular set of parameters chosen, the power requirement for the rover is about 45 watts/km/sol. This can be reduced by going to a smaller rover, which will reduce the drive power, but the computational requirements for the navigation will remain about the same. Hence, to reduce the power requirements for drive and navigation, the main emphasis should be on increasing the efficiency of the on-board computers.

- **Estimate of required panel size**

In Section 3, it was estimated that 20 W/m^2 is a reasonable value to use for estimating required panel size, a value about midway between the silicon and GaAs cell values. With this value, the array size required for the rover described in Section 5-6 above is shown in Figure 5-10. Again, using 1 km/sol as a basis, the rover drive requires about 0.6 m^2, whereas the navigation

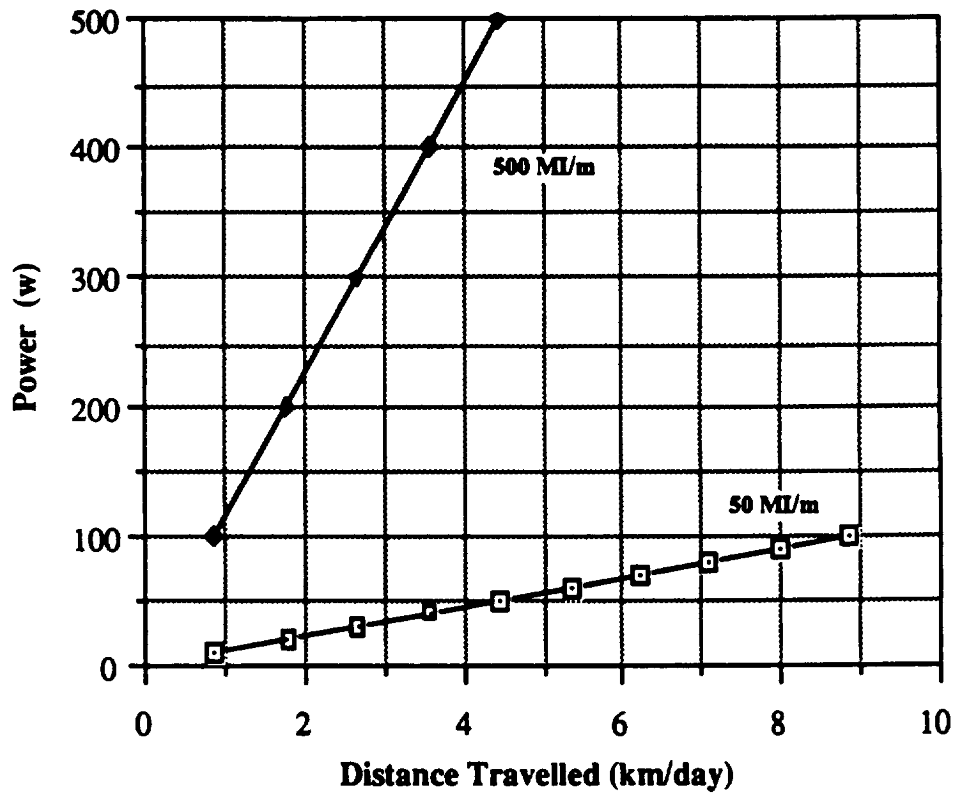

Figure 5-8. The estimated average power requirement for navigation.

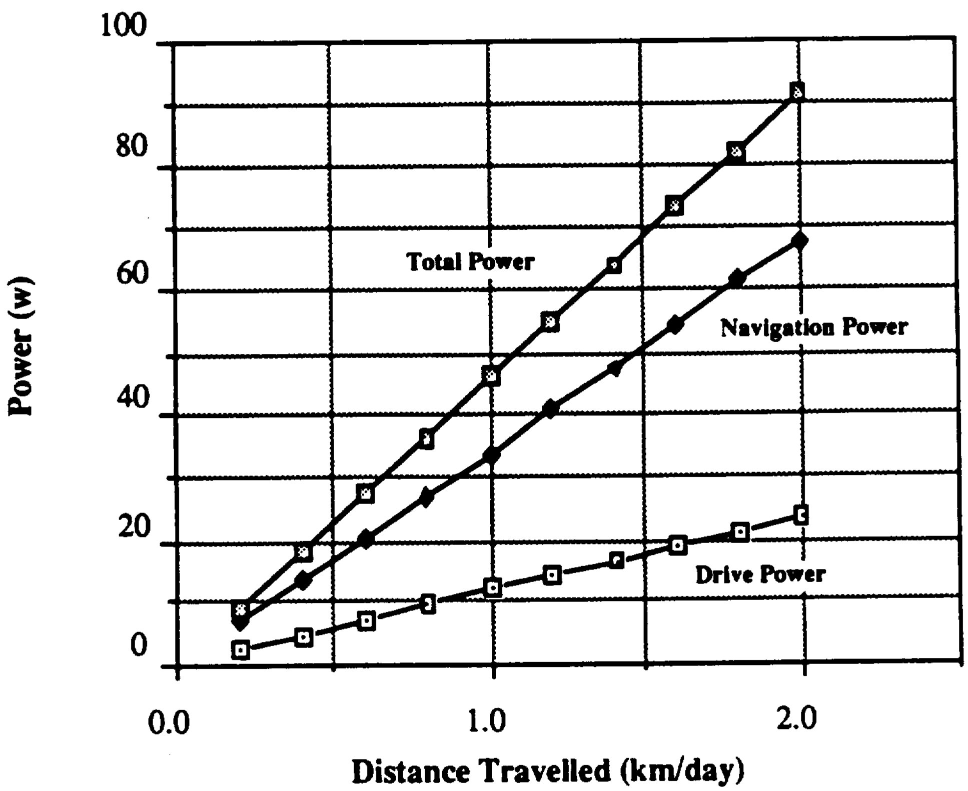

Figure 5-9. The total estimated power for rover drive and navigation. Rover mass is 500 kg, drive efficiency is 0.5 and rolling resistance is 0.30.

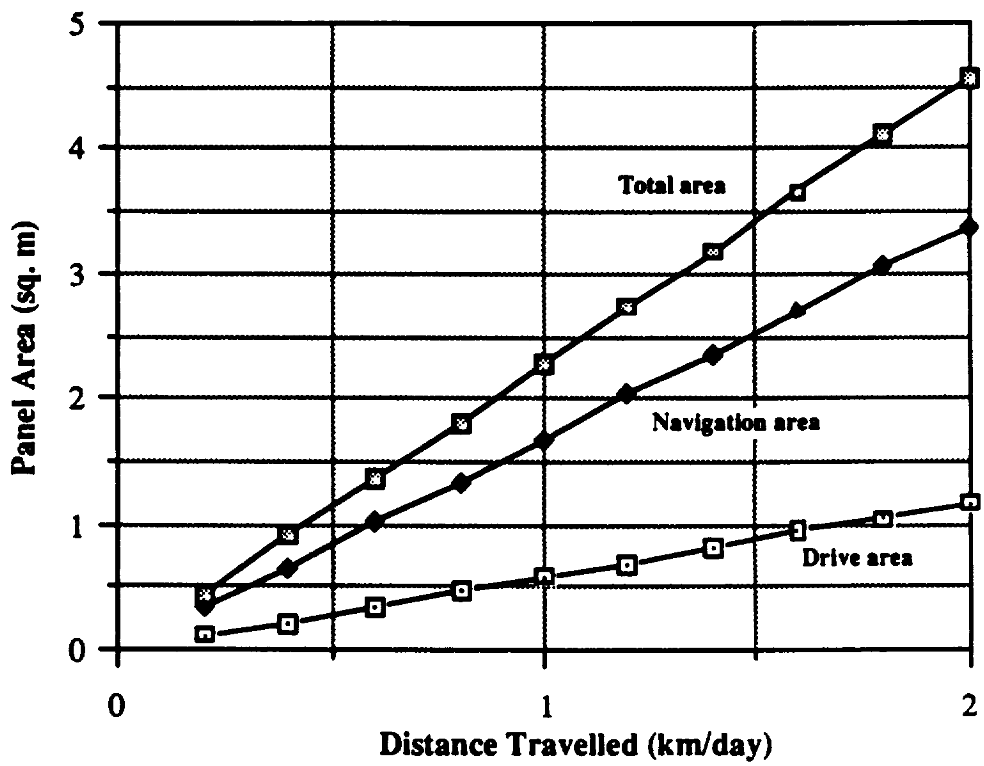

Figure 5-10. Estimated array size required to power the drive and navigation systems. Rover mass is 500 kg, drive efficiency is 0.5 and rolling resistance is 0.30.

requirement is 1.6 m^2. The total area required for drive plus navigation is about 2.2 m^2. Again, the largest reductions in this area can be made by reducing the power required for computation.

5.3 Thermal Control System

The thermal control system keeps the various systems of the rover at an appropriate temperature for operation. The most critical systems for thermal control are the electronics systems, such as the computers and the data storage systems, and the instruments. For these systems, temperatures of about 0° C should be maintained.

Much of the equipment can be expected to keep itself warm. For example, on the MRSR vehicle, the computer is expected to need 75 watts of power. As virtually all of this power will end up as heat, no additional heating is likely to be required. There will be some power requirements for temperature sensors and movement of thermal control louvers for cooling, but this will be small.

Still, 50 watts of thermal control power are specified for the MRSR vehicle. If the thermal control system obtains this from solar electric collectors, then an area of about 2.5 square meters will be needed. This is more than the area needed for mobility. If the heat is obtained from solar thermal collectors, then an area slightly less than a square meter will be needed.

5.4 Power Needs of the Complete Vehicle

The power needs of the mobility system and the thermal control system are only a small portion of the total requirements for the rover. Other systems include the vehicle control system, the data handling system, communications, sample gathering, and the science instruments. Various rover operating modes use these systems to differing degrees and at different power levels. To estimate the average rover power needs, a typical operating scenario as given in Muirhead (1988), the baseline scenario, was examined.

The scenario is for a traverse and the collection of a sample. The total time for this process is 16 hours, and includes time for Earth-based decisions to be made. A plot of power as a function of time is shown in Figure 5-11. The distance covered by the traverse can be found from the time spent in the traverse mode, about 100 minutes, the portion of time spent moving, 50 percent, and the speed while moving, taken as 0.1 meters per second. The traverse is 300 meters, for this case. Note that 72 percent of the time is spent in the idle mode. The power level in idle mode is 250 watts. The average power level for this scenario is 275 watts, only slightly more than the idle power. The average rover speed in this scenario is 0.48 kilometers per sol.

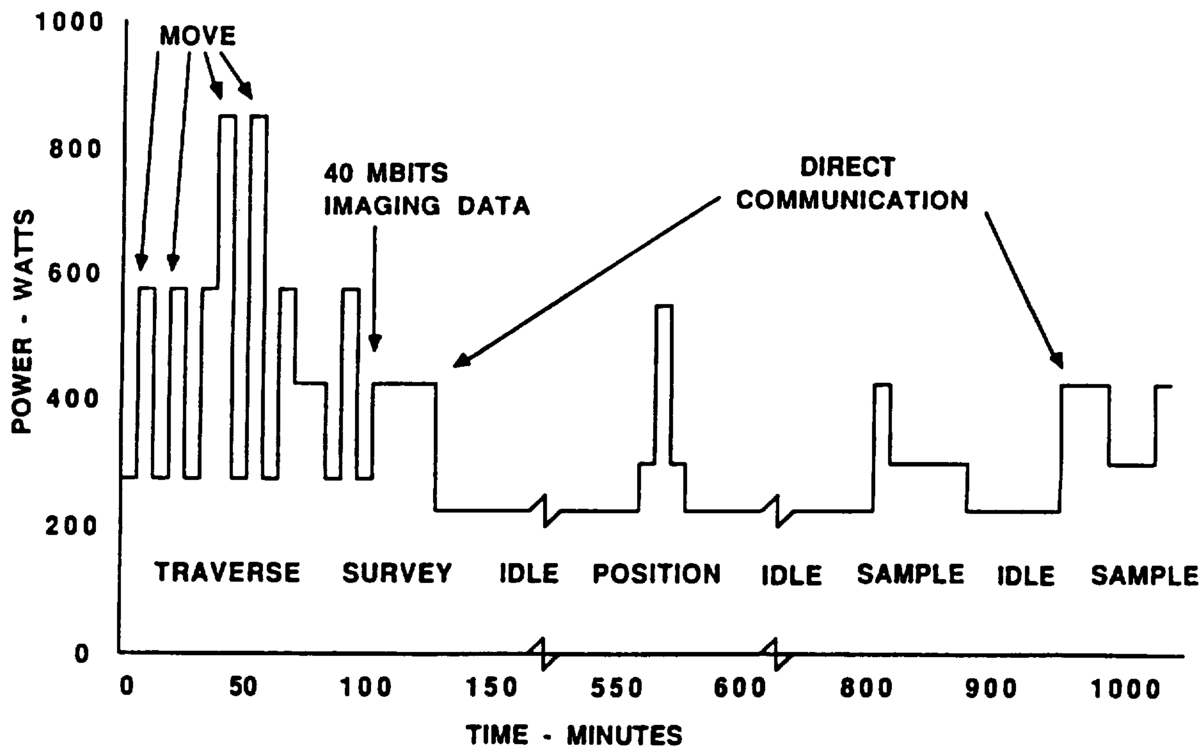

Figure 5-11. Power vs time for the baseline operations scenario.

If the traverse distance is increased to 3000 meters, before the sample is taken, then the average power will increase to 380 watts, and the total time increases to 31 hours. The average speed for this case is 2.4 kilometers per sol.

A number of interesting conclusions can be drawn from these scenarios. First, most of the energy is consumed by systems that are always on, even in idle mode. These systems are computation at 75 watts, data handling at 20 watts, vehicle control at 30 watts, power system losses at 20 watts, science at 10 watts, thermal control at 50 watts, and a 45-watt margin. In terms of reducing power needs, the largest gains can be made by improvements in these systems. For example, the computation power needs could be reduced during the idle mode. The computers are expected to use CMOS technology, which has the characteristic that the power use is almost directly proportional to the clock rate at which the computer is driven. During idle mode, the clock rate could be reduced, saving power. The clock rate could not be reduced to zero, as the computer is still needed to do some tasks in idle mode. If a savings of 60 watts is possible, then the average power needs would be reduced by 20 to 40 watts, for the 3000-meter and the 300-meter traverse cases respectively.

Another conclusion comes from the small difference between the idle power and the average power. One proposed operating mode for a solar rover is to operate only every other sol and charge batteries on the off sol. The benefit from such an operating mode can be quantified by considering what would be the resulting two-sol average power. During the off sol, the power required would be equal to the idle power, 250 watts. During the on sol it would be 275 watts, for an average of 263 watts. This is not much of a savings over the 275-watt requirement of full-time activity.

This situation would change if a new "survival" or "sleep" mode could be specified. In this mode, the rover would close down as many systems as possible in order to save power. It is possible that the power requirements could be reduced to about 80 to 100 watts, with much of this being thermal power. With the sleep mode, operating on every other sol would become a feasible method for saving energy. For example, if the power needs in the sleep mode were 30 watts electrical and 50 watts thermal, then operation every other sol would reduce the electrical power needs from 100 watts average to 65 watts average. Operation only every third sol would reduce the average power needs to 53 watts. The ability to reduce the average power in this manner would be of great use to the rover mission. It would allow the rover to survive long, severe dust storms. In addition, year-round operation at the higher latitudes would become possible.

The low average speed of the baseline scenario is also of interest. The CARD method of

navigation is capable of covering 0.2 to 0.4 kilometers per sol if no time is used for sample collection. This distance is similar to the baseline scenario considered above. The range using CARD could be more than doubled if communication with the rover was continuous (by use of a communications satellite), allowing ample time for sample collection. If the baseline scenario is considered to be representative of typical rover operations, then CARD navigation may be sufficient.

The average power is quite high compared to the power needed for mobility. Based on the results given above, the average power needed to move 0.48 kilometers per sol is 25 watts, about 10 percent of the total power needs. The MRSR baseline design includes 500 watts of power production capability. Based on the average power found above, this is excessive. (Note that the power requirements used to get the average did include a margin). The excess power is intended to allow for operating modes that use more than the average power level without the need for a large battery bank. For an RTG-powered system, it is better to have a larger than needed power production capability instead of a large battery bank. However, a large solar array area would result in handling difficulties. In addition, a solar rover will need a large battery bank in any case to provide power at night. For these reasons, the solar array will be sized for the average power requirement, not the peak power.

Based on these results, the power needs for the rover will be assumed to be 275 watts total: 225 watts electrical and 50 watts thermal. This constitutes the baseline case.

Two other cases will also be considered. The second case is a power system that mimics the current RTG power system, capable of supplying 500 watts continuously.

The third case is a low power one. In this case, the idle mode is assumed to be replaced with a "sleep" mode that uses 30 watts electrical and 50 watts thermal, so that the average electrical power need is 100 watts.

5.5 Peak Power and Storage

Neither the production of energy nor its use will be continuous and uniform. As shown in Figure 5-11, the power requirements of the rover vary by as much as a factor of three depending on the task it is doing. There are some cases where the power requirements could be as high as 2000 watts, such as climbing over a large block. The output of the solar panel also varies with time, as its average power output is about 15 percent to 20 percent of its peak output. Thus a panel designed to provide an average output of 320 watts will, at its peak, output up to 2100 watts. Of course, for half the time, the Martian night period, the solar panel produces no power at all. Due to

these variations in supply and demand of power, some form of storage will be needed. The storage must be capable of both providing and absorbing the peak power levels given above and must also have sufficient capacity to provide for rover operations through the night. Note that as the idle power is so close to the power needed for normal operations, there is little savings in not operating at night. This situation could change if a "sleep" mode became available.

The storage capacity should be large enough for about 16 hours of rover operations. This will see the rover through the time from late afternoon through the night to early morning, the period when the solar power can be expected to be less than the power demand. Using the above assumption of 225 watts electrical and 50 watts thermal of average power requirements, the capacity of the storage would be 3.6 kWh electric and 0.8 kWh thermal. In order to insure that the storage system will have a long life, that is, a large number of charge-discharge cycles, the depth of discharge should be less than 100 percent. For the electrical storage, a 50 percent depth of discharge is assumed to be satisfactory, bringing the total capacity up to 7.2 kWh. For the thermal storage, a 75 percent depth of discharge is assumed, bringing the total thermal storage up to 1.1 kWh.

The electrical storage is currently defined as being done by lithium titanium disulfide batteries. These batteries can store 100 W-hr/kg at 100 percent depth of discharge (O'Donnell, 1988). Thus 72 kg of batteries will be needed for the rover. The thermal energy will be stored in water using the phase change water from ice as the storage method. This stores 64 W-hr/kg, so the total ice mass required is 17 kg.

If the sleep mode replaces the idle mode, then the required electrical store size is reduced. If rover operations at night are similar to daytime operations; that is, no attempt to save energy overnight is made, then the required electrical store size is reduced to 44 percent of the values given above, 32 kg. If all of the idle time is shifted to nighttime, then the electrical store requirements drop to 13 percent of that given above, 9.4 kg.

If a power system is required that duplicates the capability of the MRSR power system, then the batteries must be able to provide 500 watts continuously. As shown in above, this much power is not needed,but a design that can supply it provides a worst case data point. For this case, the battery mass would have to be 167 kg.

There will be some losses in the storage system. In order to properly define these losses, a full energy flow model would be needed. At the level of analysis of this study, it will simply be assumed that the storage system losses will be about 20 percent of the energy stored. About 60 percent of the energy produced by the solar cells passes through the storage, so the losses are taken as 12 percent of the total energy collected. Thus, to provide 225 watts electrical to the rover

on an average basis, 256 watts average must be produced by the panel. For the case with the sleep mode, the panel must provide 116 watts average, so that the rover will receive 100 watts after storage losses.

The peak power requirement for the battery store is about 2000 watts in both charge and discharge modes. This corresponds to 28 watts per kilogram for the 72 kg pack, or a C/3.6 rate (i.e., a rate that would change the amount of energy stored in the battery by 1/3.6 = 28 percent in one hour). Most battery technologies, such as Ni-Cad, lead acid, and nickel hydrogen can handle such rates. Little information is available on lithium batteries, but it is anticipated that no problems will occur.

Section 6
PANEL DESIGNS FOR THE MRSR ROVER

6.1 Area Required

The results of the previous sections give the required power output of the solar panel and the power per unit area for a solar panel on Mars. The required panel size can now be found.

The average power output of the panel will be taken as 22 watts per square meter for GaAs cells and 17 watts per square meter for silicon, in line with the approximation given in Section 3. A more exact value would require better knowledge of the rover is operating position on Mars and the season. These have not yet been determined for the MRSR mission. However, these values are valid for the majority of Mars over the majority of the seasons.

For solar thermal collection, the average energy collected will be taken as 60 watts per square meter. This is the appropriate value for the case of no thermal diode. Again, this value is also valid for the majority of Mars over the majority of the seasons.

In order to provide the needed 256 watts of electrical power, a 11.6 m^2 GaAs solar electric panel will be needed, or a 15.1 m^2 silicon solar panel. For the thermal energy collector, an area of 0.83 m^2 is sufficient. The total solar collector area is thus 12.43 m^2 using GaAs cells and 15.9 using silicon. If the full 500 watts of electrical power are needed, then the panel sizes would be 22.7 and 29 m^2 respectively .

For the case where the sleep mode is available, the area of the electrical portion of the solar panel is 5.3 square meters of GaAs cells, of 6.8 square meters silicon cells. The total area is thus 6.13 and 7.63 m^2, respectively. If a panel design is desired with a width no greater than that of the main body of the rover, then its area would be about 7.6 square meters, comparable to the required panel size.

Figure 6-1 shows a planform view of the MRSR vehicle, with the solar panel overlaid onto it. Three panel sizes are shown with areas of 6.13 m^2, 12.43 m^2, and 22.7 m^2. For both the larger panels, the width is greater than the main body of the rover, but for the smallest one it is narrower than the body.

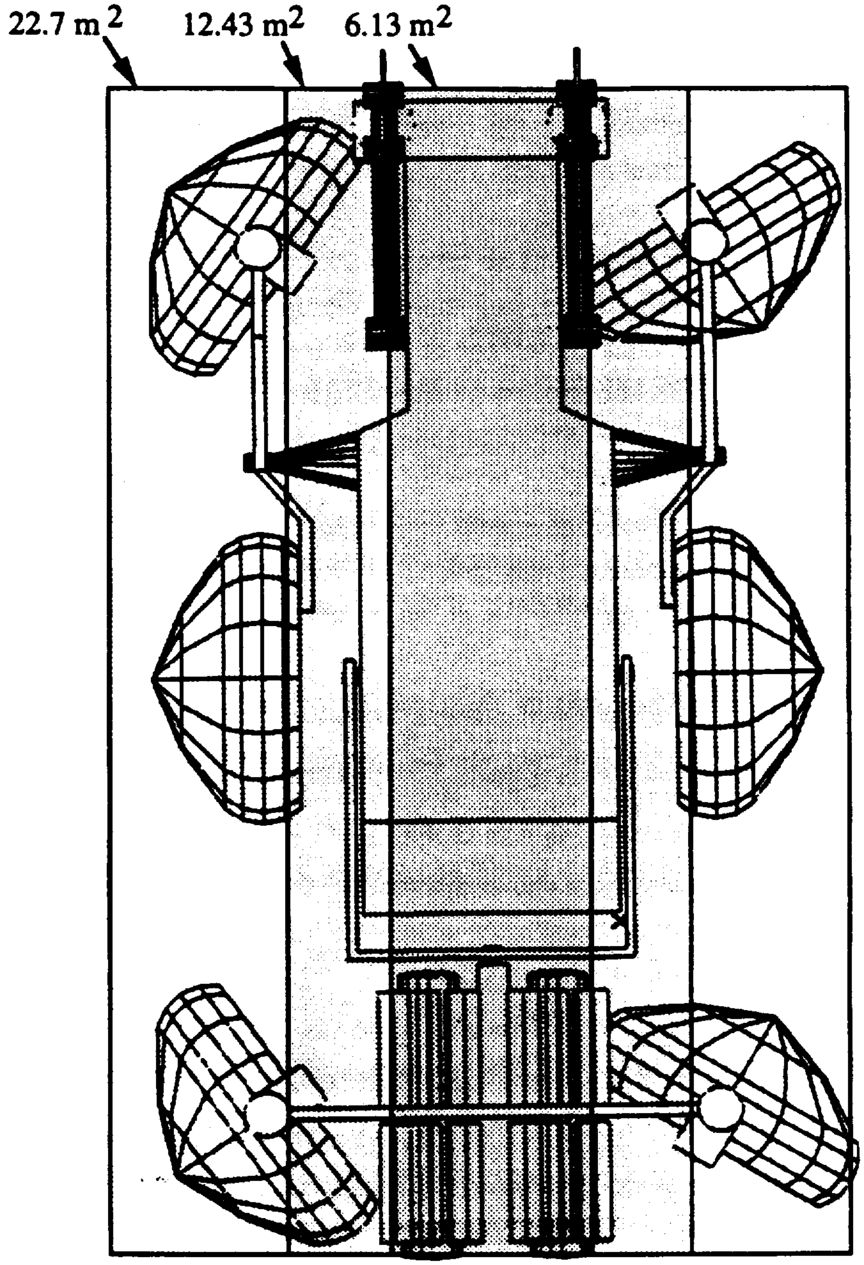

Figure 6-1. MRSR vehicle with solar panels of various sizes

6.2 Power System Mass

The mass of the power system is the sum of the panel mass, the battery mass, and the mass of the electronics. Here only the mass of the first two components will be considered, as they contain most of the system mass and have the greatest variability from case to case. The mass of the panels per unit area depends on the cell type and the level of technology. In the case of GM Sunraycer (Sturtevant, 1989), the electrical portion of the silicon panel had a mass of 1.2 kg per m^2, and for GaAs, 2.3 kg per m^2. The structure mass was 1.6 kg per m^2 in both cases. The cells to be used for the rover will most likely be lighter than those used on Sunraycer. Using data provided by Stella (1989) the mass of a GaAs panel can be expected to be 1.88 kg per m^2, and silicon, 0.98 kg per m^2. Assuming that a similar improvement can be made in the structural mass, a total mass of 2.88 kg per m^2 will be used for the GaAs panel, and 1.98 for the silicon panel.

Table 6-1 shows the mass of the photovoltaic collectors and batteries for several cases covering the two cell types, three average power levels, and a case where the rover is in sleep mode during the night.

Table 6-1

Electrical system mass

Case	Panel area m^2	Panel mass kg	Battery capacity kWh	Battery mass kg	Total mass kg
500 W, GaAs	22.7	65.4	16.7	167	232.4
500 W, silicon	29	57.4	16.7	167	224.4
225 W, GaAs	11.6	33.4	7.2	72	105.4
225 W, silicon	15.1	29.9	7.2	72	90.1
116 W, GaAs	5.3	15.3	3.2	32	47.3
116 W, silicon	6.8	13.5	3.3	32	45.5
116 W, GaAs*	5.3	15.3	0.94	9.4	24.7
116 W, silicon*	6.8	13.5	0.94	9.4	22.9

* Sleep mode at night (no night operations)

The mass of the thermal collector is equal to 17 kg for the thermal storage plus an estimated 4 kg

for the collector, bringing the total to 21 kg. Note that the thermal collection and storage system would not be necessary for the 500-watt electric case.

Table 6-1 shows that there is a large range of possible masses for the electrical collection and storage system. Depending on the case, the mass ranges from 241.8 to 22.9 kg, over a factor of 10. The mass of the proposed RTG power system is 129 kg, comparable to the 225-watt case, when the mass of the thermal collection and storage system is included, and some allowance is made for electronics.

The cases where the sleep mode is available show the lowest masses, especially when the sleep mode is used exclusively at night. For these cases, using nickel cadnium batteries is a reasonable consideration. Ni-Cads store only 28 W-hr/kg at 100 percent depth of discharge, and can typically be used at 75 percent depth of discharge. As a result, a Ni-Cad pack will mass 2.5 times the mass of the lithium titanium disulfide batteries.

Section 7
CONCLUSIONS

The solar power available on Mars ranges from a low of zero to a maximum of 700 watts per square meter. Averaged over one sol, the maximum is 225 watts per square meter, during summer in the Southern hemisphere, which occurs near perihelion. Near the equator, the average solar power is 100 watts per square meter or more all year. In any season, over half of Mars receives 100 watts per square meter or more. Most places on Mars get this much for more than half the year.

Atmospheric dust is not a major issue in terms of reduction of surface solar energy. The above figures include the effects of a normal amount of dust. During the planetary storms, the amount of solar energy is reduced to about 60 percent of these clear sky values. Previous estimates of solar energy during dust storms were much lower, as they considered only the direct portion of the solar energy. During dust storms, the direct portion is quite low, but the scattered portion is high, resulting in only a moderate loss.

Silicon and GaAs solar cells of current technology both achieve high efficiency on Mars, with silicon achieving 17 percent, and GaAs achieving 22 percent. The low temperatures on Mars increase the efficiency of the cells, with silicon benefiting to a greater extent than GaAs.

Any energy needed for heating is best obtained from solar thermal collectors instead of electrical heaters, provided the mass to be heated is at a temperature near 0 degrees C. The efficiency of solar thermal collectors is 60 percent to 80 percent, three to four times better than that of solar electric collectors. The collector itself can be quite simple, no tracking, concentration, or even a cover glass is needed, although it is necessary to coat the collector with a selective surface. The thermal store can be integral with the collector, or separate. If integral, the store loses heat throughout the night, resulting in a net collection efficiency of 60 percent. Separate storage allows the efficiency to increase to 80 percent, but requires the use of a pump or some system to transfer heat from the collector to the store as needed.

The rover uses energy for several purposes. Of these, movement uses a small portion of the total, on the order of five percent. The energy used by the computers that carry out the calculations required for movement is two to four times as great as the energy needed for movement itself. Various other systems on the rover (communications, data handling, science, thermal control, and vehicle control), many of which run continuously, consume the remaining energy.

The rover spends most of the time in an idle mode, while signals are traveling between Mars and

Earth, and plans are being made. The power level during idle mode is quite high, 250 watts. In the present MRSR design, little effort has gone into lowering the idle power level, as the power available from the RTG is always 500 watts whether or not it is needed. The average power needed by the rover is actually not much greater than the idle power, 275 watts. This average power use is based on scenario where the rover moves 300 meters and collects a sample, all during a 1000–minute period. In this scenario, 72 percent of the time is spent in idle mode.

The small difference between the average power needs and the idle power needs means that little energy can be saved by operating the rover only every other sol, with the off sol being used to recharge the batteries. This type of operation will only lower the long term average power from 275 watts to 262.5 watts. Reducing the power needs during the idle mode will greatly reduce the average power needs. The introduction of a "sleep" mode, with a power requirement of 80 watts, in place of the idle mode, would reduce the average power needs to 150 watts. Of this, 50 watts would be thermal power, 100 electrical. With a sleep mode, the option of operation only every other sol becomes viable, and would lower the average power to 65 watts electrical and 50 watts thermal.

The solar rover will require a large energy storage system. The main driver on the size of the energy store is the need to survive the night. Even in idle mode, the nighttime energy need is large, so that, based on the average energy needs, 7.2 kWh of battery capacity and 1.1 kWh of thermal storage capacity will be needed. This requires 72 kg of batteries for the electrical store and 17 kg of water for the thermal store. Introduction of a sleep mode would reduce the electrical store size by a factor of two to seven, depending on how much of the idle time is shifted to nighttime.

The losses associated with the storage system can be expected to increase the total energy requirements by 16 percent for electrical energy and a negligible amount for thermal energy. Thus, the total average electrical power requirements are 256 watts for the baseline case and 116 watts for the case with the sleep mode. In both cases, the thermal power needs are 50 watts.

The panel areas for the baseline case are 11.6 square meters electrical and 0.83 square meters thermal for a total of 12.43 square meters. For the case with the sleep mode, the areas are 5.3 square meters electrical, 0.83 square meters thermal, for a total of 6.13 square meters. For comparison, the area of the top of the present MRSR rover, including the top of the RTG's (which would be replaced by the battery packs), is 7.3 square meters. If a panel that could supply an average 500 watts to the rover is required, the same power level as the RTGs, then an area of 22.7 square meters is required. All of these areas are for GaAs cells. Use of silicon cells, the area of the electrical collector will increase by 29 percent.

The mass of the power system, including both the electrical and thermal portions, ranges from a maximum of 253.4 kg for a system that provides 500 watts on a continuous basis, to 43.9 kg for a system that has power for normal rover operations during the day and is in sleep mode during the night. A system that provides for the baseline power has a mass of 126 to 111 kg for GaAs and silicon cells, respectively. For comparison, the mass of the proposed RTG system is 129 kg.

The overall conclusion is that a Mars solar rover is possible, but not easy. If the solar power system is required to duplicate the power output of the RTG system, then it will be large and heavy. If it is sized to supply the energy needed for typical rover operations, then it is of reasonable size and its mass is comparable to the RTG system. If an energy-saving sleep mode is used in place of the present idle mode, then the panel size becomes about equal to the size of the rover body, and the system mass is about half of the RTG system.

Section 8
RECOMMENDATIONS

The model used in this effort for the solar radiation on the surface of Mars had several simplifications. An improved model is needed. A new model should give the distribution of the scattered radiation about the sky, so that the amount incident on a solar panel in any orientation can be found. In addition, the changes to the solar spectrum due to the atmosphere and the dust should be taken into account in finding the photovoltaic cell performance. This is necessary as the efficiency of the cells is a function of the wavelength of light. For panels not oriented horizontally, the radiation scattered off of the ground needs to be considered.

An improved model for the temperature of the Martian atmosphere is needed. Such a model would improve the prediction for the efficiency of both solar thermal collectors and photovoltaic cells. This model should give the mean expected temperature for any time of day, season, and latitude.

The effective use of solar thermal collectors on Mars requires the use of selective surfaces. At present, no such surface is Mars qualified. Investigations of surfaces that can be qualified is needed. The surface must be able to withstand the effects of the Martian atmosphere, the dust, and the radiation.

The thermal control system of the rover could be greatly simplified or even eliminated by covering the rover with a selective surface. This would in effect make the entire rover a solar collector. Such an arrangement should be studied to determine if it is feasible and what problems need to be solved in its implementation.

Although this study has shown a solar-powered rover to be possible, the required panel size is quite large. The panel size can be reduced if the energy needs of the rover can be lowered. In the current rover most of the energy is used while it is in idle mode. A new "sleep" mode has been proposed that would reduce the average power needs of the rover by a factor of two or more, assuming it replaced the idle mode. The feasibility and development of this mode should be investigated.

The current configuration of the MRSR vehicle is not optimum for the collection of solar energy. Rovers that allow for better solar collector integration need to be investigated. Factors that need to be considered are the upper surface area of the rover body, placement of the cameras with respect to the solar panel, and placement of the antenna with respect to the solar panel.

Questions remain concerning the use of solar energy on Mars. The models for the solar radiation

distribution need better verification. Also, the effect of dust on the panel performance is not known. The long term degradation of cell performance due to radiation of other effects needs to be better understood for the case of operation on the Martian surface. These questions may best be addressed experimentally. A small probe placed on the surface of Mars could be used for this purpose. The probe would be little more than a solar panel and a transmitter. Every sol or so it would transmit data that gives the amount of energy collected by the panel as a function of time. This probe may weigh only a few kilograms (perhaps even less than one kilogram) and could be sufficiently rugged that it can be lowered to the Martian surface by parachute.

A somewhat more complex probe would be a small rover. Such a rover would be much smaller and simpler than the MRSR vehicle. It would carry no other instruments than a camera and the solar panel. The CARD method would be used for movement. This rover would not only allow for the examination of the usefulness of solar power on Mars, but would give some preliminary information about the surface conditions, as far as mobility is concerned.

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CONCEPTUAL STUDIES ON THE INTEGRATION OF NUCLEAR REACTOR SYSTEMS TO MANNED MARS ROVER MISSIONS

PART I

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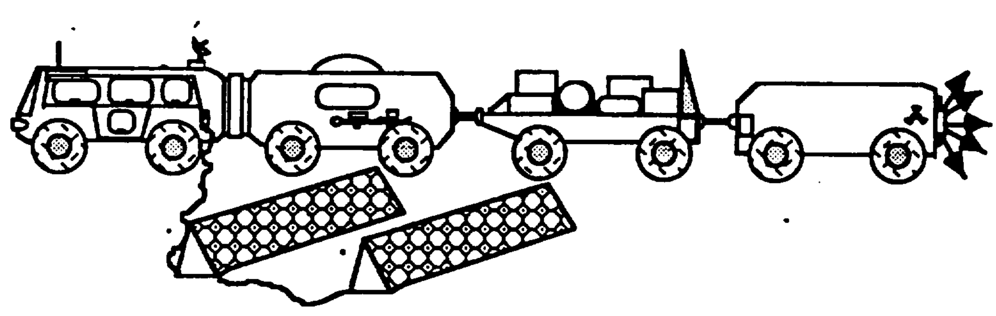


TABLE OF CONTENTS

List of Figures

LIST OF TABLES

1. INTRODUCTION

Throughout the next three decades the focus of the U.S. space program is being directed towards manned outposts in space. Through the Civil Space Leadership Initiative (CSLI), as currently being defined by NASA's Office of Exploration (OEXP), the successful completion of human exploration of the solar system will require system capabilities to evolve beyond their current status. Innovative designs of both spacecraft and surface elements will be required along with thoughtful approaches to meeting the long-term space power needs [Andrews, 1989].

For centuries the red planet has intrigued man. The nearest outward neighbor to the earth, Mars has been the focus of much speculation as the home for Alien life, although it is known now that the planet houses no intelligent life forms. The first successful probe to Mars was the Mariner 4, which on July 14, 1965 flew to within 10,000 kilometer of Mars. In 1969 two more Mariner missions were sent to fly by Mars in order to photograph the surface. Mariner 9, the final Mariner mission, was sent in 1971 to photograph the two martian moons Phobos and Deimos and preform a photographic survey of the Martian surface. This survey set the stage for the first two successful landing missions on Mars. On July 20, 1976 the Viking I lander touched down on the Chryse Plain on the surface of Mars. A month and a half later the Viking II lander also came to rest on the surface of our sister planet [Ezell and Ezell, 1984). Although both the Viking missions were highly successful, NASA has yet to revisit Mars; however, one of the main objectives of the CSLI is the return to Mars.

The first step in the return to Mars is an unmanned Mars Rover/Sample Return (MRSR) mission. The MRSR mission as it is now being planned would place two spacecraft in orbit about Mars and land a small (<1000 kg) robotic Rover to the surface of the planet. This Rover would collect up to 100 kg of soil, rocks, and atmospheric samples over a period of 11 to 18 months. These samples would in turn be transferred

to a accent vehicle for return to Earth. In addition to collecting the samples returned to earth, the Rover would be capable of performing additional experiments over the lifetime of its extended mission. The Solar System Exploration Committee (SSEC) of the NASA advisory Council is strongly recommending that a Mars Sample Return mission be undertaken before the end of the century [Bents, 1989]. This unmanned MRSR mission would then pave the way for manned exploration of our neighbor planet.

An evolutionary strategy is currently planned for manned missions to Mars. This strategy begins with an initial exploration of Mars and builds towards the establishment of an *in situ* propellant facility and ultimately a permanent manned base. Some of the mission design guidelines include an initial crew size of 4 growing to 8, reusability of selected vehicle elements, and aerobraking at both Mars and Earth. The program is to be carried out through a series of unmanned cargo flights and manned exploration missions [Andrews, 1989].

Multiyear civilian manned missions to explore the surface of Mars are thought by NASA to be possible early in the next century. Expeditions to Mars, as well as permanent bases, are envisioned to require enhanced piloted vehicles to conduct science and exploration. Piloted Rovers with extended range, have been identified as a viable means of achieving global access of Mars. For these mission, a Rover Vehicle with 30 kWe user net power is being considered. The operations covered by this power includes: drilling, sampling and sample analysis, onboard computer and computer instrumentation, vehicle thermal management, and astronaut life support systems. In addition to the 30 kWe user power, electric power will be needed to drive the Rover across the Martian terrain. Current technology in the area of space power generation yields several different options to supply the required energy to the Rover. These options include photovoltaic (PV) collection systems, radioisotope

thermoelectric generators (RTG's), Dynamic Isotope Power Systems (DIPS), High performance Fuel Cells, and Nuclear Reactor Power Systems. The following is a brief discussion on the merits of each of these systems.

Photovoltaic power systems have been flown by NASA since the late 1950's with the launch of the Vanguard satellite in 1958. These first cells, while only delivering milliwatts worth of power (used to operate a tracking oscillator), operated for over 6 years demonstrating the reliability the PV systems [Ralph, 1989]. The cells used for this mission were p on n silicon cells, which exhibited ~5% efficiency at air mass zero (AM0). After 1962 NASA switched to n on p type Silicon cells in order to improve the radiation resistance of the PV's. During the 60's the efficiency of PV cells was increased to 12% by the introduction of gridded front contacts. This substantial increase in efficiency was not to be the end, throughout the 1970's and 80's improvements continued to raise the efficiencies of Silicon Photovoltaic cells to their current level of 15% AM0 [Ralph, 1989]. Although efficiencies of 18.1% have been measured in the laboratory with simulated AM0 conditions, it is predicted that an efficiency of 22% can be obtained with the use of technics such as light trapping and surface passivation [Landis et al., 1989]. While the efficiency have increased over the years so has the resistance of the cells to radiation damage.

Although Silicon has been the primary material used for the construction of Photovoltaic cells in America's space program (due to the experience with the material, ease of manufacturing, and low cost of materials), several other materials are now under consideration. The use of GaAs as Photovoltaic cell material also has certain inherent advantages. ~~By using GaAs~~ typical efficiencies for single junction cells of ~19% are currently being seen. The maximum calculated efficiency for GaAs cells is 27.5%. By going to duel junction cells of GaAs on Ge substrait, efficiencies of up to

22% have been obtained [Ralph, 1989]. For these cascade cells, a maximum ~~calculated~~ efficiency of 35.7%, ~~is predicted~~ [Landis et al., 1989]. While Gallium Arsinide cells have been used for many years in terrestrial applications, production for the space program has only recently begun. The primary disadvantage of these cells is their high cost.

Another emerging technology in the area of space photovoltaics is the use of ultrathin films. Although there are several materials being researched, the primary candidates are amorphis Silicon (a-Si) and Copper Indium diselenide ($CuInSe_2$). Thin film cells have several inherent advantages in that they have a high radiation tolerance, high specific power (potentially in the kW/kg range), can be formed into flexible blankets, and have a large manufacturing experience. However, thin films are disadvantaged by lower efficiencies, lack of spacecraft experience, and the fact that they are not currently produced on lightweight substrates [Landis et al., 1989]. Although the second two disadvantages are easily overcome by the increase in experience with the cells, the maximum efficiency is limited by the physical properties of the materials. Laboratory efficiencies of 10% are now being produced for a-Si cells; however, by using multiple junction designs this could increase to grater than 15%, ~~in the future~~. Projected power-to-weight ratios for 10% cells should reach 350 W/kg AM0 in earth orbit [Ralph, 1989].

The experience and information relating to silicon PV Solar Cells is quite extensive. However, while photovoltaic Solar Cells have been shown to be extremely useful on orbital spacecraft, their effectiveness is directly dependent on the availability and intensity of the solar insolation. The solar insolation in turn is a function of the squared distance the cell is from the sun. As one travels further from the sun the available insolation decreases. Since the orbit of Mars places it ~1.5 times further from the sun than the earth is, the average solar insolation is only 590 W/m^2 compared to the 1371 W/m^2 for an earth orbit [Appelbaum and Flood, 1989]. If this value is used

with a projected efficiency of 30 %, the Rover's 30 kWe user power alone would require ~340 m^2 of panels. Although this area is already prohibitive it does not take into consideration the decrease in insolation due to atmospheric effects, nor include the energy loses due to storage for night consumption, and the power required by the Rover for mobility. It is easily seen that a PV system would be far too large to be practical in powering a Manned Mars Rover.

Radioisotope Thermoelectric Generators (RTG's) are another possibility as a power system for the Manned Rover. RTG's are extremely reliable long term static power supplies which operate on the thermoelectric energy (TE) conversion principle. Thermal energy is produced internally by the natural decay of Plutonium-238 ($t_{1/2}$=87.7 yr) and is carried across an array of TE couples and rejected by radiative heat transfer through external fins [Kelly,1987]. The temperature difference between the heat source and the fins is the driving force for the TE conversion [Angrist, 1976]. RTG's have been demonstrated to be safe and reliable with millions of hours of flight experience. The first RTG (SNAP 3B) launched in 1961 and was used to power the Navy's Transit 4a and 4b navigation satellites. Although this small unit only produced $4W_e$ beginning of mission (BOM) power, it successfully operated for over 6 years.

The next generation RTG's were developed under the SNAP program and showed a substantial increase in power and specific mass. The SNAP 19 and 27 RTG's were flown on the Viking and Apollo missions and produced a specific mass of $\leq$2.2 W_e/kg. The SNAP RTG's employed lead-telluride (Pb-Te) thermoelectric couples for energy conversion [Hartman, 1988]. With the Apollo RTGs the heat source and TE units were stored separately for the journey to the moon. Pioneer 10 and 11 are also each powered by 4 SNAP 19 RTG's. Pioneer 10 has past the mean orbit of Pluto and

should have sufficient power to transmit data through 1990 (18 years after being launched) [Skrabeck, 1987]. Again, these units operated in excess of the predicted performance. The Multi Hundred Watt (MHW) RTG was the next step in RTG design. These systems, which were used to supply power to DOD's LES 8 and 9 spacecraft and NASA's Voyager 1 and 2, yielded a specific mass of ~4.0 W_e/kg (BOM). The MHW RTG used Silicon Germanium thermocouples (unicouples) for energy conversion and modular Pu-238 fuel sphere packs for the heat source [Hartman, 1988].

The current generation of RTG uses the modular General Purpose Heat Source (GPHS) and unicouple thermoelectrics. The GPHS-RTG developed under sponsorship of the Department of Energy, has shown a specific mass of 5.3 W_e/kg in ground flight tested for both NASA's 4.2 year Galileo mission to Jupiter and the joint NASA/ESA 4.7 year Ulysses solar polar mission. The GPHS-RTG has a mass of 55.9 kg [Bennett, et al.].

Under current design is the Modular-RTG (MOD-RTG). This next generation RTG has the distinguishing feature of true modularity. By varying the number of modules assembled, power levels ranging from 20-342 W_e can be obtained. The heat source for the MOD-RTG is again the Pu-238 GPHS developed by DOE; however, the TE converters are SiGe/GaP multicouples (each multicouple contains 40 thermoelectric legs). MOD-RTG's are predicted to have specific masses up to 7.7 We/kg [Hartman, 1988].

Although RTG's are extremely reliable, they have some inherent drawbacks. First, the conversion efficiency of a static thermoelectric converter system is in the range of 4-6%. This implies that for the 30 kW_e specified user net power of the Rover, a total of 500 kW_t would be needed (assuming an optimistic 6% efficiency), requiring ~882 kg Pu-238. In addition to being in short supply at present, the use of Pu-238 in a Rover

RTG power supply is highly unlikely due to economical and safety considerations.

The other RTG drawback is that because the power produced by the radioactive decay of the Pu-238 is continuous, heat must be rejected during the journey to Mars. Also, the power level is always decreasing (~1.3% per year electrical power loss due to fuel decay) [Kelly, 1987]. For these reasons and the low specific mass of RTGs, they are not an appealing alternative for the Rover's Primary Power Source.

Dynamic Radioisotope Power System (DIPS) also utilizes an isotope heat sources, but are designed to produce power in the 1 to 10 kW_e range with potential growth up to 15 kW_e. Currently the DIPS program is being conducted under joint DOE/DOD sponsorship [Bennet, 1988]. In DIPS the Radioisotope heat source is coupled to a dynamic energy conversion system which allows for a much higher conversion efficiency than TE's. Two cycles are being investigated for use with DIPS. These are a Closed Brayton Cycle (CBC), being studied by Sunstrand, and an Organic Rankine Cycle (ORC) under investigation at Garrett [Pearson, 1988]. Schematic diagrams for the CBC and ORC are shown in Figures 1 and 2 [Bennett, et. al., 1988], respectively. So far system efficiencies of up to 3 times that obtained using Thermoelectric Generators have been demonstrated. However, currently projected specific mass [~150 kg/kWe] of the DIPS systems limit there usefulness in a Mars Rover [Angelo and Buden, 1985]. Also, DIPS are still plagued with the same difficulties as RTG's (power degradation and heat rejection during transit) due to their Radioisotope Heat Source.

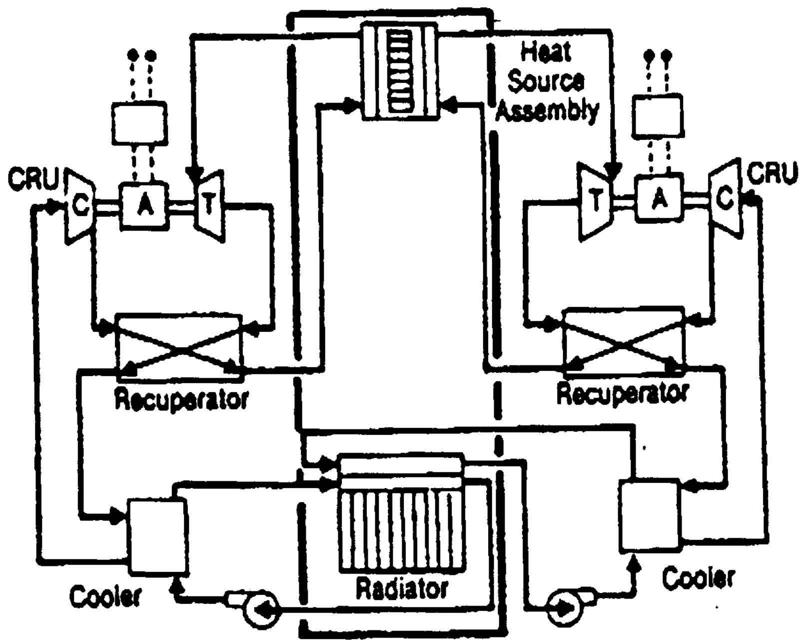


FIGURE 1: Schematic of Closed Brayton Cycle for DIPS [Bennett, et. al., 1988]

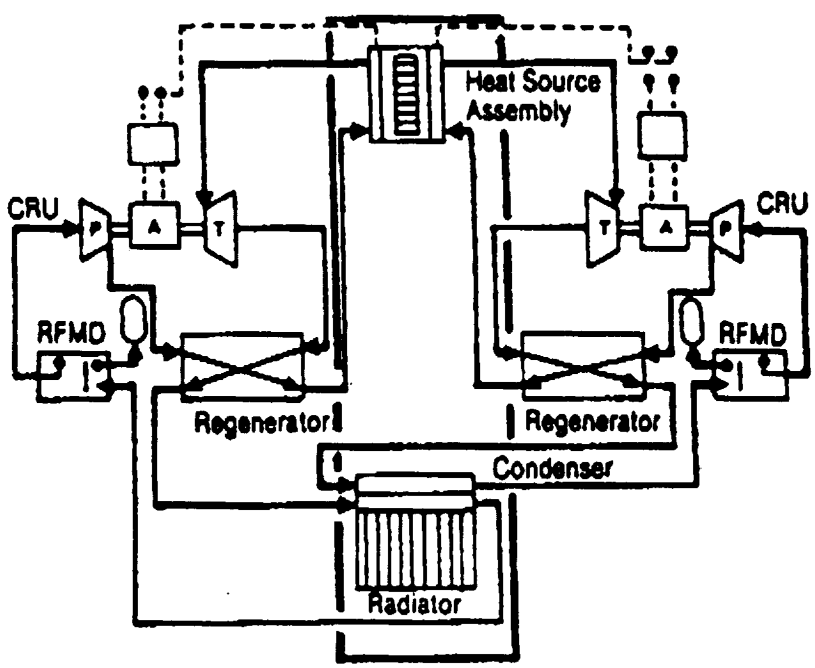


FIGURE 2: Schematic of Organic Rankine Cycle for DIPS [Bennett, et.al., 1988]

High Performance Fuel Cells such as Hydrogen-Oxygen Primary Fuel Cells are a very efficient way to store energy. The major draw back with these systems is that the energy is not being produced but instead merely stored. Thus as the mission length and the total energy storage increases the mass of the reactant storage becomes prohibitively large. Although Hydrogen/Oxygen fuel cells are not an acceptable primary energy source for the Rover, they will be considered as an auxiliary backup system in case of Primary Power System failure.

Nuclear Reactor Power Systems exhibits many highly desirable characteristics as a Primary Power Source for a Manned Mars Rover. First, the reactor core will produce virtually no heat until it has been and is brought up to power after safely landing on the surface of Mars. Second, the power output of the core can be controlled and adjusted to the level needed. Unlike PV systems, the power and size is not dependent on the distance from the sun and the rotation of the planet. Perhaps the most appealing feature is the low specific mass of the nuclear reactor power systems at high power levels. Current design efforts in the area of space reactors is focusing on the SP-100 reactor being designed by General Electric. Although this design yields a specific mass of approximately 40 kg/kWe, the reactor is equipped with a shadow shield with a cone angle of 34°. For a 2π -4π shield allowing human occupation of the surrounding area, the specific mass of the reactor system will increase significantly. Even with the shield requirements the reactor system is felt to be the best alternative to give the Rover sufficient power for global access.

However, in using a Nuclear Reactor to power a Manned Mars Rover many technical issues must be resolved in order to efficiently and effectively integrate the system into the Rover vehicle. These issues include minimization of shield mass and optimization of shield configuration, optimal integration of the heat rejection system

with the Rover configuration, thermal management, protection from the harsh Martian environment, and contingency systems. The objective of Parts I and II of this report is to answer the questions that arise from these issues. While Part I focused on the Rover design and assessment of power requirements for traversing, Part II will report the results of parametric analysis to: (a) identify the suitable conversion system/heat rejection combination, (b) the radiation shielding design and integration of nuclear reactor withe the Rover, and (c) mass optimization of the Man Manned Rover.

Although the focus of this research is primarily the feasibility of the design and integration of a Nuclear Reactor Power System into a Manned Mars Rover, a clear understanding of the mission requirements and a detailed descriptions of the Rover design are needed, in order that the an accurate estimate of the power requirements may be made. The next section deals with the mission requirements and Rover vehicle design, which the reactor system is to power

2. ROVER LAYOUT AND DESIGN

The integration of a Nuclear Reactor Power System with a Manned Mars Rover requires a detailed description of the design and functional requirements of the Rover. These requirements included:

(a) a 7 year operational lifetime for the Rover and power system with intermittent operation (months on/months off) requiring restart capability;

(b) a vehicle range which will allow for global access, user and housekeeping power of 30 kWe (including Life support for four shirt sleeve astronauts);

(c) the employment of SP-100 reactor technology with either static or dynamic energy conversion;

(d) heat rejection and material selection compatible with the martian environment and mission duration.

Although these design requirements specify the vehicle for the reactor design, a more detailed description of the Rover is needed in order to determine the mobility power requirements, and hence, determine the electric power output of the nuclear reactor system.

Currently, there is little information on the design of a multi-purpose global access rover for Mars; however, extensive investigations on moon based rovers have been completed [Bekker, 1969;EEI, 1988]. Since many of the design parameters for a lunar rover are similar to thoughs of a Martian vehicle many parallels may be drawn.

The first step in understanding the characteristics of the rover vehicle is to define the design capabilities and requirements of the rover. However, the Martian scenario exhibits several qualities which are unique unto itself. Three major considerations for the Martian Rover are: (1) The extreme distance from the Earth demands the lowest specific mass in order to minimize the launch cost; (2) The rover power system should contain the highest degree of reliability and redundancy with capabilities for an

adequate auxiliary power system in case of a complete malfunction of the primary power system, (3) The rover need to be compatible with a Martian atmosphere composed primarily of CO_2, which can be life limiting to many materials.

There are many general configurations for planetary surface rovers. Associated with these designs are many parameter which can affect the mobility power requirements. The soil/vehicle interaction is the most critical power variable. Different traction types including wheels, tracks, and walkers have been studied. In the area of wheels alone there are several options available including rigid, pneumatic tires, wire mesh tires, elliptical wheels, hemispherical (and cone) wheels, and hubbless wheels [Bekker, 1969].

Another unknown for determining the mobility power requirements is the actual configuration of the vehicle, and hence the mass of the rover. The layout of the rover could vary from a large single vehicle, articulated crank mounted wheeled vehicle, to a multi-car overland train [Eagle Engineering Inc., 1988]. Since the Power requirements call for a nuclear reactor power system, it is important that the radiation sensitive equipment and the crew be kept at a sufficient distance from the reactor. For these reasons the multi-car overland train rover option was chosen. In this case, the crew and radiation sensitive equipment are located in the forward cars, while the radiation insensitive equipment and supplies can be stored in the rear of the train. Also, to minimize the radiation shield mass, the reactor would be housed as close as possible to the back end of the last car to maximize the distance between reactor and crew.

The proposed configuration for the Manned Mars Rover is shown in figure 1. It consists of four units: the Primary Control Vehicle (PCV), the Experimental Unit (EU), the Supply Car (SC), and the Reactor Car (RC). The function and the component mass breakdown for each of the units are discussed separately in the following sections.

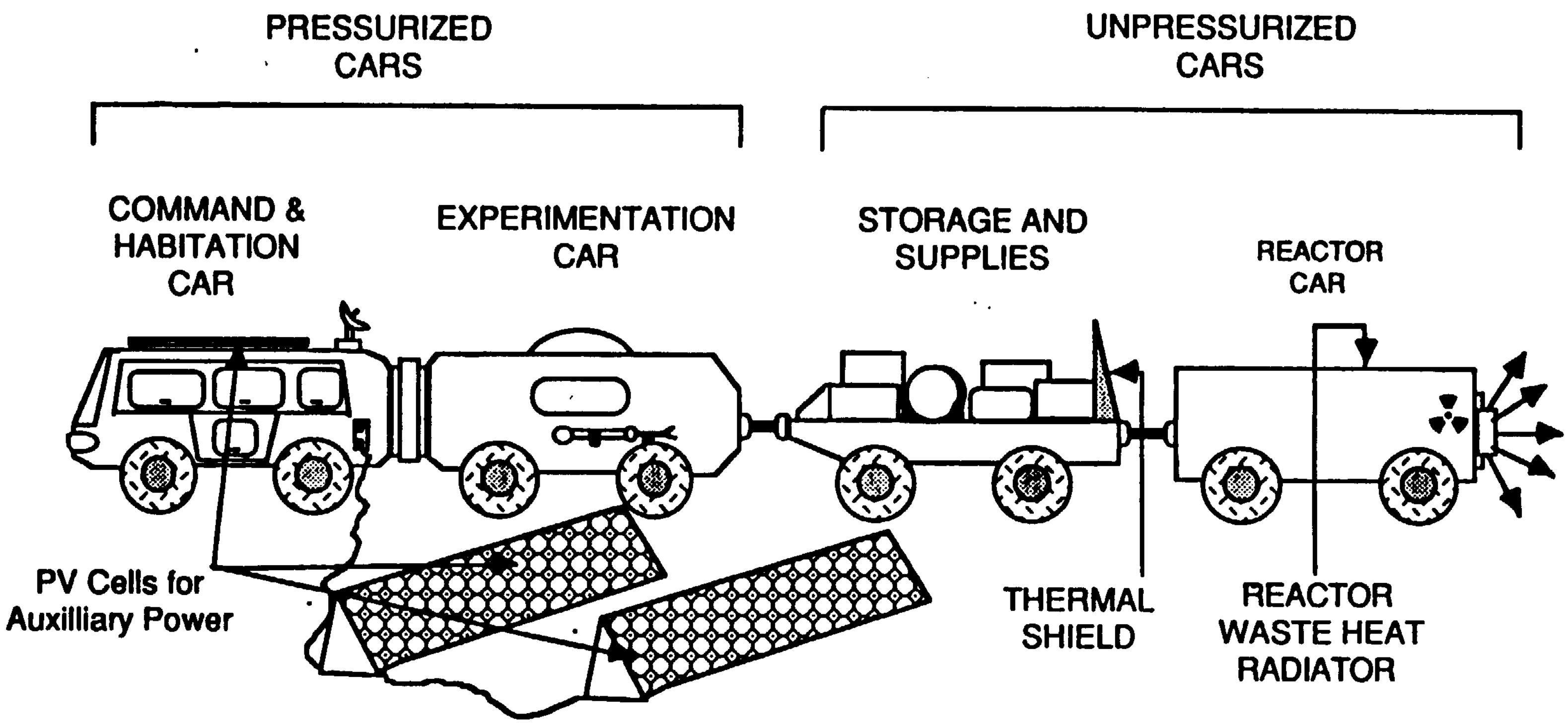


FIGURE 3: LAYOUT FOR TRAIN TYPE MANNED MARS ROVER

2.1. Primary Control Vehicle

The Primary Control Vehicle (PCV) is a double wall, pressurized, climate controlled car. The PCV houses the driving, navigation, and general control systems for the rover vehicle. The basic car design accommodates 4 astronauts. The PCV also includes the sleeping quarters, galley, a work station, life support systems and an auxiliary power system. The basic exterior dimensions of the vehicle are 3x3x10 meters, which give an interior volume of approximately 80 m^3. Figure 4 shows a generalized configuration for the PCV.

In the case of Primary Power System (PPS) failure the immediate concern is the survival of the rover crew. Due to the extreme distance from the earth it is essential that the rover be equipped with an emergency power system, to maintain life support systems and a degree of Rover mobility for several days, in the event of a PPS failure . For this purpose a hybrid PV/Regenerative Fuel Cell power system is employed, see figure 5 (a and b). Figure 5a illustrates the system configuration during collection . Here the PV's not only supply energy for the PCV, but also to electrolyze H_2O into H_2 and O_2 for the Regenerative Fuel Cells (RFC's). In figure 5b the Night configuration is shown. Here, no energy is being collected by the PV's and all of the Rover's power is supplied by the RFC's.

Since the primary concern in the event of a PPS failure is shifted from the experimental objectives to the survival of the crew, all nonessential equipment and supplies are to be jettisoned. The scenario call for disconnecting the PCV from the Experimental Unit, Supply Car, and the Reactor Car. The PCV is equipped with up to three days worth of life support power stored in the Hydrogen/Oxygen RFC. The total mass of the fuel cell system including reactant is 720 kg. This figure is based on a crew of 4 astronauts, a life support power of 1 kW_e/astronaut, and a system specific

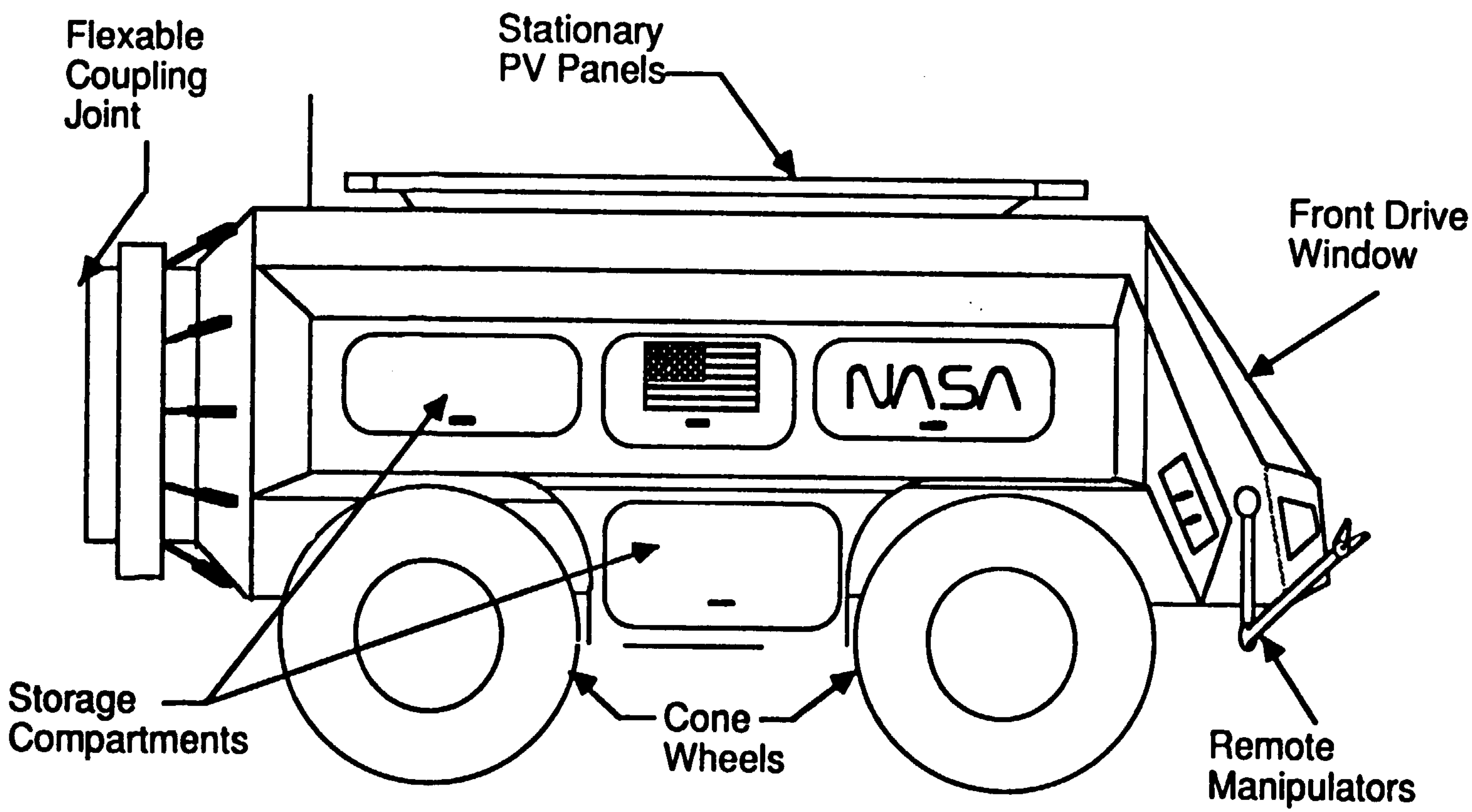
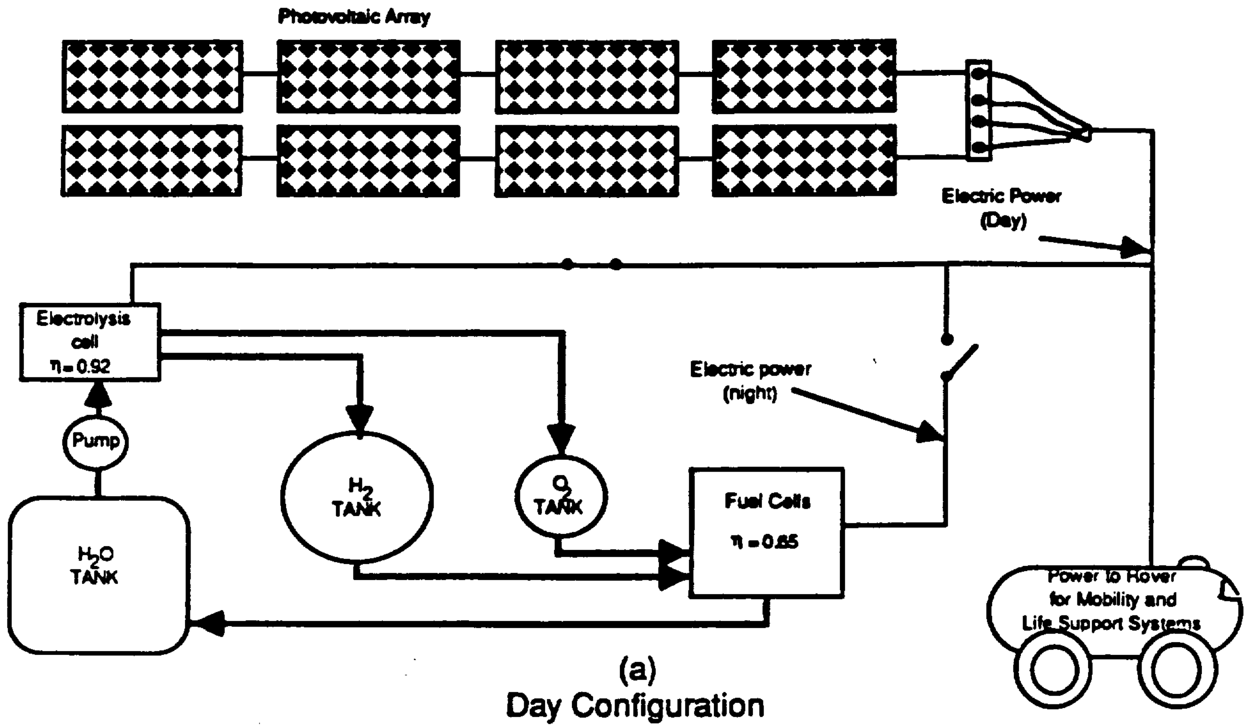
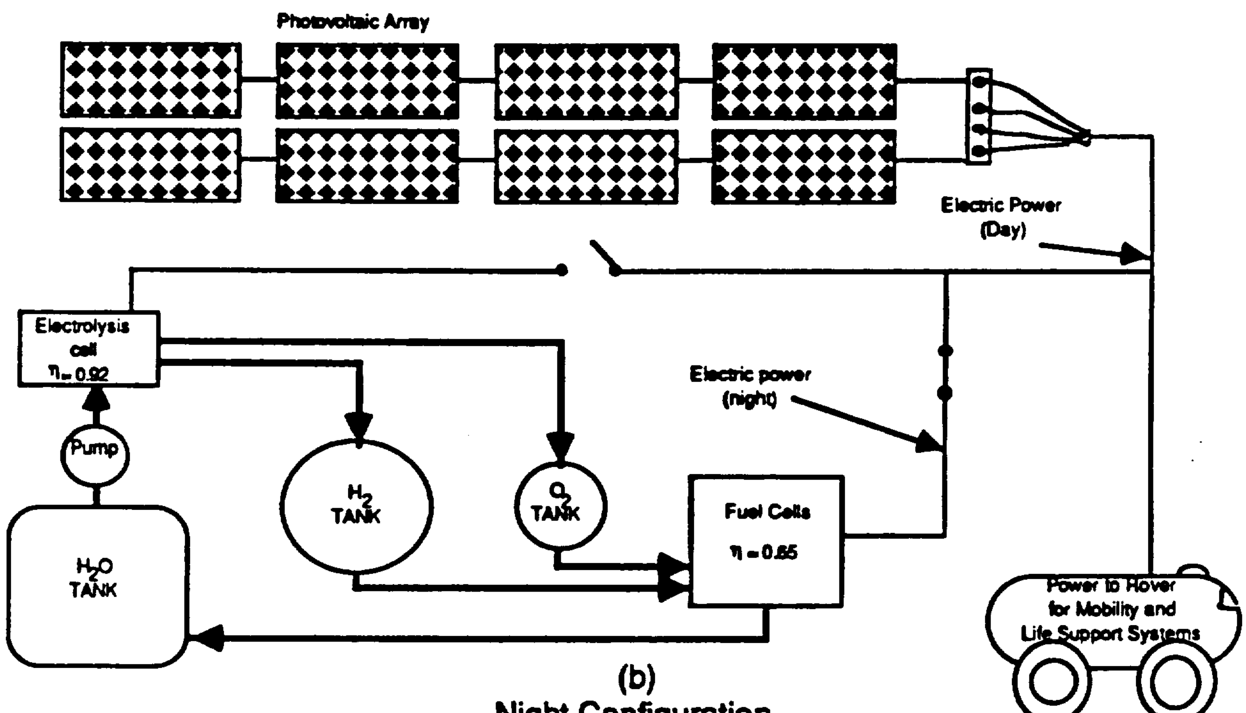


FIGURE 4 : Layout for Primary Control Vehicle



(a)
Day Configuration



(b)
Night Configuration

Figure 5: Auxiliary Power System Configuration

energy of 400 W_e/kg [Cataldo, 1989]. For this storage a total reactant mass of 107 kg is required (12 kg H_2 and 95 kg O_2). For reactant storage tanks pressurized to ~2MPa [Angrist, 1976] a total storage volume of 9.5 m^3 is necessary. Since the RFC's are sufficient for three day of life support power, the astronauts would have sufficient time to arrange for a rescue mission launched from a Mars Base or orbiting spacecraft.

2.2 Mass Estimates of Auxiliary PV Panels

In addition to the Fuel Cells, the Auxiliary system will include a set of Photovoltaic (PV) arrays in order to recharge the fuel cells on a daily basis and provide for both life support and mobility power. The mass of the PV is a function of the total mass of the PCV (this includes the PV array itself), the driving scenario (daily range and cruising speed), and the efficiency of the RFC cycle.

The PV panels mass can be given in a general form by:

$$M_{pv} = P_{aux} \cdot SM_{pv,} \qquad (1)$$

Where P_{aux} is the auxilliary power and SM_{pv} is the specific mass (kg/kWe) of the pv panels. The auxiliary power required from the PV's can be divided into power required for life support, P_{ls}, and power required for mobility, P_m. While the life support power is constant for a given number of astronauts, the power requirements for mobility will be dependent on the terrain, emergency cruising speed, daily range, and the total mass of the PCV according to the following equation:

$$P_m = \chi(8.0x10^{-5} \text{ kW}\cdot\text{hr/km}\cdot\text{kg})M_{tpcv}\cdot V. \qquad (2)$$

The value of $8.0x10^{-5}$ kW·hr/km·kg for the soil traction coefficients is based on the average energy consumption of the Apollo Lunar Rovers [Eagle Engineering Inc., 1988]. The traction factor modifier χ is a correction factor to account for the difference in soil conditions and gravitational acceleration from the Moon to Mars. Although the calculation seem straight forward at first it is important to note that the PCV mass also

includes not only the values shown in table I, but also the mass of the PV array.

TABLE 1: Primary Control Vehicle Mass Breakdown
Based on 4 Astronauts
(From EEI Report 88-188)

	Mass (kg)
Structure and Pressure Vessel	
Inner Shell	490
Outer Shell	500
Other Structures	200
Insulation	130
Galley	70[a]
Personal Hygiene	90[a]
Emergency Equipment	30
Man-Locks	230
EMUs	680[a]
Avionics	90
Environment Control and Life Support	200[a]
Workstation	40
Drive Stations	80
Sleep Quarters	500[a]
Experiments and Payload	500
Crew	360[a]
Active Thermal System	
Radiator	160
Pump	20
Heat Exchanger	50
Piping	100
Refrigerant	300
Wheels and Locomotion	300
Fuel Cells	740
Total	5860

[a]Values will vary depending on the number of astronauts

The mass for the the PV panels, which is a function of several parameter, can be expressed as:

$$M_{pv} = \frac{n_a \cdot p_{ls}[t_c+(t_d-t_c)/\eta_{fc}]}{t_c} + \chi(8.0\text{x}10^{-5}\text{kW}\cdot\text{hr/km}\cdot\text{kg})(M_{pcv}+M_{pv})V(t_m)/\, t_c\eta_{fc} \quad (3a)$$

and by rearranging:

$$M_{pv} = \frac{n_a \cdot p_{ls}[t_c+(t_d-t_c)/\eta_{fc}] + \chi(8.0\text{x}10^{-5}\text{kW}\cdot\text{hr/km}\cdot\text{kg})M_{pcv}\cdot V(t_m/\eta_{fc})}{[t_c - \chi(8.0\text{x}10^{-5}\text{kW}\cdot\text{hr/km}\cdot\text{kg})V(t_m/\eta_{fc})SM_{pv}]} \quad (3b)$$

Equation (3b) gives the mass of the PV panels in terms of the collection time, t_c, the mobility time during both the day and night (t_m), the number of astronauts (n_a), the cruising speed (V), the traction factor modifier (χ), and the specific mass of the PV's (SM_{pv}). The total mass of the PCV then becomes the sum of the tabulated masses in table I and that calculated for the PVs by equation (3b).

Determining the total PCV mass then becomes an issue of accurately estimating the PV parameters and identifying the sensitivity with which each of these parameters affect the PV mass. For the base case scenario the mass of the PCV and the number of astronauts have already been set at 5860kg (table I) and 4, respectively. Since the gravitational acceleration on Mars is approximately twice that of the Moon a value 2.0 is used for χ. This value is comparable to the estimated values for the rolling resistance of the Martian soil of 0.35 and a Rover drive efficiency of 50% [Aerovirnment, 1989]. Although the value of 2.0 for χ is 15% lower than that suggested by AeroVirnment (1989), it is still considered a reasonable value.

The time parameters including PV collection (t_c) and Rover mobility (t_m) are dependent on the driving scenario assumed; however, for the base case a collection time of 12.3 hr (12.3hrs is the number sun light hours at 0 degrees latitude during both the Martian Aphelion and Perihelion) and a mobility time of 6 hours were used. The

discussion on the selection of these base case values will be presented later in this report. Although the cruising speed for the emergency return, like the mobility and collection times, will be determined by the specific return scenario, a maximum speed of 5 km/hr was assumed. An efficiency of 60% was used for the RFC system [Cataldo, 1989]. As for the final parameter, the Specific Mass of the PVs, it was calculated using an average surface solar insolation, G, on the planet's surface of 250 W/m^2, a PV efficiency, η_{pv}, of 20%, and a panel surface density, SMA, of 2.3 kg/m^2 as:

$$SM_{pv} = SMA/(G \cdot \eta_{pv}/1000\ W/kW). \tag{4}$$

The base case values for the variable in equations 3b and 4 are listed in table II.

TABLE II: BASE CASE AUXILIARY POWER SYSTEM VARIABLES

Photovoltaic panel efficiency	η_{pv}	= 20%
Average solar insolation	G	= 250 W/m^2
Specific mass area of PV panels	SMA	= 2.3 kg/m^2
Fuel cell reserve life support energy storage (PCV)	t_{ap}	= 72 h
Fuel cell reserve life support energy storage (EU)	t_{ae}	= 72 h
Fuel cell reserve life support energy storage (SC)	t_{as}	= 24 h
Emergency return speed	auxspd	= 5 km/h
Power required for life support	P_{ls}	= 1kW/astro.
PV collection time	t_c	= 12.3h
Day Mobility	t_{md}	= 0 h
Night Mobility	t_{mn}	= 6 h
Efficiency of the fuel cell cycle	η_{fc}	= 0.6
Specific mass of the fuel cells	SM_{fc}	= 2.5 kg/kw·h

From the Base case parameters, PV panels having a mass of 667 kg are required to power the PCV during the emergency return. This corresponds to a PV panel area of 290 m^2. If the panels were dimensioned such that they were 10 m wide (the same as the length of the PCV), a 29 m length of PV panels would be needed. While this is

not felt to be impossible to accomplish, a substantial increase in panel size would limit the auxiliary power system's usefulness to supply power for an emergency return trip. It is, therefore, important to evaluate the effects of reasonable variation in the base case parameters on the mass and area of the PV array.

In order to minimize the PV mass it is important that the collection time be maximized and the mobility scenario be optimized for minimum power requirement. Therefore, it is assumed that energy collection takes place for 12.3 hr, representing all daylight hours, during which the PV panels are deployed and the PCV is not moving. Table III list the dark periods (in hours) at several Northern latitudes for both the Aphelion and Perihelion planetary positions [Kaplan, 1988]. Given 24.62 hr as the length of a Mars Sidereal day and an equatorial location, the collection time can be determined by:

$$t_c = 12.31 - t_{md} \tag{5}$$

When determining the total mass of the PV panels it is important to consider the advantages and disadvantages of choosing a particular emergency return mobility scenario. The operating region for equatorial deployment is shown in figure 6. The four curves presented in this figure represent PV mass values for the base case parameters given in table II. The effect, upon the PV mass, of varying the collection time (or mobility during the day) and the mobility during the night, t_{mn}, is presented. Starting from the base case of 6 hours of night mobility (@5 km/hr this represents a daily travel of 30 km/day) and a 12.3 hr collection time (0 hr day mobility) a Photovoltaic mass of 667 kg is required. It is recognized that driving during the night can place additional constraints on the Rover crew, but it reduces the mass of the PV panels, therefore, the merits of nighttime travel are discussed.

Keeping the travel time constant at 6 hr, a PV mass of 1500 kg is needed if traveling is limited to daylight hours. This increase from 667 kg to 1500 kg (an

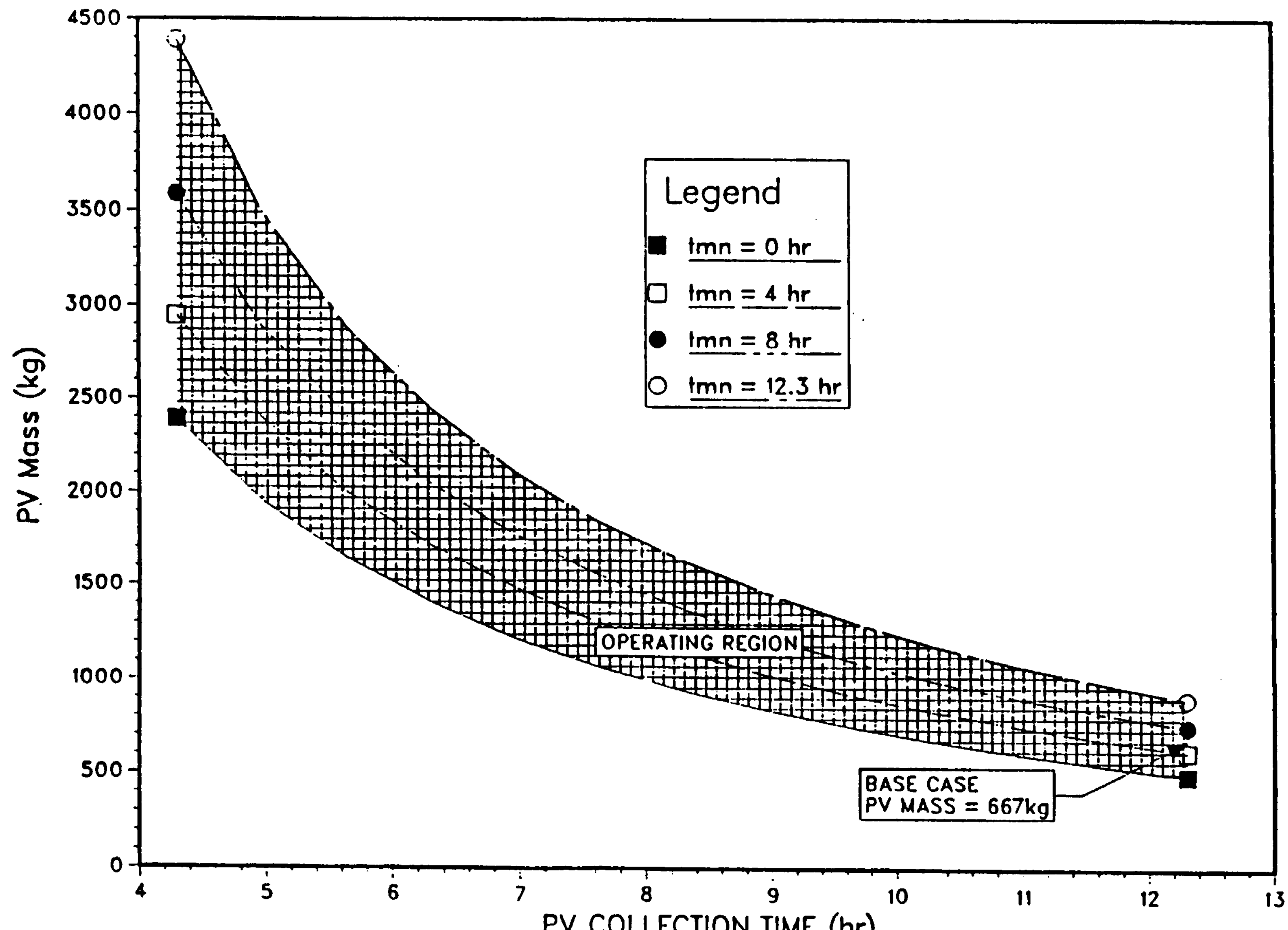


FIGURE 6: Operating Regime for Auxiliary Power System

TABLE III: MARTIAN SEASONAL INFORMATION

Northern Hemisphere Summer (Aphelion)

Latitude (degrees)	Mean Radius (km)	Ratio (night/day)	Dark Period (hours)
0	3393.0	1.00	12.31
9	3351.2	0.910	11.73
21	3167.6	0.793	10.89
33	2845.6	0.670	9.88
44	2440.7	0.538	8.61
55	1946.1	0.361	6.54
65	1433.9	0.000	0.00

Northern Hemisphere Winter (Perihelion)

Latitude (degrees)	Mean Radius (km)	Ratio (night/day)	Dark Period (hours)
0	3393.0	1.00	12.31
9	3351.2	1.10	12.90
21	3167.6	1.262	13.74
33	2845.6	1.493	14.75
44	2440.7	1.858	16.01
55	1946.1	2.768	18.09
65	1433.9	undefined	24.62

than twice the base case mass) is due to the shortened collection time for the PV panels, compounded by the increased storage and consequent energy loss during energy storage and recovery from the fuel cells. In addition to the mass increase, the size of the array becomes prohibitively large. For example by traveling during the day, instead of night, the 290 m^2 surface area will more than doubled to 652 m^2. This area will increase the array length from 29 m to over 65 m, for a width of 10 meters. It is therefore recognized, that in order to effectively reduce the size and mass of the PV, night travel during an emergency return should be considered.

When the collection time is held constant as in the base case, and the night mobility time is increased to 12.3 hr (this assumes instantaneous deployment of the PV's; Not likely), the PV mass increases to 900 kg. This value is still considerably less than the 1500 kg for only 6 hours of day travel. Had the additional 6.3 hr been driven during the day instead, then a PV mass of ~2000 kg will be needed, requiring a total of 870 m^2 of PVs, or 3 times that needed for the base case.

Figure 7 presents the PV mass optimization surface in terms of nighttime travel and daytime collection (or daytime travel = 12.3hr-t_c). It is apparent that the lowest PV mass requirement is that corresponding to 0 hrs of night travel and 12.3 hrs collection time (0 hrs day mobility). This PV mass of 490 kg corresponds to the PV panels needed only to maintain life support. By traveling along the night mobility axis to 6 hr, the base case in table II is again reached. It is important to note that by increasing the night travel from 0 to 12.3 hr, the PV mass is effectively doubled, while increasing the day travel from 0 hr to 8 hrs, increases the mass of the PV panels by a factor of 5. For this reason the night driving scenario is preferred.

To illustrate the typical duty cycle that can be used, a log of daily activities is presented in figure 8 for the base case scenario (see table II). This diagram shows the activities preformed for a full Martian day, during an emergency return. It is assumed that the RFC's are fully charged (three days life support power) at sunset. One full hour each is devoted to folding and deploying of the PV panels. Although the actual time needed for these activities will depend on the amount of automation, an hour was determined to be more than sufficient for each activity, should the automatic deployment mechanisms fail. The schedule also allows for three meals and 8 hours sleep each day. The 6 hr driving scenario allows for a total traveling distance of 30 km/day. It is important to reemphasize that in the scenario delineated in figure 6 the entire 12.3 hr of daylight is used for solar energy collection. The daylight hours

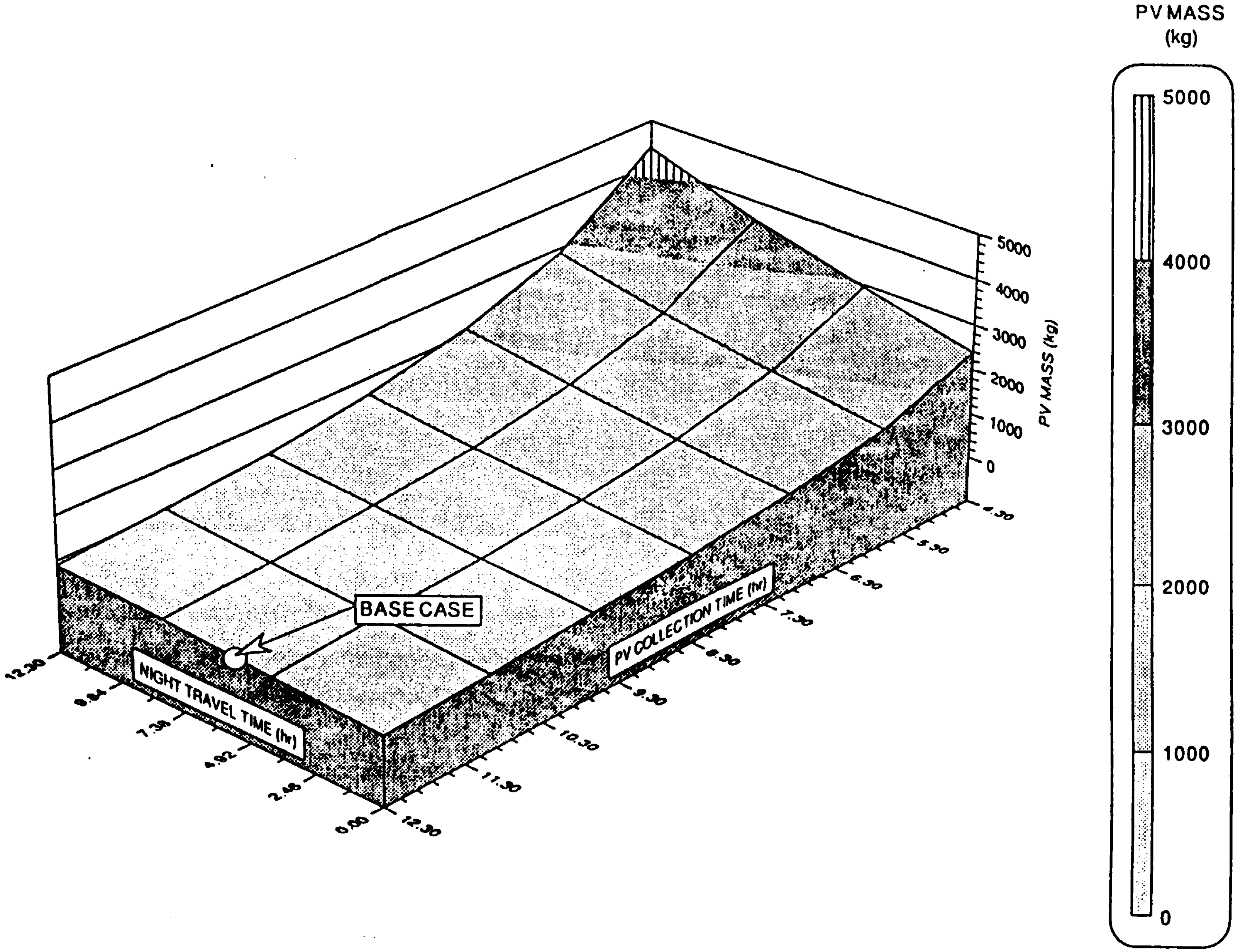


FIGURE 7: Auxiliary Power Systems Operating Surface

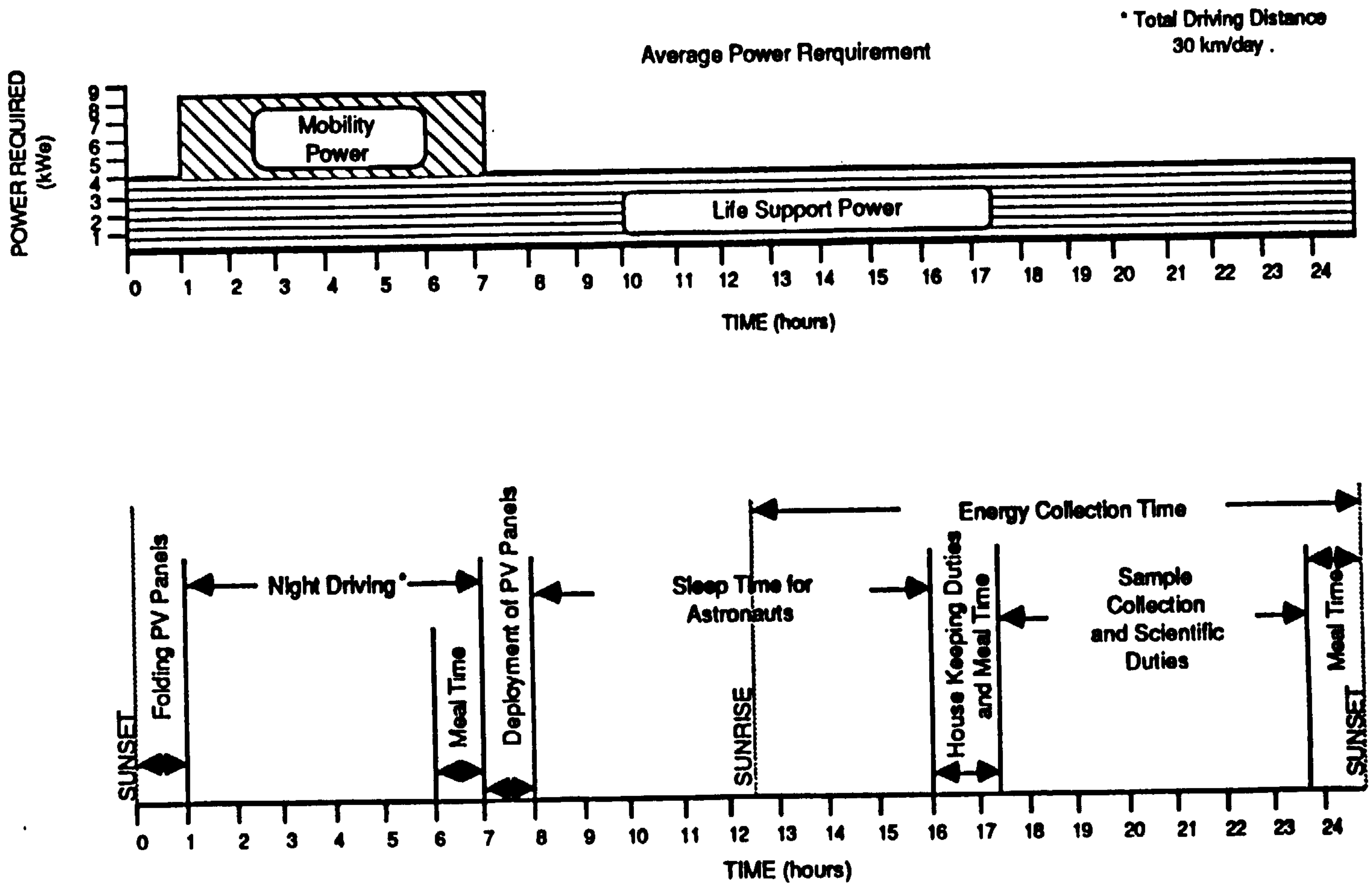


FIGURE 8: Base Case Mission Scenario for Emergency Return @ 5km/hr

during which the astronauts are awake, are designated for sample collection and scientific duties. The power utilization curve shown above the duty cycle represents the average power requirements based on life support power of 1 kW_e/astro· and a maximum cruising speed of 5 km/hr. In the case where daily range must be increased, the night driving time can be increased from 6 to 10.3 hr. For this scenario the total PV mass becomes 820 kg. This 20% increase in mass of the PV panels over the base case mass will increase the daily travel distance by 70% (from 30 km/hr to 51 km/day). For a 1000 km return trip only 25 days will be required as compared with the 33 days for the base case. A possible duty cycle for the 10.3 hr daily driving time is shown in figure 9. Here the time allocated to scientific duties is decreased from 6 hrs to 2.7 hrs in order to free up time for travel. In figure 7, the average power utilization of the longer driving time scenario is shown above the duty cycle.

Figure 10 is a plot of mass contours of the PV mass required as a function of both the collection time and the nighttime travel. Similar to the surface in figure 7, this plot allows for all the night/day driving combinations to be examined for given a mass requirement for the PV panels. In figure 10 the minimum night driving scenario, maximum night driving scenario, and the base case scenario masses are all labeled with their respective locations on the contour. The lowest PV mass of 490 kg, represents the mass of PV panels needed to only supply sufficient energy to run the life support systems.

2.3 Parametric Analysis of the Effect of Design Parameters on the Auxiliary PV Mass

Because of the conceptual nature of the Rover design the exact values for the parameters to determine the auxiliary power system mass are unknown. Even

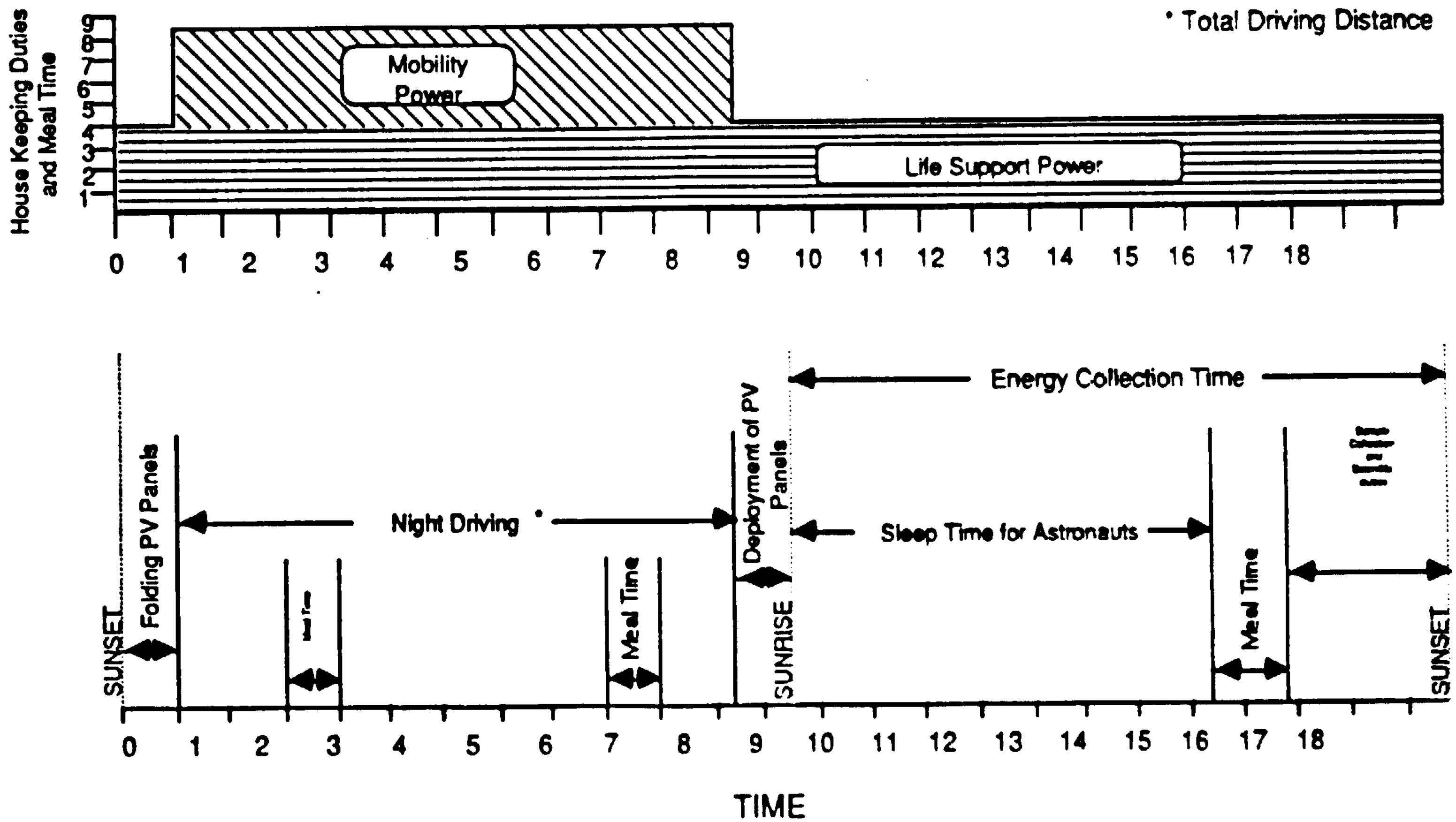


FIGURE 9: Maximum Night Travel Mission Scenario for Emergency Return @ 5km/hr

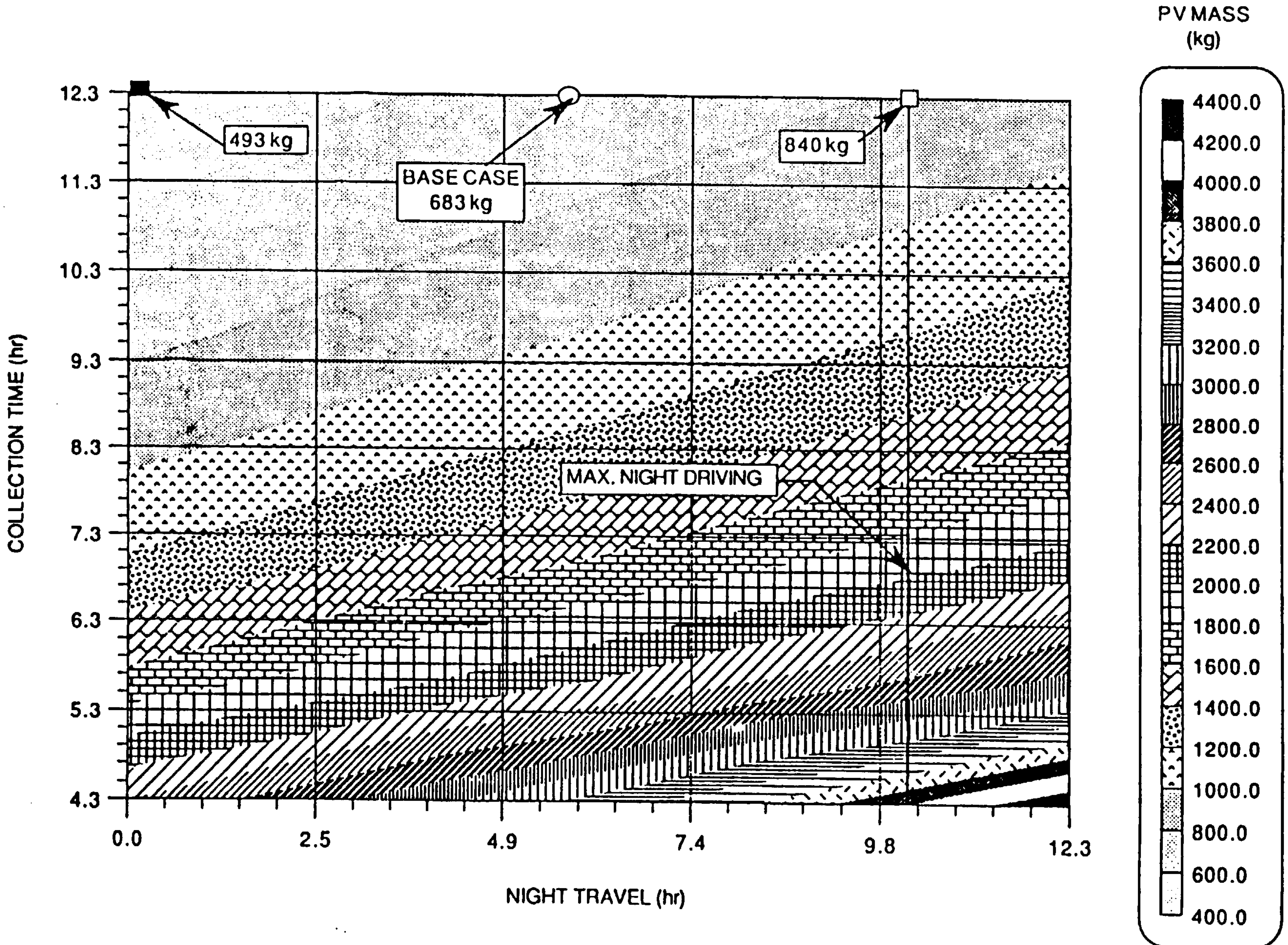


FIGURE 10: PV Mass Contours

though care has been taken to use values judged to be reasonable, given current technology and expected advances, a degree of uncertainty in the results is still introduced. For this reason a parametric analysis was preformed to examine the sensitivity of the mass of the auxiliary power system to various operation and design parameters in equation 3b. These parameters include cruising speed, number of astronauts, life support power per astronaut, average PV specific power, efficiency and specific mass of the PV panels, and the amount of fuel cell reactants stored. The result of the parametric analysis are presented here and discussed in the following subsections.

2.3.1 Effect of Cruising Speed

The most important variable in equation 3b is the Primary Control Vehicle's cruising speed. While reducing the cruising speed reduces the mobility power requirements for a given traveling distance, it increases the duration of the return trip. Therefore, the appropriate cruising speed should be selected based on considerations of power requirements and traveling range. In figure 11 the mass of the PV panels is plotted against the the PCV cruising speed with the base case identified on the 1 kW_e/astro life support curve. The point on this curve corresponding to 0 km/hr cruising speed, again shows the PV mass required to only maintain life support (490 kg). At a cruising speed of 10 km/hr, the PV mass increases by 350 kg to 840 kg. These results suggest that the mass of the auxiliary power system will be limited more by the number of astronauts on board than by the cruising speed. This mass will also be strongly impacted by the life support power requirements.

2.3.2 Effect of Life Support Power Requirements

As figure 11 shows, maintaining a constant cruising speed of 5 km/hr, and doubling

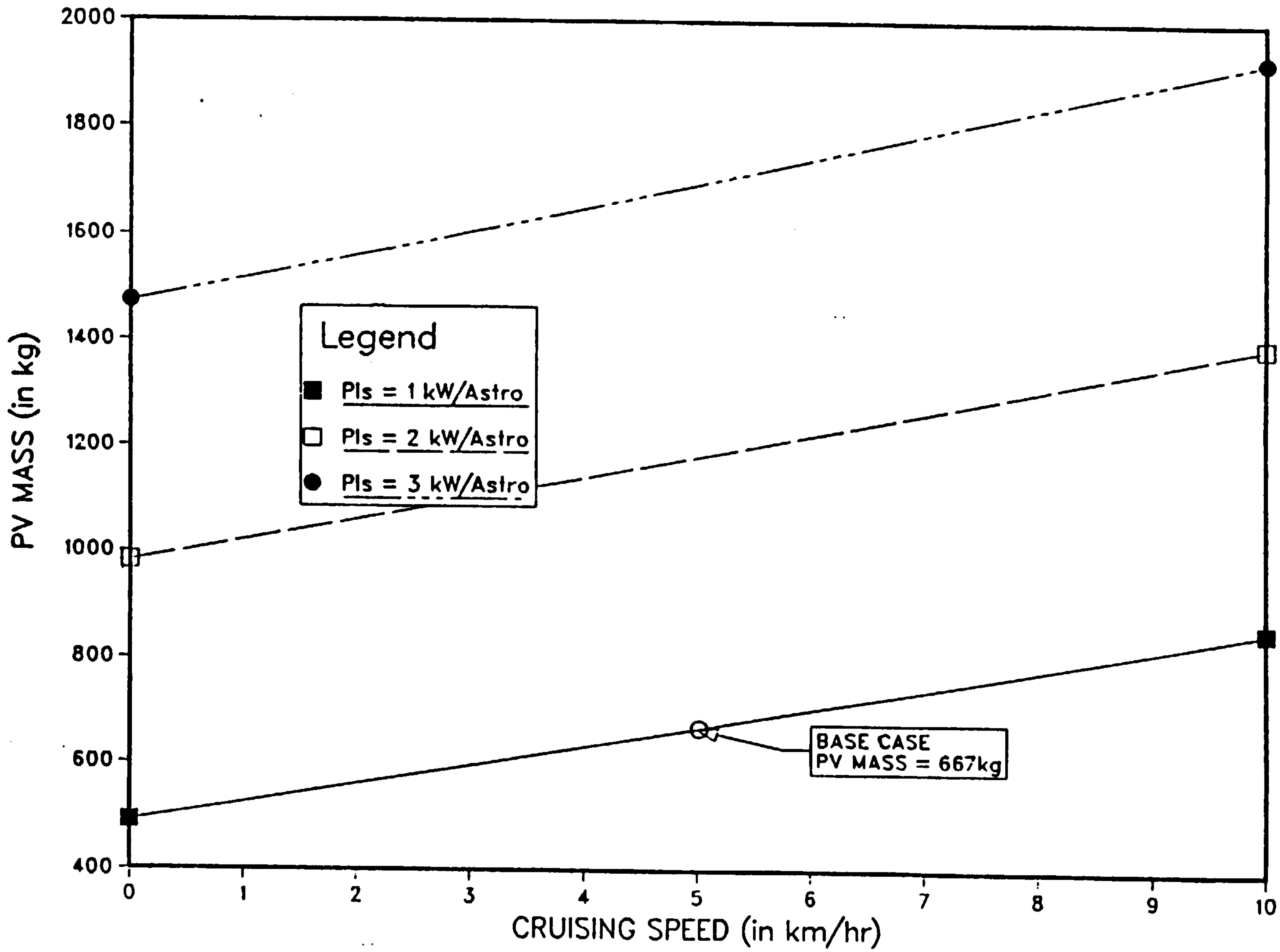


FIGURE 11: Effect of Life Support Power on PV Mass

the life support power to 2 kW_e, increases the PV mass from 667 kg to just under 1200 kg. The strong dependance of the PV mass on the life support power is due to two considerations. First, unlike the cruising speed the life support power is continuous throughout the day. Secondly, for a crew of four astronauts doubling the life support power per astronaut increases the total life support power requirement by a factor of 4. Further increase of the life support power to 3 kW_e/astro will bring the PV mass requirement to nearly 1700 kg for the base case. As demonstrated in figure 11, an auxiliary power system that is designed for 1 kW_e/astro at a cruising speed of 10 km/hr would not be able to deliver sufficient power to maintain the life support at 2 kWe/astro, even, without mobility. These results suggests that great care should be taken to minimize the life support power requirements, since an inflated estimate could excessively increase the mass of the auxiliary power system for the rover.

2.3.3 Effect of PV Specific Power

In figure 12 the PV mass is plotted against the PCV cruising speed for different values of the PV specific mass. The specific power values in figure 12 are simply the average solar insolation (250 W/m^2) multiplied by the efficiency of the array (from 20 to 30%). As figure 12 shows lowering the specific power of the PV panels increases the rate at which the PV mass increases with cruising speed. This rate increase is caused by the fact that PV panels with lower specific power require a larger collection area of the PV panels, resulting in a larger mass. In turn, this increased mass requires more power to move. In figure 12, beginning at the base case, increasing the specific power to 100 W/m^2 brings the PV mass down to 300 kg. Conversely, decreasing the PV specific power to 20 W/m^2 raises the PV mass to as much as 1750 kg. The sensitivity of the PV mass to its specific power is much greater than that of the cruising speed at high specific power values (>50 W/m^2); however, as the specific power decreases the

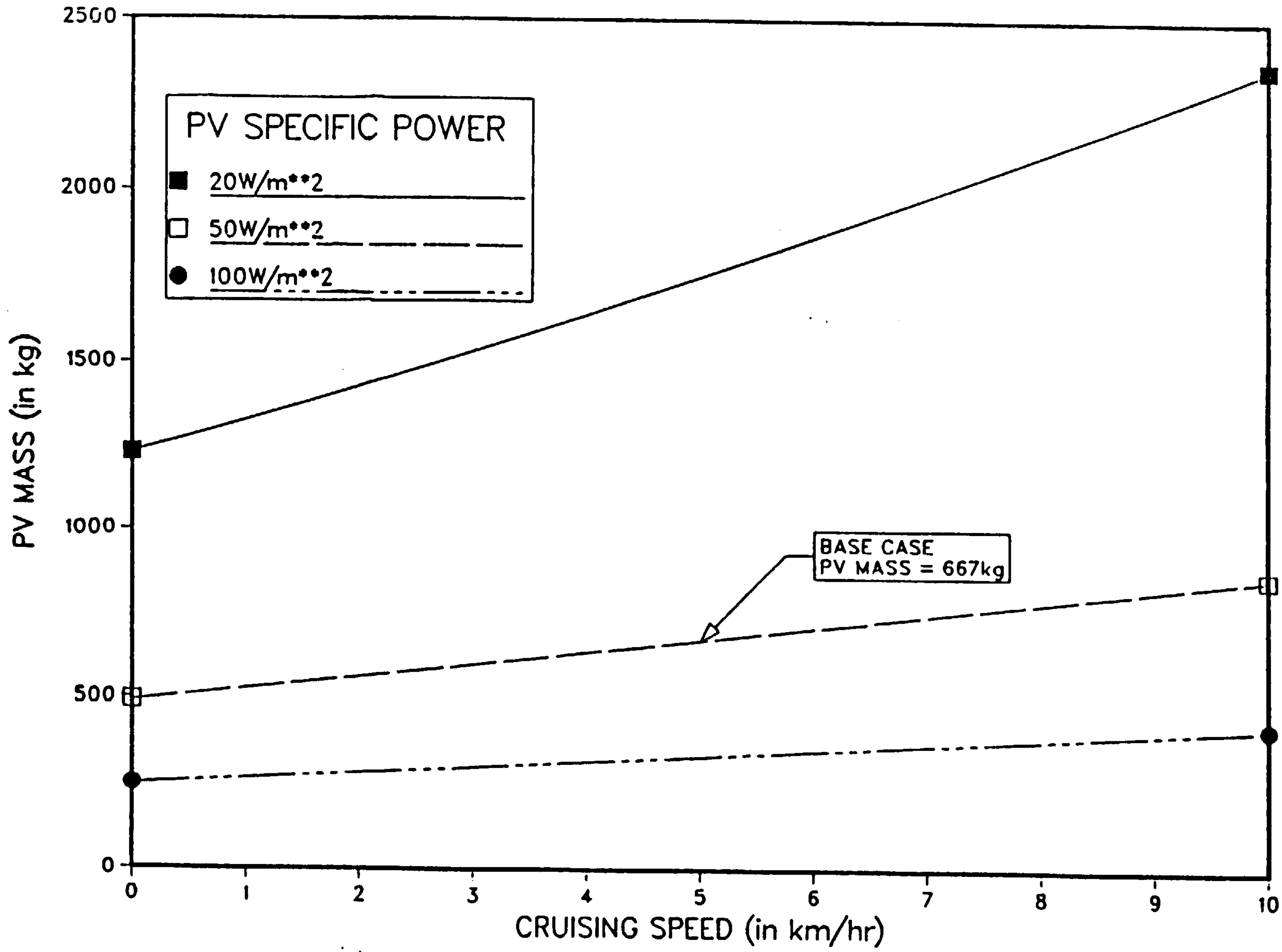


FIGURE 12: Effect of PV Specific Power on PV Mass

sensitivity of the PV mass to the cruising speed increases drastically. These results suggest that the savings in the PV mass by lowering the cruising speed becomes less significant, as the PV panel becomes more efficient. Therefore, future advances in PV technology should serve to shorten the travel time on Mars for the same PV panel mass.

2.3.4 Effect of PV Panel Efficiency

The Specific Power of the PV panel, SP_{pv}, is related to the Photo cell efficiency, η_{pv}, as:

$$SP_{pv} = G \cdot \eta_{pv} \tag{6}$$

Where G is the average solar insolation available at the surface of the planet. From current PV technology and expected future advancements, a range of 20 to 30% for cell efficiency is expected (see figure 13). It should be noted that the base case value of 20% was chosen because it represents the projected efficiency for amorphous silicone (a-Si) cells which are light, flexible, and easily stored. If the PV efficiency increases by 25%, the same PV panel mass needed for the base case cruising speed of 5 km/hr could power the PCV at a cruising speed over 10 km/hr. Moreover, increasing the efficiency to 30% would increase the PV effectiveness to the point where a 10 km/hr speed for 6 hours of mobility would require less PV mass than that required for life support alone at 20% efficiency. Therefore, future advances in PV technology are expected to have significant impact on reducing the PV mass.

2.3.5 Effect of PV Surface Density

Another parameter that is not completely independent of the efficiency, is the PV surface density, or the amount of mass per square meter of PV panels. The value used for this parameter in the base case is 2.3 kg/m^2. This value is based on current

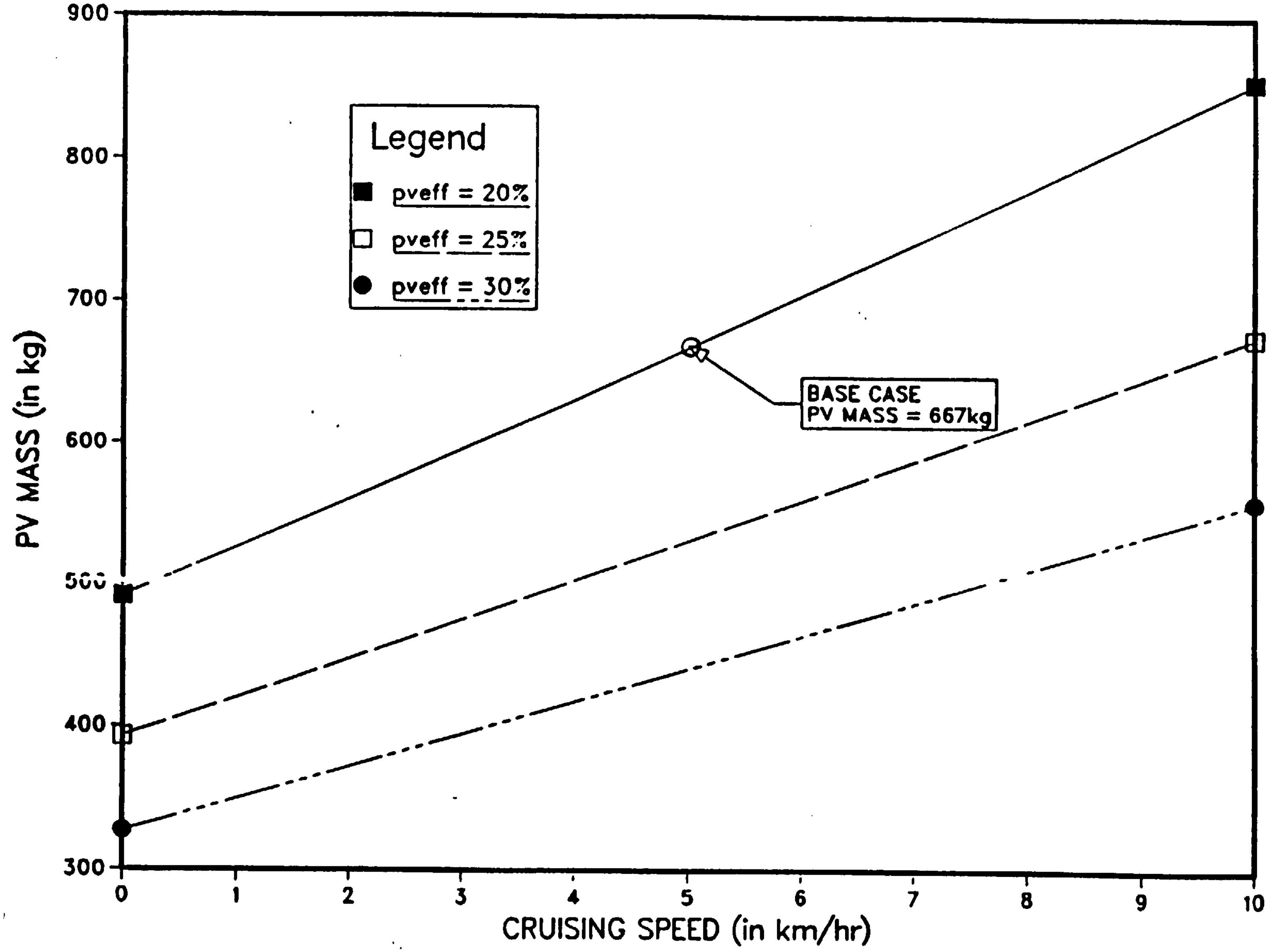


FIGURE 13: Effect of PV Panel Efficiency on PV Mass

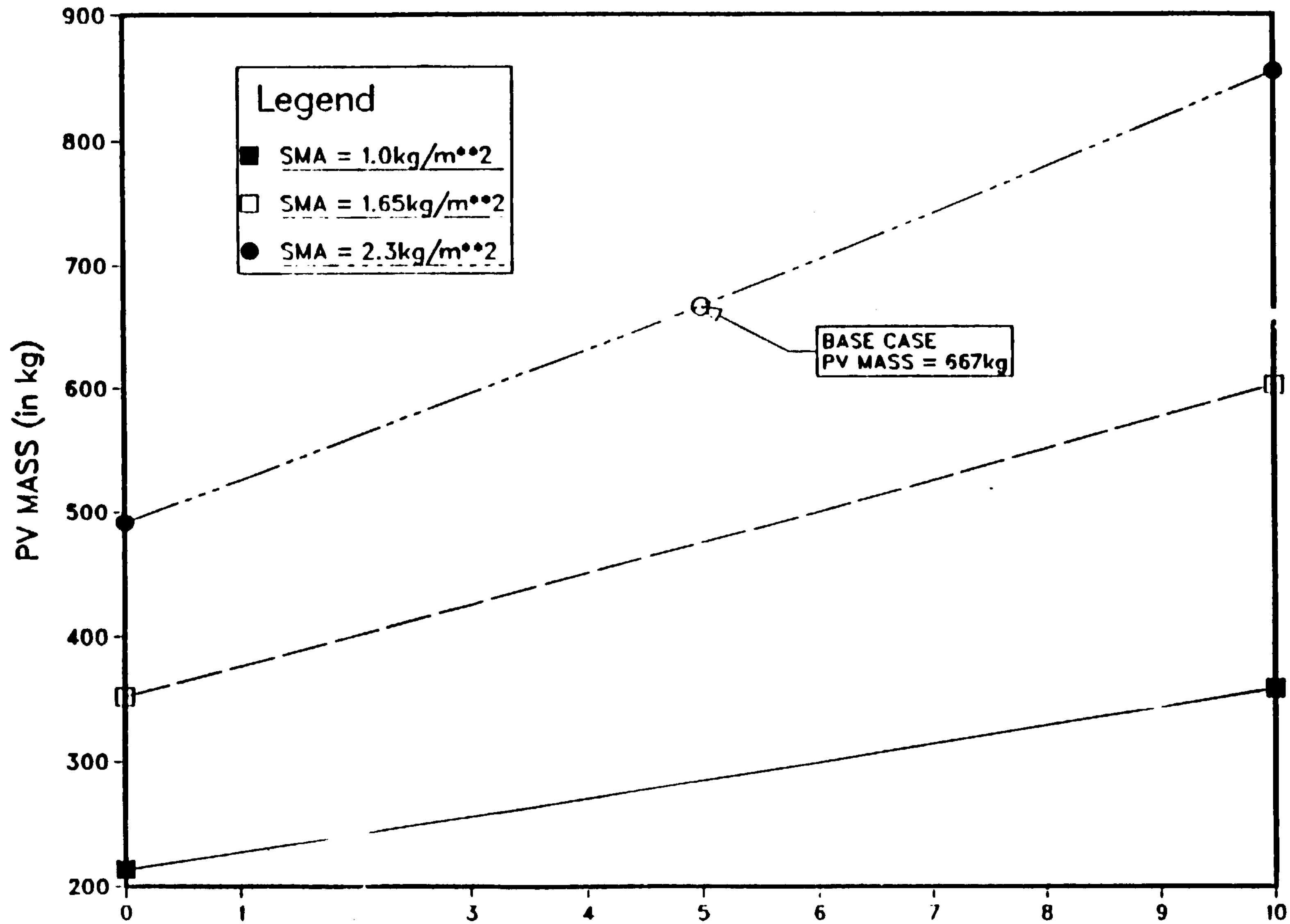


FIGURE 14: Effect of PV Panel Surface Density on PV Mass

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silicone cell technology for a sun tracking system, with a cell efficiency of 20%. With the amorphous-Si cells the specific mass is expected to decline to as low as 1.0 kg/m^2, resulting in a 200% reduction in the mass of the PV panels for the base case. While the mass of the PV cells decreases as the surface density goes down, the total PV area required does not decrease significantly. The PV panel area, however, decreases as the PV cell efficiency increases. Therefore, the trade off is whether to use a larger, less efficient, yet lighter a-Si panels, or smaller, more efficient, but heavier tracking GeAs panels. In either case, the values presented for the base case, should be conservative.

Figure 14 demonstrates the effect of a surface density for 1.0, 1.65, and 2.3 kg/m^2. Note that by using a surface density 1.0 kg/m^2, the total PV mass decreases from the base case (SD = 2.3 kg/m^2) 667kg down to as low as 280kg, but the array area only decrease in from 290 m^2 to 280 m^2 this is because the mass of the PV panels represents a small fraction of the total PCV mass.

2.3.6 Effects of the Number of Astronauts and Reserve Energy Storage

The final two parameters that affect the mass of the PV system are the number of astronauts on board, and the amount of reserve energy storage of the Regenerative Fuel Cells. While both affect the PV mass by increasing the PCV mass and, hence, the mobility power requirements, the number of astronauts has a profound effect on the life support power requirements, which is such a significant portion of the total auxiliary power needs. In figure 15 the mass of the auxiliary PV panels is plotted verses the PCV cruising speed for 2, 4, and 6 astronauts. As figure 15 shows, the change in slope with the number of astronauts is hardly noticeable, yet the offset between the three curves due to the difference in life support power requirements, is significant. These results again emphasize the importance of employing an efficient life support

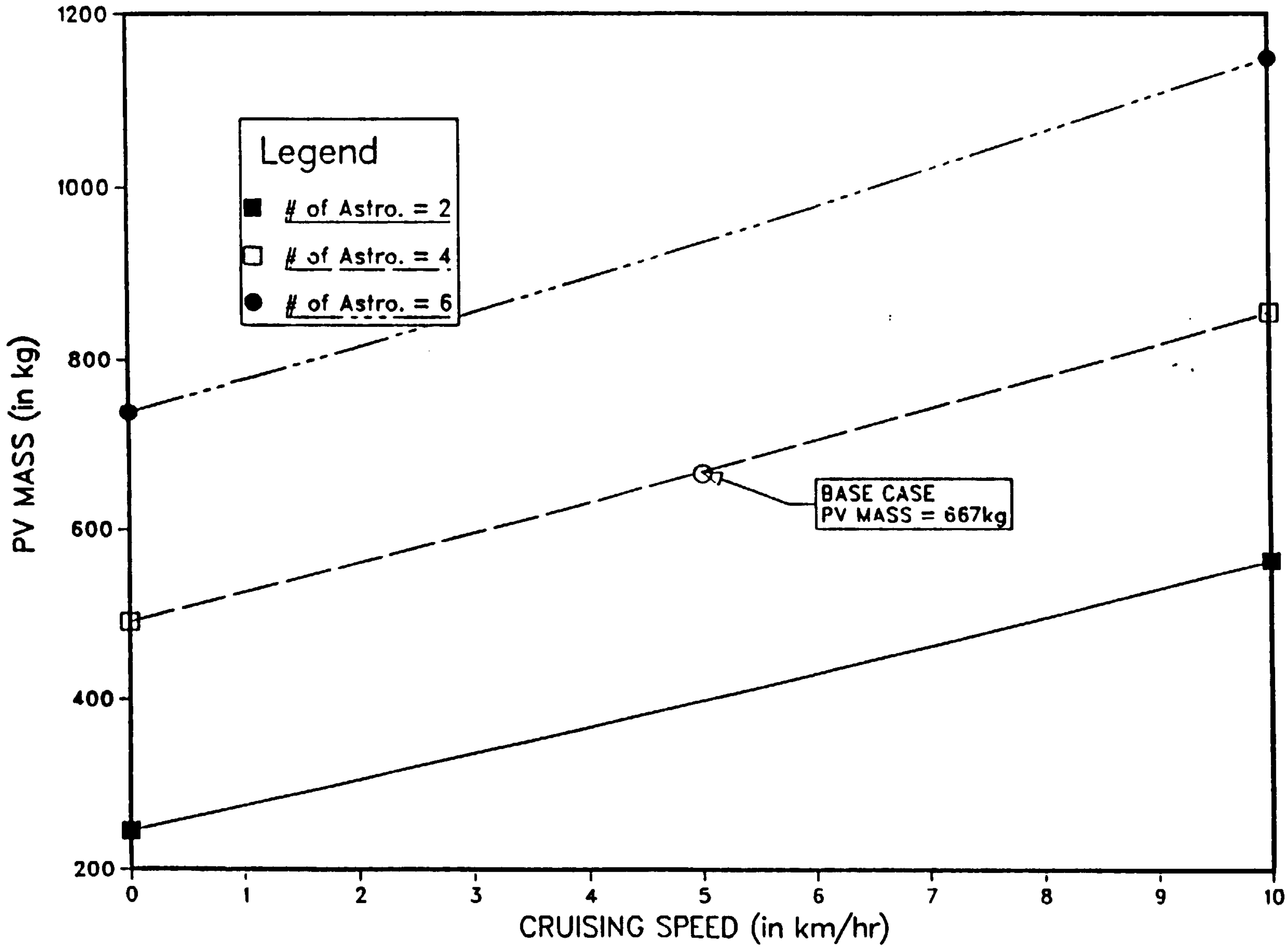

FIGURE 15: Effect of the Number of Astronauts on PV Mass

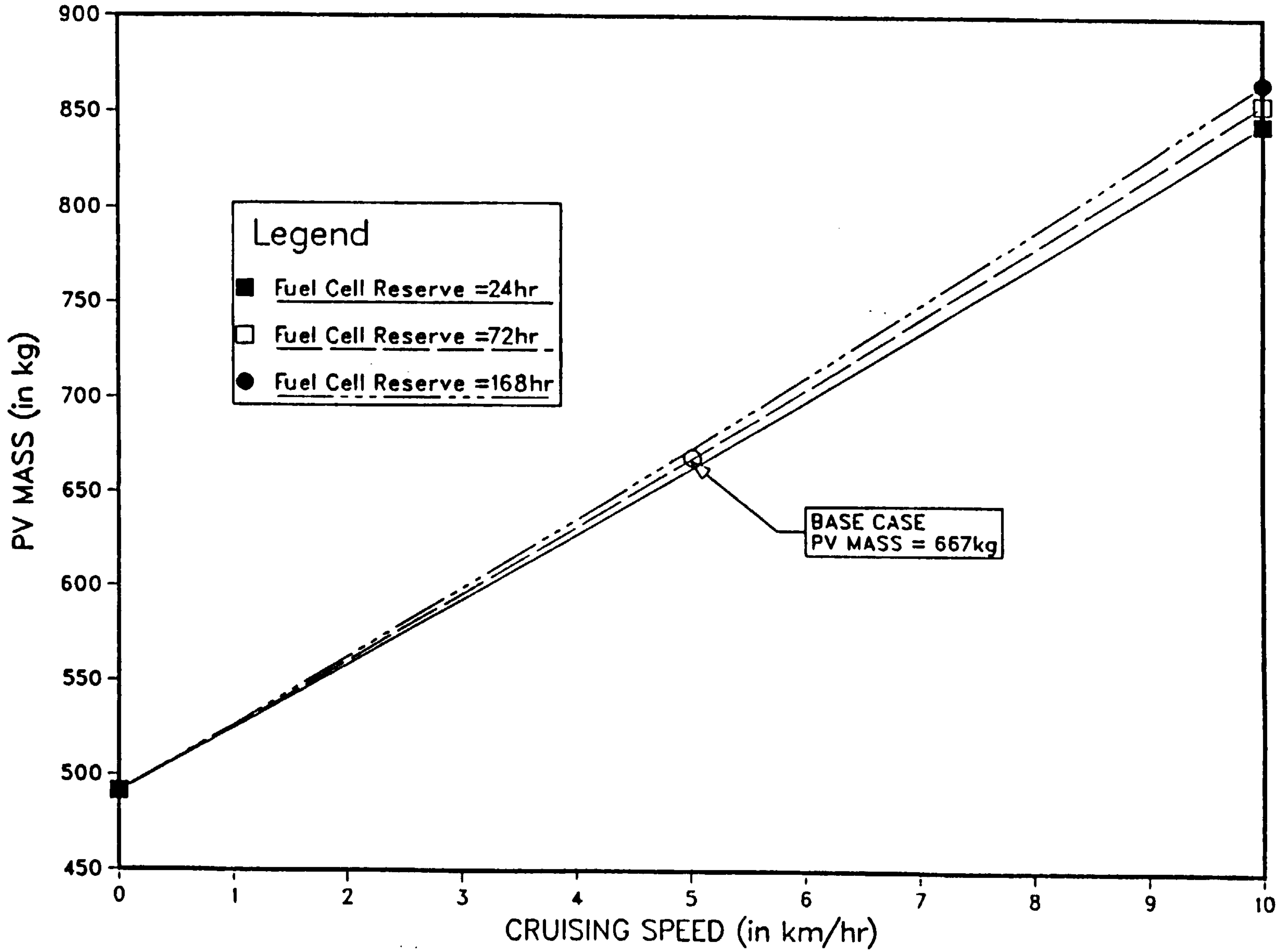

FIGURE 16: Effect of Fuel Cell Reactant Storage Capacity on PV Mass

system. As demonstrated in figure 15 , a PV system that provides for four astronauts at a cruising speed of 10 km/hr, is sufficient to sustain six astronauts, but at a reduced speed of ~3 km/hr.

Increasing the amount of reactants for the reserve fuel cells does not significantly change the mass of the PV panels, since the increase in the reactants mass only affects the mobility power requirement. Also, because the reactant mass is a small fraction of the total PCV mass, increasing the fuel cell capacity from 24 to 168 hr (1 day to 1 week) has only a small affect on the PV panel mass (see figure 16). Since the fuel cells on board of the PCV are regenerative type and the reactants are not stored as cryogenic liquids, but instead as compressed gases, it is not their weight, but rather their storage volume that becomes prohibitive. For example, a one day supply of reactants requires a total storage volume of 3.1 m^3 (1.05 m^3 for O_2 and 2.05 m^3 for H_2), while for 72 hr, the storage volume will be 9.5 m^3 (3.2 m^3 for O_2 and 6.3 m^3 for H_2). For one week storage, a reactant volume of 22.1 m^3 will be needed (7.4 m^3 for O_2 and 14.7 m^3 for H_2).

To summarize the results of the auxiliary PV mass study several points can be made:

(1) The life support power requirements significantly affect the mass and size of the PV panels. It is for this reason, that every effort should be made to design an energy efficient life support system.

(2) In addition to the life support power, the PV panels' specific power also plays a major role in determining the size and mass of the PV panels for the auxiliary power system.

(3) Parameters that affect the mass of the auxiliary PV power system to a lesser extent, but are still important to the overall mass and size determination are the cruising speed of the PCV, the surface density and the efficiency of the PV

cells, and the number of astronauts on board the Rover.

(4) The parameter that has insignificant effect on the auxiliary power system mass, but strongly impacts the volume of the fuel cells, is the reserve capacity of the RFC's.

(5) In order to minimize the size and mass of the auxiliary power system, traversing should be made at night; the day time travel should be limited to as little as possible, and

(6) The PV mass calculated for the base case scenario appears to be reasonable and actually slightly conservative. Therefore, the mass of 667 kg will be added to the PCV mass of 5860 kg from table I, giving a total PCV mass of 6527 kg.

2.4 Experimental Unit

The Experimental Unit (EU) is very similar to the Primary Control Vehicle in that it is a double walled pressurized car capable of maintaining 4 astronauts in a climate controlled environment (see figure 3). This car provides facilities for sample collection and laboratory analysis as well as additional living space for extended missions. The EU also houses the control systems for the primary power system (PPS). The components and equipment pertaining to the EU are listed along with their respective masses in table IV. Again, these masses have been derived from the Eagle Engineering Inc. Report on Lunar Surface Transportation Systems, but have been modified to suit the manned Mars Rover mission.

Unlike the PCV, the EU is not equipped with a PV/RFC auxiliary power system. In the event of a PPS failure, the EU would be detached from the PCV and the PCV would go on using only the auxiliary systems. To account for the possibility of a minor malfunction in the PPS, the EU is equipped with 3 days worth of life support power in

the form of primary fuel cell (PFC). The reactants for these cells are stored in liquid form in order to decrease the required volume. In the event of a malfunction or unscheduled scram of the reactor, the PFC could be used to buy the Rover crew to gain additional time during which they could initiate repairs, before they would be forced to begin an emergency return.

TABLE IV: Experimental Unit Mass Breakdown (EEI Report 88-188)

	Mass (kg)
Structure and Pressure Vessel	
Inner Shell	490
Outer Shell	500
Other Structures	200
Insulation	130
Galley	70
Personal Hygiene	90
Emergency Equipment	30
Man-Locks	230
Work Station	40
EMU's	340
Reactor Control Systems	100
Avionics	90
Environment Control and Life Support	200
Showers	80
Experiments and Payload	900
Active Thermal System	
Radiator	160
Pump	20
Heat Exchanger	50
Piping	100
Refrigerant	300
Wheels and Locomotion	300
Fuel Cells	720
Total	5500

2.5 Supply and Storage Car

The Supply and Storage Car (SC) is much simpler than either the PCV or the EU. The SC is an open trailer which carries all of the radiation insensitive equipment and tools (see figure 3). At the rear of the car a thermal shield is installed to protect the contents from the thermal radiation from the waste heat rejection surface mounted on the reactor car. The SC also contains an additional 24 hr worth of reactants for the PFC in order to supplement that in the EU. The mass breakdown for the EU is given in table V.

TABLE V: Storage Car Mass Breakdown (EEI Report 88-188)

	MASS kg
Thermal Shield	200
Wheels and Locomotion	300
Other Structure	500
Experimental Equipment	887
Fuel Cells	240
Additional Supplies and Equip.	1000
Total	3127

2.6 Reactor Car

Unlike the other three cars of the Rover, the mass of the Reactor Car (RC) cannot be determined by simply tabulating the components and summing their masses. Because mobility power depends upon the mass of the Rover (including the RC), and the mass of the reactor system is a function of the total power needed, an iterative approach, similar to that taken for the PV auxiliary system, is needed to determine the total power required.

In designing the reactor power system, it is essential to determine the electric load

requirements before any real design can take place. For the Manned Mars Rover, a net user power of 30 kW_e was specified as a design requirement. However, since the mobility power is directly related to the total mass of the Rover including that of the Primary Power System, some calculation is needed to assess the power range over which the Rover is likely to operate. Because the mass of the nuclear reactor systems also depend upon its specific mass and the total Power requirement; a model was developed to determine the total electric power requirements of the PPS as a function of: mass of Rover Utility Cars (RUC, including the PCV, EU, SC), cruising speed, number of astronauts, user net power, mobility traction parameter, and the specific mass of the PPS.

Although a good estimate of the power requirement is essential for subsequent reactor and radiation shield design, minimizing the mass of the reactor power system is important, when considering the launch and deployment costs (current estimate figure is approximately \$1M/kg). Therefore, the model was set up to calculate both the total power requirement and the total mass of the rover, including the PPS. The total electric power delivered by the reactor system is given by the general form:

$$P_t = P_a + \chi(8.0\times10^{-5}\ \text{kW·hr/km·kg})M_t\cdot V. \quad (7)$$

Where the mass of the reactor system, which is proportional to its power output, can be given as:

$$M_r = SM_r\cdot P_t. \quad (8a)$$

The reactor systems mass includes the mass of all the reactor components (the radiator, shield, reactor core, energy conversion, thermal management, and plumbing). However, the frame and drive system mass of the Reactor Car, must also be included; This mass is taken as 10% of the total load mass. Hence equation (8a) becomes:

$$M_r = 1.1(SM_r\cdot P_t). \quad (8b)$$

The total rover mass (M_t) is the sum of the masses of the individual Rover cars:

$$M_t = M_{pcv}+M_{eu}+M_{sc}+1.1(SM_r \cdot P_t). \qquad (9)$$

Substituting equation (9) into equation (7) results in the following general form for the total power requirement as:

$$P_t = \frac{P_a + \chi(8.0\text{x}10^{-5}\ \text{kW}\cdot\text{hr/km}\cdot\text{kg})\,[M_{pcv}+M_{eu}+M_{sc}]\cdot V.}{1 - \chi(8.8\text{x}10^{-5}\ \text{kW}\cdot\text{hr/km}\cdot\text{kg})(SM_r)\cdot V} \qquad (10)$$

It can be seen from equation (10) that the total power requirement for the Manned Mars Rover depends on five primary variables: P_a, χ, SM_r, V, and the masses of the Utility Vehicles (M_{pcv}, M_{eu}, M_{sc}). The next chapter presents the results of parametric analysis investigating the effects of each of these variables on the total power requirements for the Rover and on the total mass of the Rover and of the nuclear reactor primary power system.

3. TOTAL MASS AND POWER REQUIREMENTS FOR MANNED MARS ROVER

In this chapter the values for the number of astronauts, the cruising speed, the surface traction parameter, power system specific mass, and the utility car mass will be varied so that their effects on the total Rover mass and the net electric power needed from the reactor can be calculated. The parameters for the base case are listed in table VI.

TABLE VI: GENERAL ROVER VARIABLES

Additional Power above mobility requirements	PA	$= 30\ kW_e$
Mobility traction parameter	χ	$= 0.16 W\cdot h/km\cdot kg$
Drive Efficiency	η_{dr}	$= 50\%$
Specific mass of the Power system	SM_r	$= 100\ kg/KW_e$
Mass of the Primary Control Vehicle (w/o PV's)	M_{pcv}	$= 5860$ kg
Mass of the Experimental Unit	M_{eu}	$= 5400$ kg
Mass of the Supply Car	M_{sc}	$= 3100$ kg
Minimum Rover speed	minspd	$= 0$ km/h
Maximum Rover speed	maxspd	$= 30$ km/h
Number of astronauts	N_a	$= 4$

3.1 Effect of Number of Astronauts

Figure 17 shows that the number of astronauts has a minimal effect on the total rover mass. Increasing the number of astronauts increases the mass of the PCV in table I, and hence increases the auxiliary power system mass and mobility power requirements, but only insignificantly increases the total rover mass. As figure 17 demonstrates, for a cruising speed of 20 km/hr, increasing the number of astronauts from 4 to 6, increases the total rover mass by only 8% (from 25,000 kg to 27,000 kg).

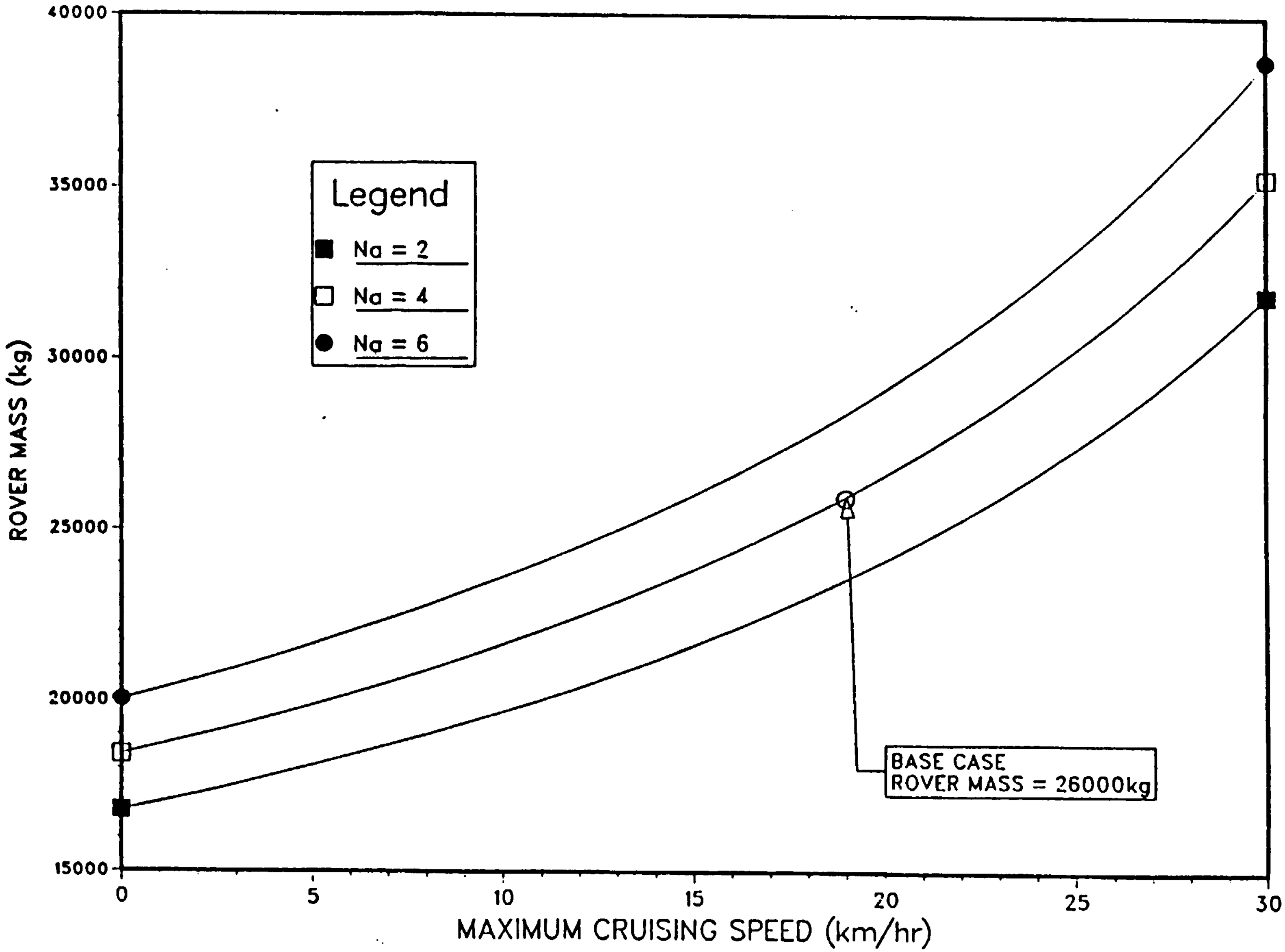

FIGURE 17: Effect of the Number of Astronauts on Total Rover Mass

Most of this mass increase (~1600 kg) reflects the weight of the additional two astronauts, their gear, and the increase in the auxiliary power system mass to provide for the additional astronauts. The remaining mass (less than 400 kg) represents the incremental increase in the mass of the primary power system for meeting the increased mobility and life support power needs.

3.2 Effect of the Cruising Speed and Soil Traction Parameter

As for the cruising speed of the Rover, it could vary depending on the terrain encountered and the sampling and analysis activities being preformed; however, for a manned all terrain vehicle powered with a nuclear reactor power system, 30 km/hr is considered in this analysis as a upper limit for the cruising speed. Since the gravitational acceleration on Mars is almost twice that on the moon, the weight of the Rover vehicle force will increase proportionally and the vehicle will exhibit a higher rolling resistance. However, the wheel design and soil consistency may also vary, which will affect the mobility power requirement.

The results delineated in figure 18 show the strong dependence of the total rover mass on the cruising speed. The mass corresponding to a zero speed represents the mass of the rover vehicles, including that of the PPS, needed to meet the User power requirement (excluding mobility). As this figure shows, increasing the cruising speed increases the mobility power requirements, resulting in higher masses for the PPS and the Rover vehicle. As figure 17 shows, for the base case, the total mass of the Rover employing a 100 kW_e primary power system is 26 metric tonnes; and the average cruising speed is 19 km/hr. As the cruising speed increases to 30 km/hr, the total rover mass for the base case of 4 astronauts increases to 32 metric tonnes (~12% increase), and the power requirements for the PPS increase to 160 kW_e (a 60% increase)

The effects on mobility power of varying the cruising speed and soil traction

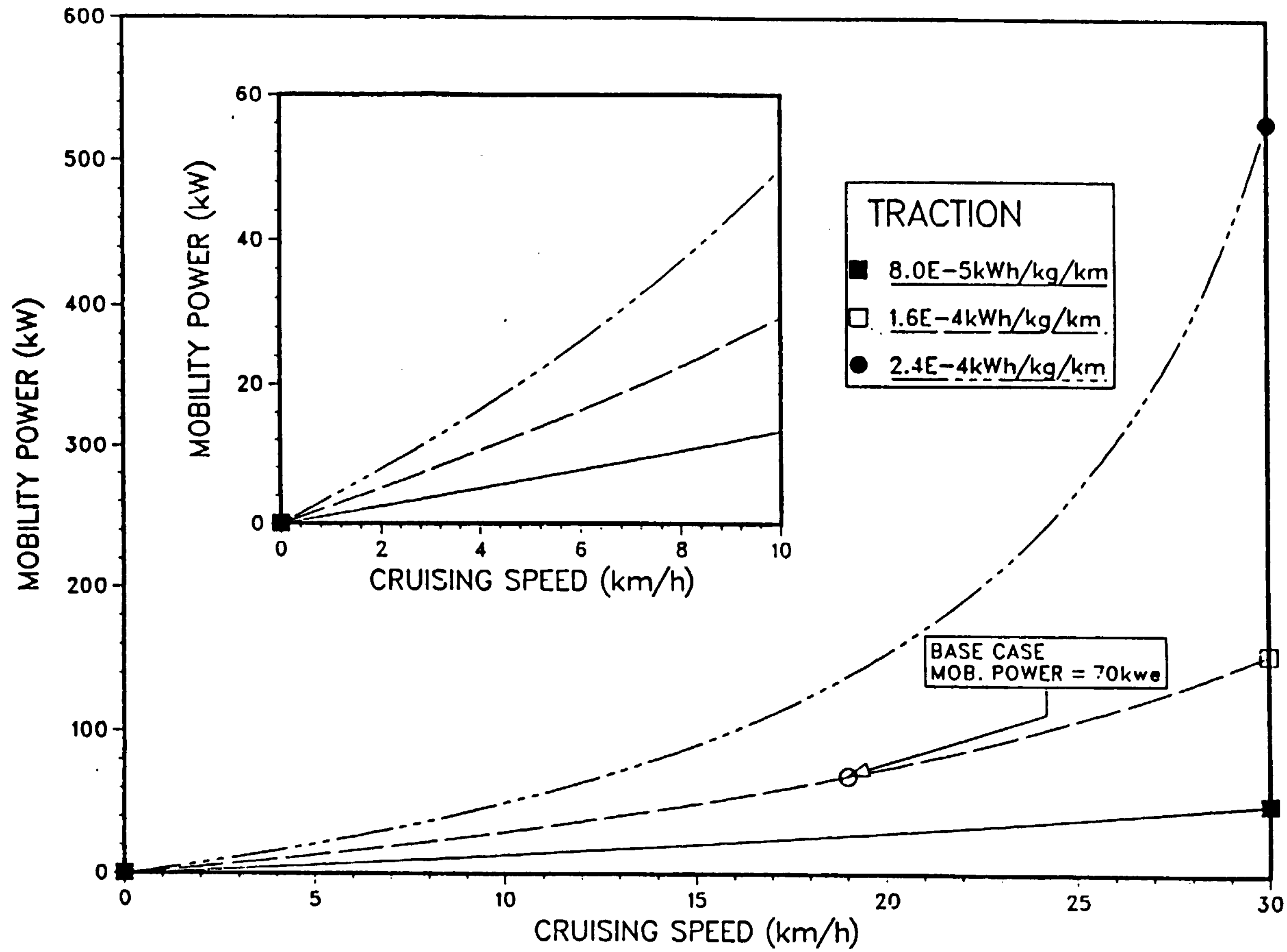

FIGURE 18: Effect of the Soil Traction Parameter on the Mobility Power

parameter are illustrated in figure 18. Here the mobility power requirement is plotted versus the cruising speed, from 0 to 30 km/hr, for three different surface traction values (8.0×10^{-5}, 1.6×10^{-4}, and 2.4×10^{-4} kW·hr/kg·km). Starting with the base case of 1.6×10^{-4} kW·hr/kg·km ($\chi = 2.0$), and a cruising speed of 19 km/hr, the required power level is 70 kW_e for mobility ($P_t = 100 kW_e$). Increasing the cruising speed to 30 km/hr will require a increasing mobility power 120% to 153 kW_e (or 83% of the total power). The effect of altering the surface traction parameter becomes highly significant at higher values of the cruising speed. For example, at a cruising speed of 19 km/hr, increasing the traction value from 1.6×10^{-4} to 2.4×10^{-4} kW·hr/kg·km doubles the mobility power (from 70 to 140 kW_e). However, at 30 km/hr the same increase in surface traction yields increases the mobility power from 153 to 526 kW_e (more than a 360% increase). At the lower value of 8.0×10^{-5} kW·hr/kg·km, the mobility power required at 30 km/hr is 49 kW_e. For the SP-100, 100 kW_e base design, a maximum cruising speed of 30^+, 19, and 12.5 km/hr can be obtained for surface traction factors of 8.0×10^{-5}, 1.6×10^{-4}, and 2.4×10^{-4} kW·hr/kg·km, respectively.

3.3 Effect of Reactor System Specific Mass

The reactor systems specific mass (SM_r) of 100 kg/kW_e, which is used in the base case, yields a power system total mass of 10,000 kg. This includes 3000 kg for all the reactor system (except the radiation shield) at the 100 kW_e level (SP-100 technology), 7000 kg for the radiation shield mass. The additional shield mass will be required to insure the crew's safety during operation. By contrast, the SP-100 shield only weighs 1000 kg, because it is a shadow shield and not designed as a biological shield. It is important to note that the value of 100 kg/kW_e for the nuclear reactor system specific mass is only a first estimate. However, for the purpose of assessing the sensitivity of the total rover mass to the reactor systems specific mass over the expected range of

possible operation, the specific mass of the reactor systems is varied for 50 to 150 kg/kW$_e$. A more accurate estimate of the reactor system's specific mass will be obtained in Part II of this report, when the shield configuration and composition, the radiator size, energy conversion system, and waste heat rejection system are better defined. The next phase of the project will focus on the design and integration of the radiation shield and the nuclear reactor with the rover.

In Figure 19, the total reactor power requirements as a function of cruising speed is shown for three different specific mass values (50, 100, and 150 kg/kW$_e$). For all three curves, the minimum power is 30 kW$_e$ (user net power only) at 0 km/hr. At low cruising speeds (< 10 km/hr) the reactor system's specific mass has virtually no effect on the reactor power. However, once the speed begins to increase beyond 10 km/hr, the specific mass begins to show a significant effect on the total reactor power requirement. For example, at 30 km/hr the total power requirements double from the 183 kW$_e$ at 100 kg/kW$_e$ to 378 kW$_e$ at 150 kg/kW$_e$. Figure 19 also shows that for the base case of 100 kW$_e$, the maximum cruising speed attainable for the rover is 23, 19, and 16.5 km/hr, for nuclear reactor system specific masses of 50, 100, and 150 kg/kW$_e$, respectively.

The specific mass of the reactor power system; however, greatly impacts the total rover mass (see Figure 20). As this Figure shows, for 100 kW$_e$, while increasing the system's specific mass from 100 to 150 kg/kW$_e$ only decreases the maximum cruising speed by 2.5 km/hr, it increases the total rover mass from 26 to 32 metric tonnes. This 6,000 kg increase in the rover mass is significant, considering the launch costs to Mars, (additional launch cost of approximately 6 billion dollars).

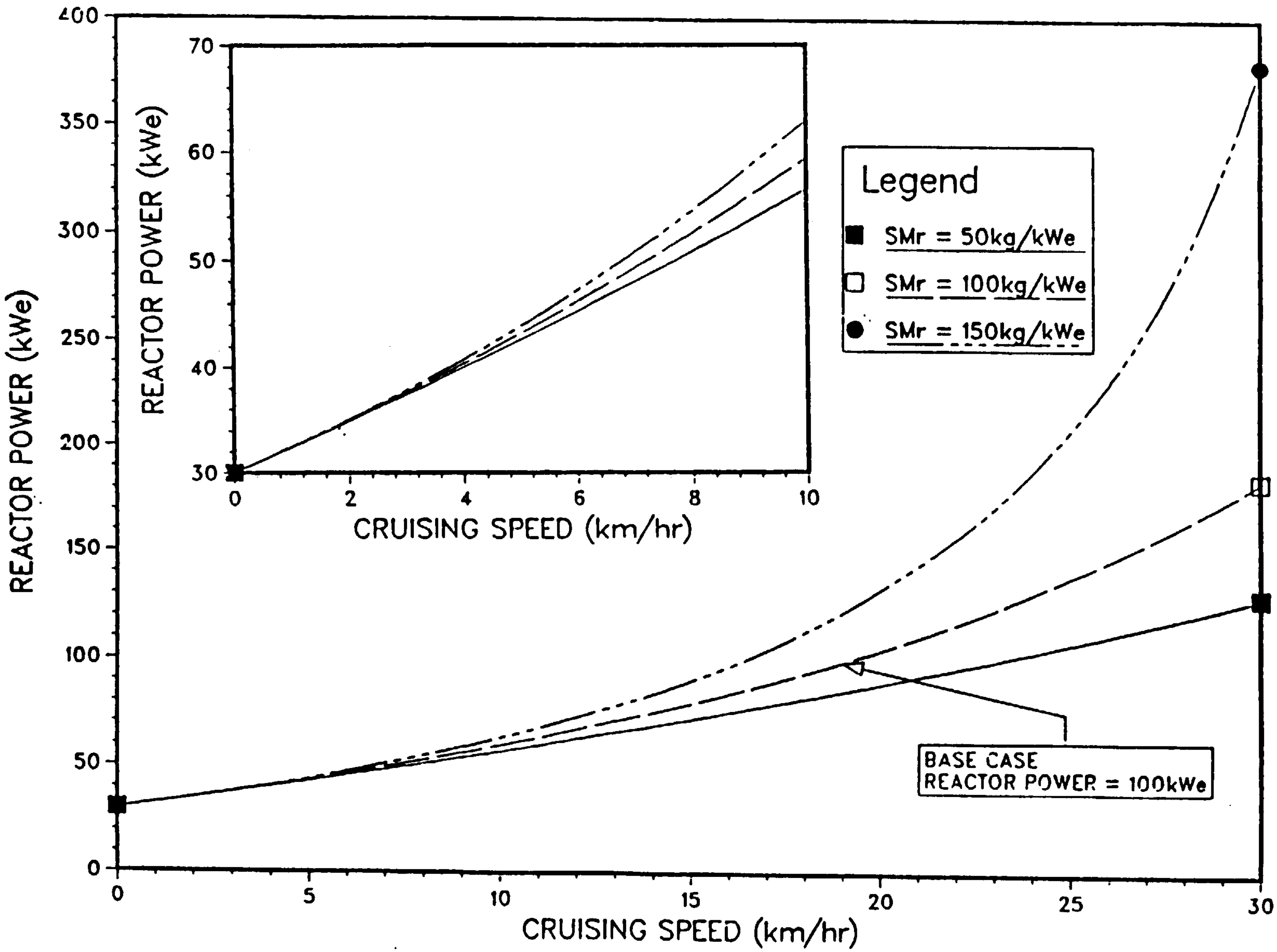

FIGURE 19: Effect of Reactor Specific Mass on Total Power Requirement

3.4 Effect of the Utility Cars' Masses

For equation (10) the masses of the Rover Utility Cars (RUC) are determined in the previous sections and are presented in tables I, IV, V. Although the values for these masses are as accurate of an estimate as currently possible, the exact masses are intimately related to the actual mission scenario. Therefore, the effects of changing the total mass of these three cars on the total rover mass and the reactor power is also investigated. The total RUC masses are varied from 15,000 to 18,000 kg.

As with the other variables, the rover mass is plotted verses the cruising speeds in the range from 0 to 30 km/hr for four values for the RUCs masses (see figure 21). At a speed of 0 km/hr the total Rover mass varies from 18,412 to 21,412 kg (a 3,000 kg difference). This difference in total mass reflects only the increase in the RUCs masses from 15,000 to 18,000 kg. At 19 km/hr the same increases in the RUCs masses increases the total Rover mass from 26,004 to 30,511 kg, a difference of 4,507 kg. The additional 1,507 kg reflects the increase in the PPS mass needed to meet the additional mobility power requirements. For same reactor system mass, total power of 100 kW_e, and 18 tonnes for the RVCs masses the rover maximum cruising speed would be reduceded from 19.6 km/hr to 17 km/hr. However, this reduction in the cruising speed should not significantly affect the rovers performance, and therefore the exact RUCs masses are not critical in evaluating the PPS power requirements.

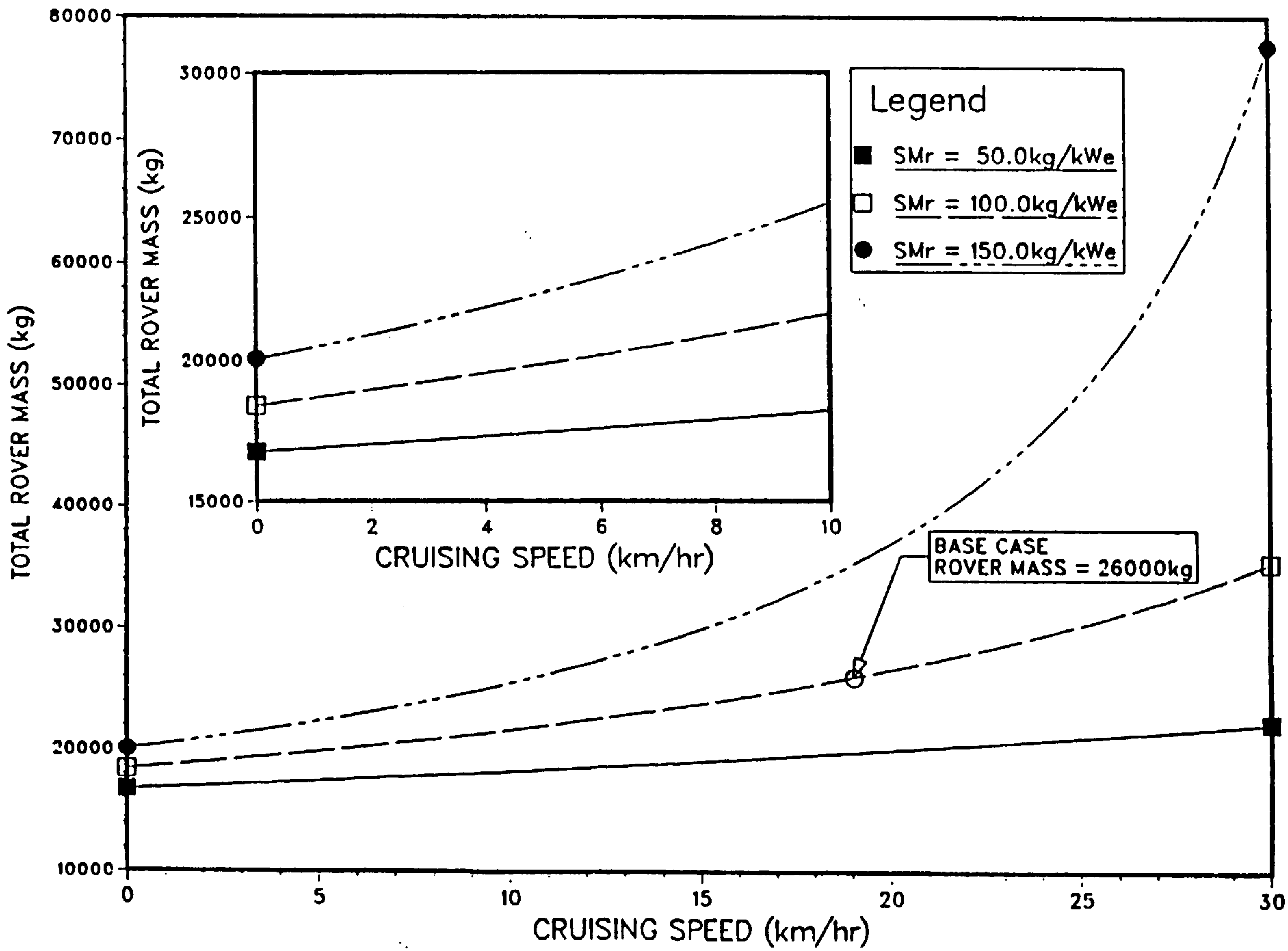

FIGURE 20: Effect of Reactor Specific Mass on Total Rover Mass

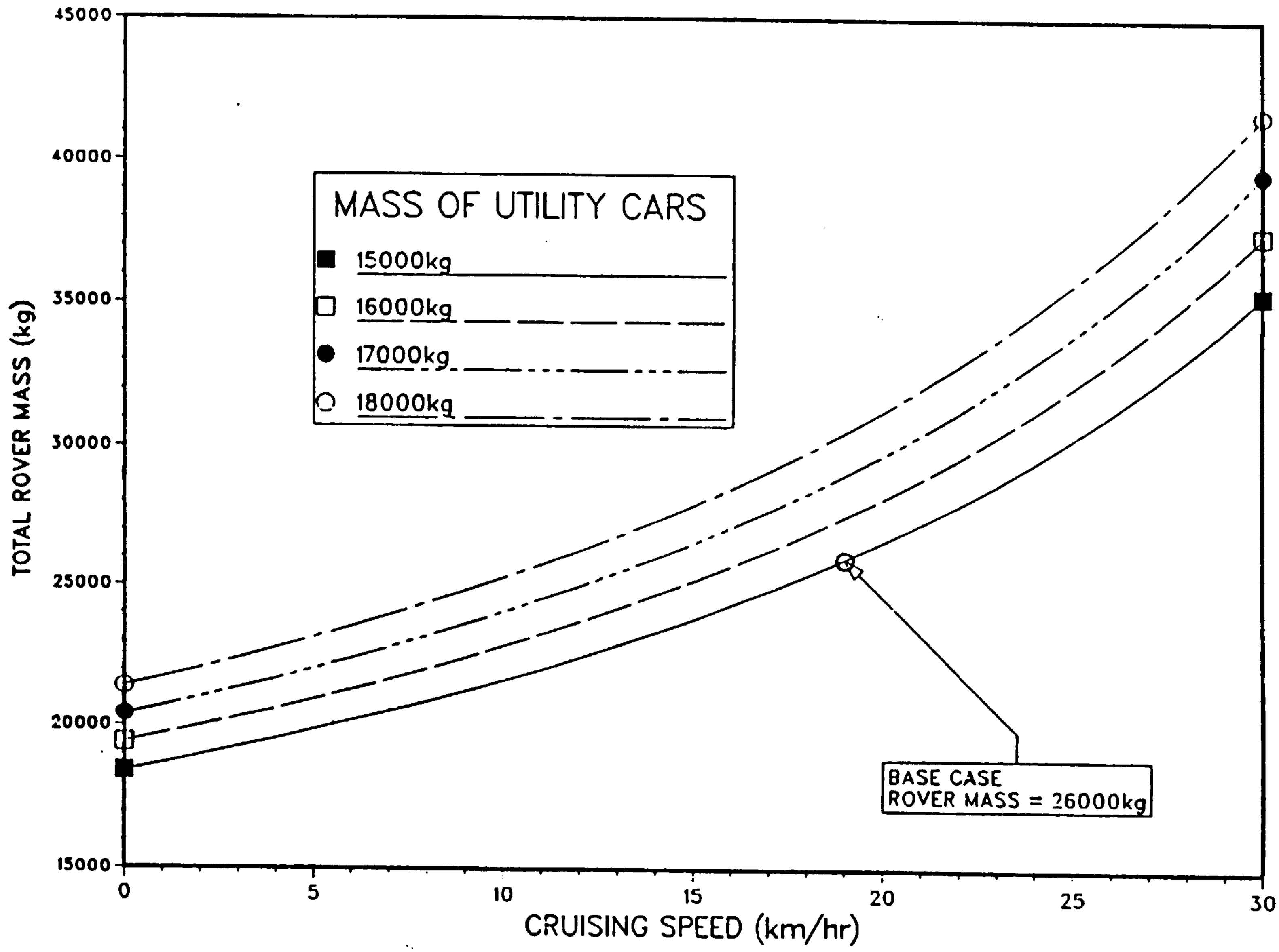

FIGURE 21: Effect of Utility Car Mass on Total Rover Mass

4. SUMMARY AND CONCLUSIONS

A parametric analysis assessing the total power requirements and mass of a nuclear reactor powered Mars Manned Rover is performed. This analysis shows that the total power and mass is strongly affected by the cruising speed, specific mass of the power system, and the surface traction and drive efficiency, and to a lesser extent by the mass of the rover utility cars. An SP-100 reactor power system, generating 100 kW_e, would provide for a maximum cruising speed of 16.5 km/hr at a systems' specific mass of 150 kg/kW_e and 19 km/hr at a system specific mass of 100 kg/kW_e. At 100 kWe power level, the mass of the PPS is 32% of the total rover mass for a specific mass of 100 kg/kW_e and 53% at a specific mass of 150 kg/kW_e. The total mass of the rover at a specific mass of 100 kg/kW_e is approximately 26 metric tonnes. This mass will be 23% higher to 32 metric tonnes at a specific mass of 150 kg/kW_e.

The auxilliary power system to support an emergency return trip is selected to be a hybrid of PV panels and regenerative fuel cells. The cells have a total storage capacity for three days of life support, mobility for 6 hr/day at a maximum cruising speed of 5 km/hr. The study found that the life support power requirement per astronuat strongly affects the mass and size of the PV panels for the auxilliary power system. Therefore, a careful assessment of life support power requirements is stongly recommended. The mass and the size of the PV panels also depend on parameters as the cruising speed, the PV panels surface density and efficiency, and the number of astronauts on board the rover. Results show that in order to minimize the size and mass of the PV panels, traversing during emergency return should be made at night and day time travel should be limited to as little as possible.

5. FUTURE WORK

In the Second Phase of the project, three components of the PPS will be studied in order to determine the nuclear reactor system's specific mass. These components include the radiation shield, the energy conversion system, and the waste heat rejection system. For the shielding calculations the maximum radiation dose to Astronaut from the reactor will be limited to 5 rem/year. Also, an average separation distance from the reactor of 25 m will be set as the base design case. This distance, which is based on the size and configuration of the rover cars (see figure 3) will be optimized for minimum radiation shield mass.

For the energy conversion system, four types will be studied and their impact on the Reactor system mass will be compared. These systems are:

(1) GeSi/GaP thermal electric (TE) converters, currently employed in the SP-100 nominal design. TEs are a highly reliable due to the absence of any moving parts and enjoy demonstrated flight experience, but posses a low conversion efficiency (less than 6%).

(2) The Free Piston Stirling Engine, although has the possibility of much higher conversion efficiency (>15%), the technology is still in the development and demonstration stage. Nonetheless, it is hoped that stirling engine technology would be readily available by the time a Manned Mars Mission is deployed (year 2020-2030).

(3) Open Loop Brayton Cycle, utilizing the CO_2 from the atmosphere of Mars as a working fluid will also be investigated. Due to the low daily temperature on the surface of the planet (190-240 K in the summer and 150 K in the winter) this conversion system has the possibility for extremely high conversion efficiencies (>30%). Although an open loop Brayton cycle would eliminate the need for a heat rejection radiator, because of the low atmospheric pressure on Mars (~10

mbar) the size and mass of the turboalternator unit is expected to be large. However, an analysis will be performed to optimize the power system for the lowest possible total mass and best integration with the rover vehicle.

(4) A He-40%Xe closed Brayton cycle will also be investigated. Because He-Xe is a better working fluid than the CO_2 for the conversion cycle, and the operating pressure could be much greater than that in the open CO_2 cycle option, the size and mass of the turboalternator unit for the former will be much smaller than for the later. On the other hand, in order to obtain reasonable efficiencies (> 15%) the exit temperature from the turbine has to be lowered, thus requiring a large mass and size for the waste heat radiator

For the waste heat rejection, three systems will also be considered. The first option is a radiative heat rejection system, limited by the surface area which can be mounted on the outer surface of the reactor car (~80 m^2) The second is a forced convection CO_2 system, in which atmospheric CO_2 is blown through a heat exchanger. The final option is a hybrid combining radiative and convective heat rejection.

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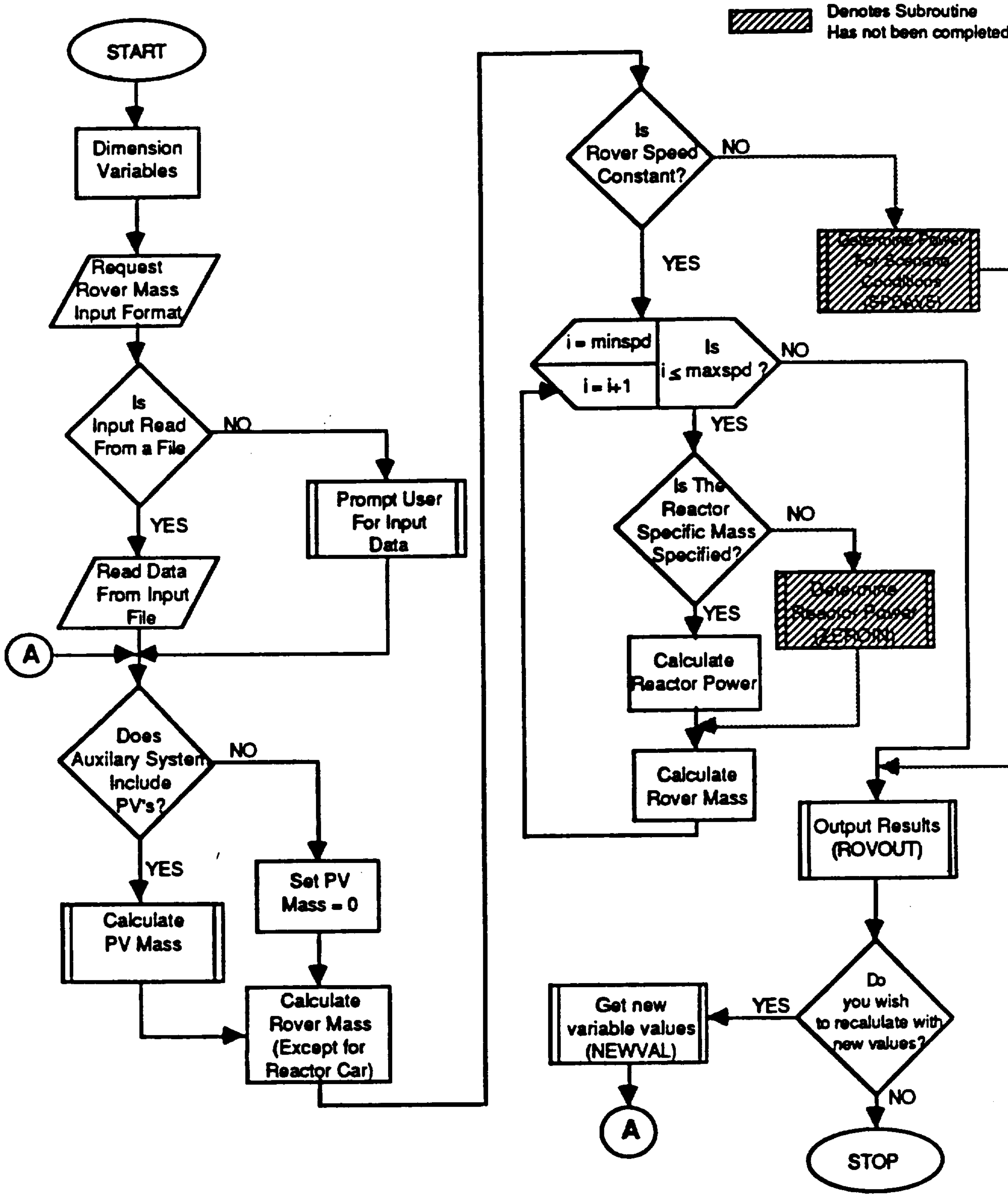
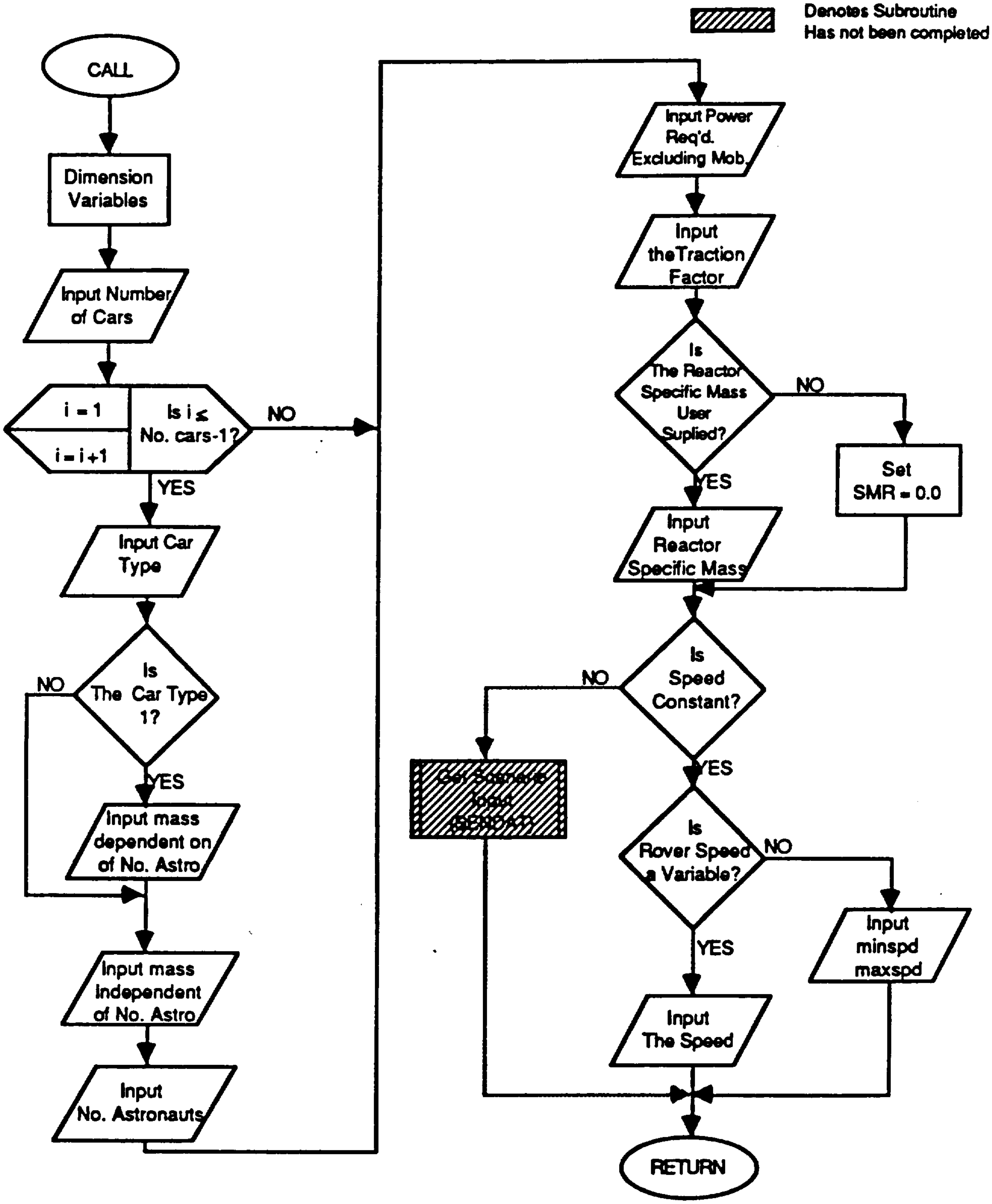
APPENDIX-A

MODEL DESCRIPTION AND FLOW CHARTS FOR ROVER MASS PROGRAM

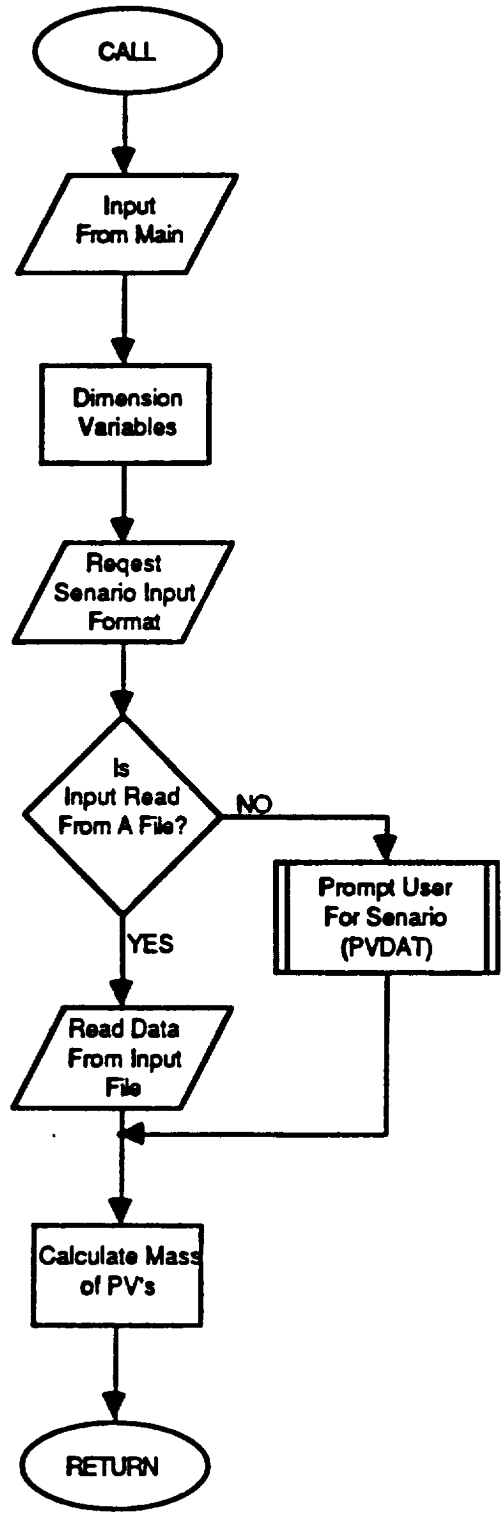
MAIN PROGRAM ROVER MASS



SUBROUTINE INPUT



SUBROUTINE PVMASS



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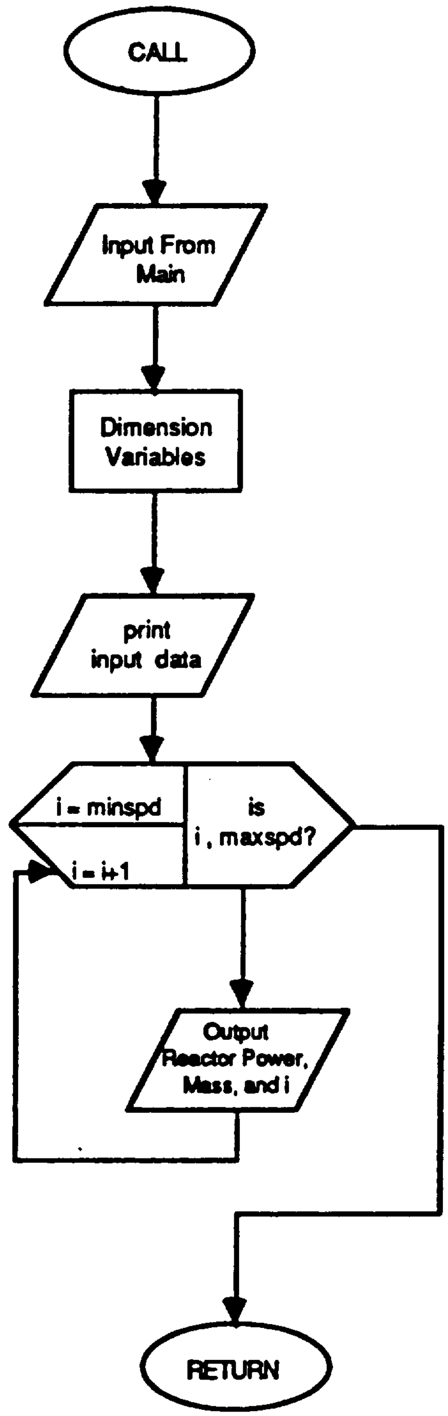


APPENDIX B

LISTING FOR ROVER MASS PROGRAM

```
c              Program Reactor Mass
c
c
c
c
      real    rovmas(0:50,5), pa, x, mpcv1, mpcv2, meu, msc, smr, v,
     &        rovp(0:50,5), mpv
      integer i, j, range, systyp, modtyp, auxtyp, smtyp
      character*60  xaxis, yaxis, title, consc(11), intype,
     &              rerun
c
      print*,'Do you what an input prompt (yes or no)?'
      read(5,10)intype
      open(unit=40,file='pvmas',status='new')
 10   format(a)
      if (intype.eq.'no') then
        open (unit=16,file='rdat',status='old')
        rewind 16
        read(16,*)pa, smr, minspd, maxspd, x, mpcv1, mpcv2, meu,
     &             msc, na, modtyp, systyp, smtyp, auxtyp
      else
        call rovdat(pa, smr, minspd, maxspd, x, mpcv1, mpcv2, meu,
     &               msc, na, modtyp, systyp, smtyp, auxtyp)
      endif
      open(unit=35,file='pwrout',status='new')
      j = 1
 15   write(35,16)j
 16   format(/,35x,'RUN #',i2,///)
      if (auxtyp.eq.1) then
        fc  = 0.0
        mpv = 0.0
      else
        call pvfc(na, mpcv1, mpcv2, x, mpv, meu, msc, fc)
      endif
      mrov = mpcv1+na*mpcv2+meu+msc+mpv+fc
      if (modtyp.eq.1) then
        call spdave(pa, mrov, x, smr, smtyp)
      else
        range = maxspd-minspd
        do 30  i = 0, range
          v = float(i+minspd)
          if(smtyp.eq.1) then
            call zeroin(smr, rovpwr(i,j), mrov, x, v, pa)
          else
            rovp(i,j) = pa+x*0.08/1000.0*(mrov*v)/(1.0-x*
     &                    (0.088/1000.0*(smr*v)))
          endif
          rovmas(i,j) = rovp(i,j)*1.1*smr+mrov
 30     continue
      endif
      call rovout(pa, j, smr, minspd, maxspd, x, mpcv1, mpcv2, meu,
     &             msc, na, systyp, mpv, rovp, rovmas)
      print*,'Do you want to change any of the values (yes or no)?'
      read(5,35)rerun
 35   format(a)
      if (rerun.eq.'yes') then
        j = j+1
        call newval(pa, smr, minspd, maxspd, x, mpcv1, mpcv2, meu,
     &               msc, na, modtyp, systyp, smtyp, auxtyp)
        go to 15
      endif
      stop
      end
c
c**********************************************************************
c
```

```
c
c
      subroutine rovdat(pa, smr, minspd, maxspd, x, mpcv1, mpcv2,
     &                   meu, msc, na, modtyp, systyp, smtyp, auxtyp)
c
      real  pa, smr, x, mpcv1, mpcv2, meu, msc
      integer na, maxspd, minspd, cartyp, systyp, modtype, smtyp,
     &        auxtyp
c
c
      print*,'Enter number of cars in the rover.'
      read(5,*)numcar
      print*,' '
      print*,' '
      do 1010 i = 1, numcar
 1000   print*,' '
        print*,' '
        print1005,i
 1005   format(' Enter the type of car number ',i1)
        print*,'Primary Control Vehicle (PCV) - 1'
        print*,'Experimental Unit       (EU)  - 2'
        print*,'Storage and Supply Car  (SC)  - 3'
        read(5,*)cartyp
        if (cartyp.eq.1) then
         print*,' '
         print*,' '
         print*,'Enter the mass of the PCV independent of the'
         print*,'number of Astronauts (in kg).'
         read(5,*)pcv1
         mpcv1 = mpcv1+pcv1
         print*,' '
         print*,' '
         print*,'Enter the mass of the PCV dependent of the'
         print*,'number of Astronauts (in kg/astro.).'
         read(5,*)pcv2
         mpcv2 = mpcv2+pcv2
        else
         if (cartyp.eq.2) then
          print*,' '
          print*,' '
          print*,'Enter the mass of the EU (in kg).'
          read(5,*)eu
          meu = meu+eu
         else
          if (cartyp.eq.3) then
           print*,' '
           print*,' '
           print*,'Enter the mass of the SC (in kg).'
           read(5,*)sc
           msc = msc+sc
          else
           print*,' '
           print*,' '
           print*,'***INVALID ENTRY***'
           go to 1000
          endif
         endif
        endif
 1010 continue
      print*,' '
      print*,' '
      print*,'Enter the number of astronauts.'
      read(5,*)na
      print*,' '
      print*,' '
      print*,'Enter the power req.''d in addition to mobility (in kWe).'
```

```
      read(5,*)pa
      print*,' '
      print*,' '
      print*,'Enter the traction factor modifier x.'
      read(5,*)x
      print*,' '
      print*,' '
      print*,'Enter the method of reactor specific mass determination.'
      print*,'User specified     - 0'
      print*,'Program calculated - 1'
      read(5,*)smtyp
      print*,' '
      print*,' '
 1020 if (smtyp.eq.0)then
        print*,'Enter the reactor specific mass (in kg/kWe).'
        read(5,*)smr
        print*,' '
        print*,' '
      else
        if (smtyp.ne.1) then
          print*,' '
          print*,' '
          print*,'***INVALID ENTRY***'
          go to 1020
        else
          smr = 0.0
        endif
      endif
 1040 print*,'Enter the traversing senario.'
      print*,'Constant speed - 0'
      print*,'Variable speed - 1'
      read(5,*)modtyp
      print*,' '
      print*,' '
      if (modtyp.eq.0) then
        print*,'Would you like to calculate the mass for a'
        print*,'Single speed    - 0'
        print*,'Range of speeds - 1.'
        read(5,*)systyp
        print*,' '
        print*,' '
 1050   if (systyp.eq.1) then
          print*,'Enter the minimum Rover speed (in km/hr).'
          read(5,*)minspd
          print*,' '
          print*,' '
          print*,'Enter the maximum Rover speed (in km/hr).'
          read(5,*)maxspd
          print*,' '
          print*,' '
        else
         if (systyp.eq.0) then
           print*,'Enter the Rover speed (in km/hr).'
           read(5,*)minspd
           maxspd = minspd
           print*,' '
           print*,' '
         else
           print*,' '
           print*,' '
           print*,'***INVALID ENTRY***'
           go to 1050
         endif
        endif
      else
        if (modtyp.eq.1) then
```

```
            call sendat (maxspd,minspd)
          else
             print*,' '
             print*,' '
             print*,'***INVALID ENTRY***'
             go to 1040
          endif
        endif
        print*,'Enter '
        print*,'0 - Photovoltaic auxiliary power system'
        print*,'1 - No PV''s in the auxilairy system'
        read(5,*)auxtyp
        return
        end
c
c
c***************************************************************************
c
c
c
c
        subroutine pvfc(na, mpcv1, mpcv2, x, mpv, meu, msc, fc)
c
c
        real    mpvd, mpcv1, mpcv2, mpcv, mfc1, mcells, meu, msc, fc,
     &          mfc2, mfc3, mpv
        integer i, j, na
c
 2000   print*,'This routine is designed to either accept input from'
        print*,'the emergency senario from a data file called auxdat.'
        print*,'If this file does file does not exist then you may '
        print*,'specify a full prompting for this information enter:'
        print*,'          0 - Senario prompting'
        print*,'          1 - Read from Auxdat'
        read(5,*)ptyp
        if (ptyp.eq.0) then
          call auxpromt (sma, tc, fceff, tmd, tmn, pls, ve, ta,
     &                  g, pveff, ta2, ta3, apvs)
          print*,tmd,tmn,ta,tc
        else
          if (ptyp.eq.1) then
            open(unit=30,file='auxdat',status='old')
            rewind 30
            read(30,*)sma, tc, fceff, tmd, tmn, pls, ve, ta, g, pveff,
     &              ta2, ta3, apvs
            close(unit=30)
          else
            print*,'***INVALID ENTRY***'
            go to 2000
          endif
        endif
        td     = 24.62
        mcells = 2.0
        mfc1   = ta*pls*na*(0.5+mcells)
        mpcv   = mpcv1+mpcv2*float(na)+mfc1
        smpv   = sma/(g*pveff)*1000.0
        do 2005 i = 1,20
        tc = .615*float(i)
        mpvd   = (na*pls*(tc+(td-tc)/fceff)-12.3*apvs*g*pveff/1000.+x*
     &           (0.08/1000.0)*(mpcv+apvs*sma)*ve*(tmd+tmn/fceff))*smpv/
     &           (tc-x*(0.08/1000.0)*ve*(tmd+tmn/fceff)*smpv)
        mpv    = mpvd+apvs*sma
        write(40,*)tc, mpv
 2005   continue
        mfc2   = ta2*pls*na*(0.5+mcells)
        mfc3   = ta3*pls*na*(0.5+mcells)
```

```
      fc     = mfc1+mfc2+mfc3
      tatot  = (ta+ta2+ta3)*pls*na
      write (35,2010)
      write (35,2020)smpv, fceff, tc, apvs, tmd, tmn, pls, ve, tatot,
     &              mfc1, mfc2, mfc3
 2010 format(25x,'Auxiliary Power System Input',//)
 2020 format(15x,'PV specific mass                      = ',f7.2,' kg/kWe',/,
     &       15x,'Fuel cell efficiency                  = ',f7.2,/,
     &       15x,'Deployable PV collection time         = ',f7.2,' hr',/,
     &       15x,'Stationary PV area                    = ',f7.2,'m**2',/,
     &       15x,'Mobility time during the day          = ',f7.2,' hr',/,
     &       15x,'Mobility time during the night        = ',f7.2,' hr',/,
     &       15x,'Power needed for emergency LS         = ',f7.2,' kWe/Astro'
     &       ,/,
     &       15x,'Auxiliary cruising speed              = ',f7.2,' km/hr',/,
     &       15x,'Total fuel cell reserve               = ',f7.2,' kWh',/,
     &       15x,'PCV fuel cell mass                    = ',f7.2,' kg',/,
     &       15x,'EU fuel cell mass                     = ',f7.2,' kg',/,
     &       15x,'SC fuel cell mass                     = ',f7.2,' kg',///)
      return
      end
c
c
c*********************************************************************
c
c
c
      subroutine rovout (pa, j, smr, minspd, maxspd, x, mpcv1, mpcv2,
     &                  meu, msc, na, systyp, mpv, rovp, rovmas)
c
c
      real pa, smr, mpcv1, mpcv2, meu, msc, mpv,
     &     rovp(0:50,5), rovmas(0:50,5)
      integer j, i, na, systyp, maxspd, minspd, spd
c
c
      write(35,3000)
 3000 format(22x,'Mass and Power Requirements for Nuclear',/,
     &       28x,'Reactor Powered Mars Rover',//,36x,'Input Values')
      write(35,3010)pa, smr, minspd, maxspd, x, mpcv1, mpcv2,
     &             meu, msc, mpv, na
 3010 format(15x,'Supplemental power above mobility    = ',f8.2,' kWe',/,
     &       15x,'Specific mass of the reactor         = ',f8.2,' kg/kWe'
     &       ,/,
     &       15x,'Rover minimum speed                  = ',i8,' km/h',/,
     &       15x,'Rover maximum speed                  = ',i8,' km/h',/,
     &       15x,'Rover traction factor                = ',f8.2,/,
     &       15x,'Astronaut independent PCV mass       = ',f8.2,' kg',/,
     &       15x,'Astronaut dependent mass             = ',f8.2,' kg/astr'
     &       ,/,
     &       15x,'Mass of experimental unit            = ',f8.2,' kg',/,
     &       15x,'Mass of the supply car               = ',f8.2,' kg',/,
     &       15x,'Mass of the Photovoltaics            = ',f8.2,' kg',/,
     &       15x,'Number of astronaunts                = ',i8,///)
      write(35,3020)
 3020 format(36x,'Output Values',/,22x,'Rover Speed',3x,'Reactor Power',
     &       3x,'Rover Mass',/,26x,'km/h',12x,'kWe',12x,'kg')
      do 3030 i = 0, maxspd-minspd
        spd = i+minspd
        write(35,3040)spd, rovp(i,j), rovmas(i,j)
 3030 continue
 3040 format(26x,i3,10x,f7.3,7x,f8.2)
      write(35,3050)
 3050 format(///,80('*'))
      return
      end
```

```
c
c
c*************************************************************************
c
c
c
      subroutine auxpromt(smpv, tc, fceff, tmd, tmn, pls, ve, ta, g,
     &                       pveff, ta2, ta3, apvs)
c
      print*,' '
      print*,' '
      print*,'Enter the specific mass of the PV cells (in kg/m**2).'
      read(5,*)smpv
      print*,' '
      print*,' '
      print*,'Enter the time of reserve power in PCV''s fuel cells',
     &       '(in hours).'
      read(5,*)ta
      print*,' '
      print*,' '
      print*,'Enter the time of reserve power in EU''s fuel cells',
     &       '(in hours).'
      read(5,*)ta2
      print*,' '
      print*,' '
      print*,'Enter the time of reserve power in SC''s fuel cells',
     &       '(in hours).'
      read(5,*)ta3
      print*,' '
      print*,' '
      print*,'Enter the efficiency of the Fuel Cell cycle.'
      read(5,*)fceff
      print*,' '
      print*,' '
      print*,'Enter the power needed for life support (in kWe/Astro).'
      read(5,*)pls
      print*,' '
      print*,' '
      print*,'Enter the daily collection time for deployable PV''s',
     &       ' (in hours).'
      read(5,*)tc
      print*,' '
      print*,' '
      print*,'Enter the area of stationary PV''s (in sq. meters)'
      read(5,*)apvs
      print*,' '
      print*,' '
      print*,'Enter the average solar insolation (in kW/m**2).'
      read(5,*)g
      print*,' '
      print*,' '
      print*,'Enter the efficiency of the PV panels.'
      read(5,*)pveff
      print*,' '
      print*,' '
      print*,'Enter the time of mobility during the day (in hours).'
      read(5,*)tmd
      print*,' '
      print*,' '
      print*,'Enter the time of mobility during the night (in hours).'
      read(5,*)tmn
      print*,' '
      print*,' '
      print*,'Enter the speed for emergency return (km/hr).'
      read(5,*)ve
      print*,' '
```

```
      print*,' '
      return
      end
c
c*********************************************************************
c
      subroutine newval (pa, smr, minspd, maxspd, x, mpcv1, mpcv2,
     &                   meu, msc, na, modtyp, systyp, smtyp, auxtyp)
c
      real mpcv1, mpcv2, meu, msc
c
 4000 write(5,4010)pa, smr, minspd, maxspd, x, mpcv1, mpcv2, meu,
     &               msc, na
 4010 format(1x,'Which value would you like to change?',/,
     &     5x,' 1-Supplemental power above mobility  = ',f8.2,' kWe',/,
     &     5x,' 2-Specific mass of the reactor       = ',f8.2,' kg/kWe'
     &       ,/,
     &     5x,' 3-Rover minimum speed                = ',i8,' km/h',/,
     &     5x,' 4-Rover maximum speed                = ',i8,' km/h',/,
     &     5x,' 5-Rover traction factor modifier     = ',f8.2,/,
     &     5x,' 6-Astronaut independent PCV mass     = ',f8.2,' kg',/,
     &     5x,' 7-Astronaut dependent mass           = ',f8.2,' kg/astr'
     &       ,/,
     &     5x,' 8-Mass of experimental unit          = ',f8.2,' kg',/,
     &     5x,' 9-Mass of the supply car             = ',f8.2,' kg',/,
     &     5x,'10-Number of astronaunts              = ',i8,/,
     &     5x,'11-None',//)
      read(5,*) nval
c
      if (nval.eq.1) then
       print*,'Enter the new power value (in kWe).'
       read(5,*)pa
       go to 4000
      elseif (nval.eq.2) then
       print*,'Enter the new reactor specific mass (in kg/kWe).'
       read(5,*)smr
       go to 4000
      elseif (nval.eq.3) then
       print*,'Enter the new minimum speed (in km/hr).'
       read(5,*)minspd
       go to 4000
      elseif (nval.eq.4) then
       print*,'Enter the new maximum speed (in km/hr).'
       read(5,*)maxspd
       go to 4000
      elseif (nval.eq.5) then
       print*,'Enter the new traction factor modifier.'
       read(5,*)x
       go to 4000
      elseif (nval.eq.6) then
       print*,'Enter the new astronaut dependent mass of the'
       print*,'Primary Control Vehicle (in kg/Astro).'
       read(5,*)mpcv1
       go to 4000
      elseif (nval.eq.7) then
       print*,'Enter the new astronaut independent mass of the'
       print*,'Primary Control Vehicle (in kg/Astro).'
       read(5,*)mpcv2
       go to 4000
      elseif (nval.eq.8) then
       print*,'Enter the new mass of the Experimental Unit (in kg).'
       read(5,*)meu
       go to 4000
      elseif (nval.eq.9) then
       print*,'Enter the new mass of the Supply Car (in kg).'
       read(5,*)msc
```

```
         go to 4000
        elseif (nval.eq.10) then
         print*,'Enter the new number of Astronauts.'
         read(5,*)na
         go to 4000
        elseif (nval.ne.11) then
         print*,'***INVALID ENTRY***'
         go to 4000
        endif
c
        return
        end
c
c***********************************************************************
c
        subroutine sendat()
c
        return
        end
c
c***********************************************************************
c
        subroutine spdave()
c
        return
        end
c
c***********************************************************************
c
        subroutine zeroin()
c
        return
        end
c
c***********************************************************************
c
        subroutine rovpwr()
c
        return
        end
c        read(5,*) chgtyp
c        do 2100, j = 1,3
c          if (chgtyp.eq.1) then
c            pveff = 0.2+float(j-1)*0.05
c          elseif (chgtyp.eq.2) then
c            g = float(j-1)*250.0
c            if (g.eq.0.0) g = 100.0
c          elseif (chgtyp.eq.3) then
c            sma = 1.0+float(j-1)*0.65
c          elseif (chgtyp.eq.4) then
c            na = 2.0+float(j-1)*2.0
c          elseif (chgtyp.eq.5) then
c            ta = 24.0+float(j-1)*72
c          elseif (chgtyp.eq.6) then
c            pls = 1.0+float(j-1)
c          endif
c          do 2050, i = 0, 10
c            ve = float(i)
c        write(40,*)ve, mpv
```

2.1.3

SP-100 THERMOELECTRIC LANDER

ADVANCED SPACE ANALYSIS OFFICE
LEWIS RESEARCH CENTER
CLEVELAND, OHIO

PRESENTED AT OEXP WORKING GROUP MEETING
FEBRUARY 22, 1989

SP-100 THERMOELECTRIC LANDER

STUDY TEAM

- HARVEY BLOOMFIELD/LeRC, POWER TECHNOLOGY DIVISION
- JACK HELLER/LeRC, POWER TECHNOLOGY DIVISION
- MARK HICKMAN/LeRC, ADVANCED SPACE ANALYSIS OFFICE

OBJECTIVES

- DETERMINE FEASIBILITY OF INSTALLATION AND OPERATION OF AN SP-100 NUCLEAR REACTOR WITH THERMOELECTRIC CONVERTERS INTEGRATED WITH LUNAR LANDER FOR LUNAR OUTPOST/BASES
- ESTIMATE POWER SYSTEM MASS AND TOTAL LANDER/REACTOR MASS AT LUNAR SURFACE

SP-100 THERMOELECTRIC LANDER (CONT'D)

APPROACH

- IN-HOUSE STUDY EMPLOYING EXISTING LeRC TOOLS
- OUTGROWTH OF FY 1988 TASK: "SOLAR PHOTOVOLTAIC VERSUS NUCLEAR POWER SYSTEMS FOR LUNAR OBSERVATORY"
  - NOT PART OF FY89 TASKS

SCHEDULE/MILESTONES

- TASK COMPLETED AS OF 2/1/89

PRODUCTS

- ARTIST'S CONCEPTION OF SP-100 THERMOELECTRIC LANDER
- FINAL RESULTS AT FEBRUARY WORKSHOP

SP-100 THERMOELECTRIC LANDER
(CONT'D)

RESULTS

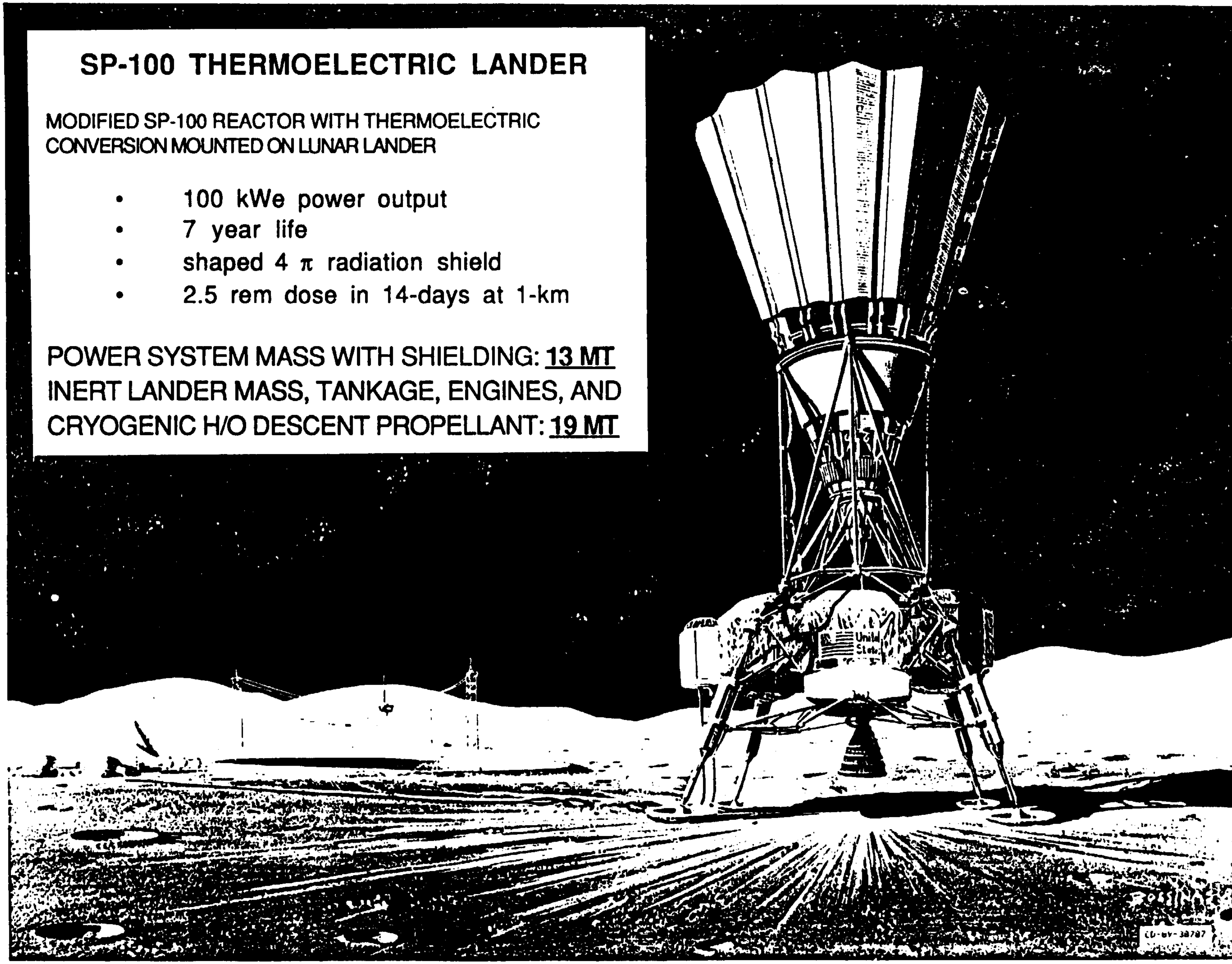
- SP-100 THERMOELECTRIC (T.E.) LANDER WOULD PROVIDE EXTREMELY EXPEDITIOUS DELIVERY OF POWER SUFFICIENT FOR EARLY STAGES OF BOTH LUNAR EVOLUTION AND MARS EVOLUTION CASE STUDIES
  - AMPLE POWER TO SHORTEN LUNAR BASE DEVELOPMENT TIMEFRAME
    - IMMEDIATE POWER SUFFICIENT FOR HUMAN-TENDED PHASE
  - BY ADDING ADDITIONAL LANDERS, BASE POWER SYSTEM IS MODULARIZED
    - FACILITATES BASE GROWTH
  - T.E. LANDER WOULD ENABLE IMMEDIATE ACHIEVEMENT OF FIRST LUNAR EVOLUTION BENCHMARK: FIRST OCCUPATION OF BASE FOR A LUNAR NIGHT
  - REACTOR START UP TO FULL POWER WELL WITHIN LUNAR DAY
  - ONLY 24 HOURS REQUIRED TO THAW OUT FROZEN COOLANT LINES AFTER CONNECTING POWER CABLES AND PAYING OUT TRANSMISSION LINE
  - HIGHER LEVEL OF TECHNOLOGY READINESS THAN OTHER OPTIONS
    - USE OFF-THE-SHELF DOD/NASA/DOE HARDWARE

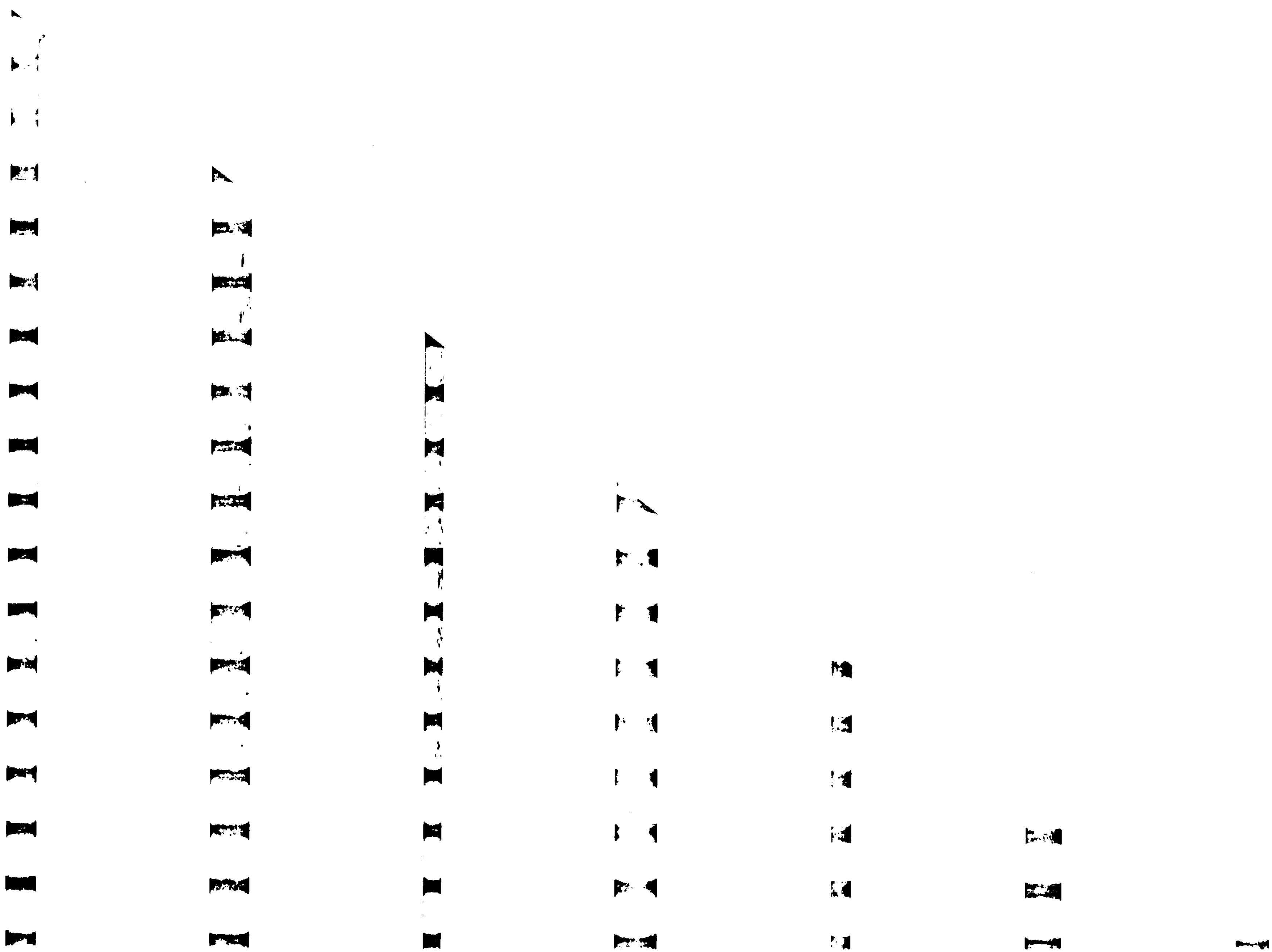
SP-100 THERMOELECTRIC LANDER
(CONT'D)

RESULTS (CONT'D)

- THE T.E. LANDER AS CURRENTLY ENVISIONED WOULD HAVE INTEGRATED RADIATION SHIELDING, INSTRUMENT RATED AT A DISTANCE OF 1 KM
  - MAN-TENDED PRESENCE OF LIMITED DURATION ONLY AT 1 KM
  - POWER SYSTEM MASS COULD BE SIGNIFICANTLY REDUCED WHILE PROVIDING GREATER RADIATION PROTECTION
    - INCREASE POWER SYSTEM DISTANCE TO BASE
    - PROVIDE SOIL RADIATION SHIELDING
- T.E. LANDER REQUIRES NEGLIBLE INFRASTRUCTURE SUPPORT AT LEO

NASA
C-89-80690





2.1.4

MARS/PHOBOS/DEIMOS POWER SYSTEM STUDY

OEXP WORKING GROUP #4
ST. LOUIS, MISSOURI
JULY 13, 1989

J. MARK HICKMAN
ADVANCED SPACE ANALYSIS OFFICE
LEWIS RESEARCH CENTER

OBJECTIVE

TO DEFINE A TRADE SPACE AND PERFORM POWER SYSTEM TRADES ANALYSES FOR THE SURFACE OF MARS, PHOBOS, AND DEIMOS

- PHOTOVOLTAIC POWER SYSTEMS
- NUCLEAR POWER SYSTEMS

PRODUCTS

- FINAL PRODUCTS TO BE COMPLETED BY OCTOBER 89
- DATABASE OF POWER SYSTEM SPECIFIC MASSES
- TRADES TO DETERMINE LIGHTEST WEIGHT POWER SYSTEM FOR VARIOUS SURFACE LOCATIONS AND ENVIRONMENTAL CONDITIONS
- TRADES TO DETERMINE POWER OUTPUT AND POWER DEGRADATION UNDER VARYING CONDITIONS

PRELIMINARY RESULTS

POWER SYSTEM PERFORMANCE AND DESIGN INTERRELATED WITH LOCATION, ATMOSPHERIC CONDITIONS, AND SEASON

- LOCATION
  - LATITUDE/HEMISPHERE AFFECTS:
    - INSOLATION FLUX TO SURFACE (kW/m^2)
    - SINK TEMPERATURE
    - NUMBER OF DAYLIGHT HOURS
- ATMOSPHERIC CONDITIONS
  - DUST STORMS AFFECT:
    - INSOLATION FLUX TO SURFACE (kW/m^2)
    - TYPE OF INSOLATION
      - DIRECT BEAM
      - DIFFUSE
  - CARBON DIOXIDE AFFECTS:
    - TYPE OF MATERIALS AND COMPONENTS SELECTED
- SEASON
  - ORBITAL LOCATION (HELIOCENTRIC LONGITUDE) AFFECTS:
    - INSOLATION FLUX TO THE PLANET (kW/m^2)
    - SPACE SINK TEMPERATURE

ENVIRONMENTAL AFFECTS ON POWER SYSTEM COULD IMPACT LOCATION OF MARS BASE

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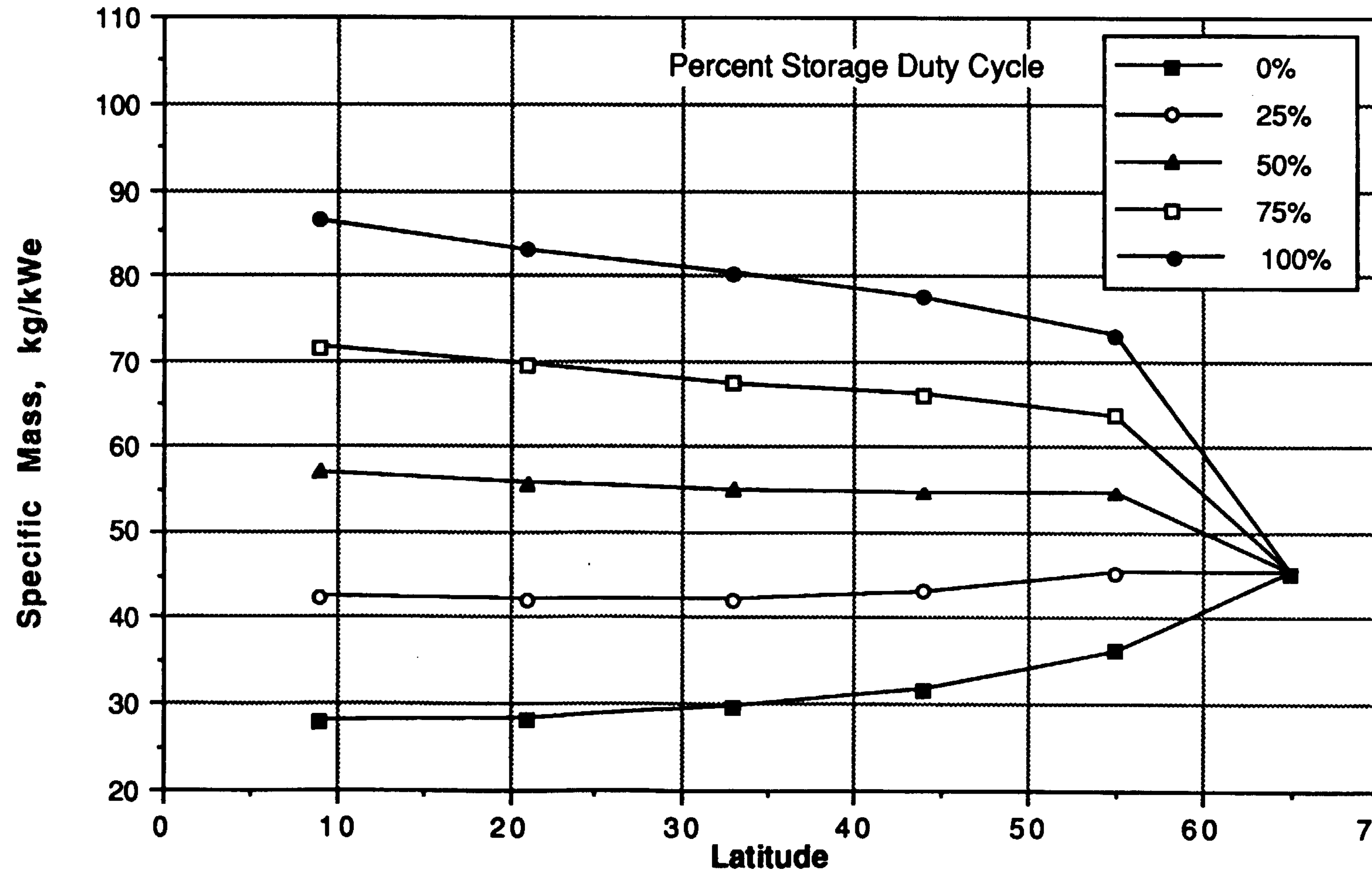
EFFECTS OF LATITUDE AND DUTY CYCLE ON SPECIFIC MASS

MARS NORTHERN HEMISPHERE AT APHELION, LOCAL DUST STORM CONDITIONS

α-Si PV ARRAY POWER SYSTEM WITH REGENERATIVE FUEL CELL STORAGE



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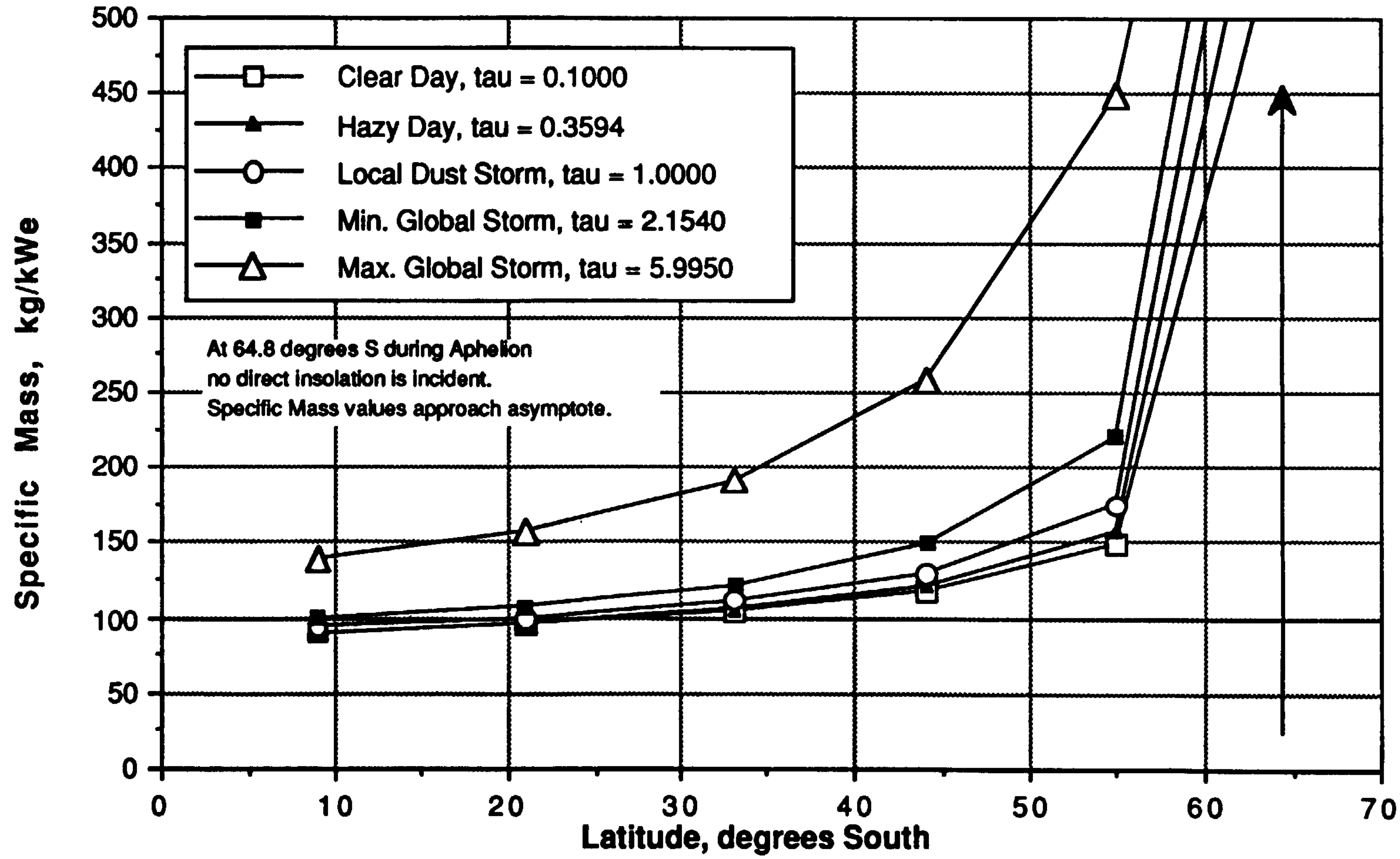
ADVANCED SPACE ANALYSIS OFFICE

NASA — LEWIS RESEARCH CENTER

EFFECT OF LATITUDE AND DUST STORMS ON SPECIFIC MASS

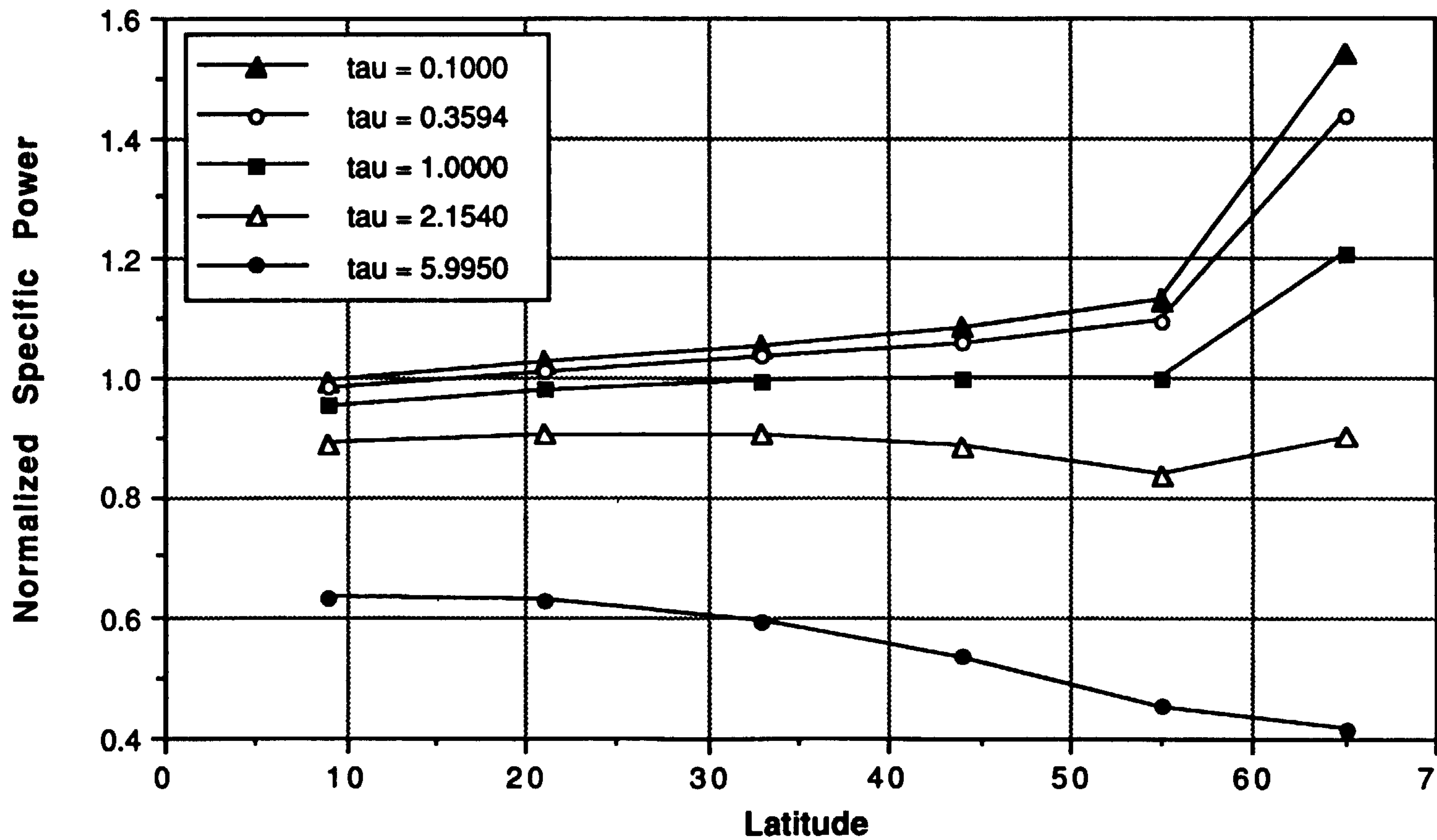
MARS SOUTHERN HEMISPHERE AT APHELION
α-Si PV ARRAY POWER SYSTEM WITH REGENERATIVE FUEL CELL STORAGE
100% DUTY CYCLE



NASA — LEWIS RESEARCH CENTER

EFFECT OF LATITUDE AND DUST STORMS ON POWER OUTPUT

MARS NORTHERN HEMISPHERE AT APHELION
α-Si PV ARRAY POWER SYSTEM WITH REGENERATIVE FUEL CELL STORAGE
DESIGNED FOR LOCAL DUST STORM CONDITIONS AT 44N
WITH 50% DUTY CYCLE (54.539 kg/kWe)



TASKS REMAINING

- INCLUDE NUCLEAR SYSTEMS
- INCLUDE DUST STORM OCCURANCE AND SEVERITY FREQUENCY INTO MODEL
- PERFORM TRADES ANALYSES FOR SEVERAL REFERENCE CONDITIONS
    - DESIGN POWER SYSTEM FOR DUST STORM CONDITIONS ?
    - DESIGN FOR CLEAR CONDITIONS AND ACCEPT DEGREDATION DURING DUST STORMS ?

FUTURE TASKS

- FUTHER INVESTIGATION OF ENVIRONMENTAL CONDITIONS ON POWER SYSTEM
  - DUST
    - LIGHT SCATTERING
    - ABRASION
    - COVER
  - CO_2
    - CHEMICAL REACTIONS WITH POWER SYSTEM COMPONENTS AND MATERIALS
- POWER SYSTEM CONCEPTUAL DESIGNS
  - PHOTOVOLTAIC
  - NUCLEAR

2.1.5

Design Considerations for Lunar Base Transmission Systems

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Introduction

There are a number of subsystems for a lunar base power system, including power sources, energy storage, energy transmission, and loads. Each of these have a number of issues, such as cost, ease of transportation and assembly, current technology feasibility, interaction with the environment, reliability, etc. All of the above subsystems interact heavily with each other. In other words, the overall system design requirements and constraints must be taken into consideration for the final and detailed design of any subsystem. Optimum utilization of power at a lunar base will include the distribution and utilization of thermal energy as well as electrical energy. The following discussion includes primarily information on possible design options for electrical lunar power transmission only. We must keep in mind that later overall system considerations will strongly effect these design options. For instance, thermal energy production, transmission, and utilization will not be discussed in any detail in this brief report.

Design Considerations

A number of initial decisions must be made for a lunar base power/energy transmission system before detailed power system analysis and design can be conducted. First the method of transmission (for electrical and thermal energy) must be chosen. This will be effected by the transmission distances, source characteristics, and environmental conditions. After selection of the transmission type, parameters such as voltage, current, and waveshape must be determined. Finally, design details, such as geometry, conductor size and type, and insulation, must be selected. Most of the information needed for these design considerations is from source power output characteristics, load characteristics, and distance constraints between sources and loads. For this discussion little detailed information is known about the load (for example its operating voltage and current, or duty cycle) and thus only general observations/suggestions can be made at this time.

Electrical transmission between the zones will most effectively take the form of transmission lines using some number of spaced conductors held apart and insulated by some form of dielectrics. (For later considerations power beaming may prove to be effective for certain applications such as powering lunar rovers). Topics which need to be discussed include voltage and current; waveshape, polarity, and frequency; and geometry, conductor material, and insulation type.

Voltage and Current

The voltage and current requirements of a transmission line are determined by three factors: the output characteristics of the sources, the load requirements, and the power to be transmitted. Certainly the source output characteristics and load requirements will effect the choice of transmission voltage, current, and waveshape. But since we don't presently have all of this information, we will assume, in general, that the transmission line characteristics will be chosen primarily to optimize the efficiency and reliability of transmission over the required distances in the lunar environment. Also assume that the source output power characteristics can be transformed to match the transmission parameters, and the transmission line output can be transformed to match the load requirements. In later considerations the cost and complexity of this transformation at each end must be taken into consideration.

Once the required power is known (1 kW, for example) then the voltage and current can be chosen such that they meet this power requirement. Obviously the higher the transmission voltage, the lower the required line current and vice versa. The dominant factor in voltage/current selection is the selection's effect on thermal losses ($P_{thm}=I^2R$). A decision to operate at high voltages (i.e. low currents) will, of course, result in a decrease in thermal losses, and a high current (low voltage) will increase the thermal losses. The effect the voltage has on the system's efficiency (which is inversely proportional to thermal losses) is illustrated in Fig. 1, and Fig. 2. The efficiencies were calculated for a temperature of -30 C (average of peak day and peak night temperatures), a power requirement of 800 kW, and a length of 5 km. This represents the power link between the reactor and oxygen mining facility.

V vs eff. for Conductor Radius = .1m

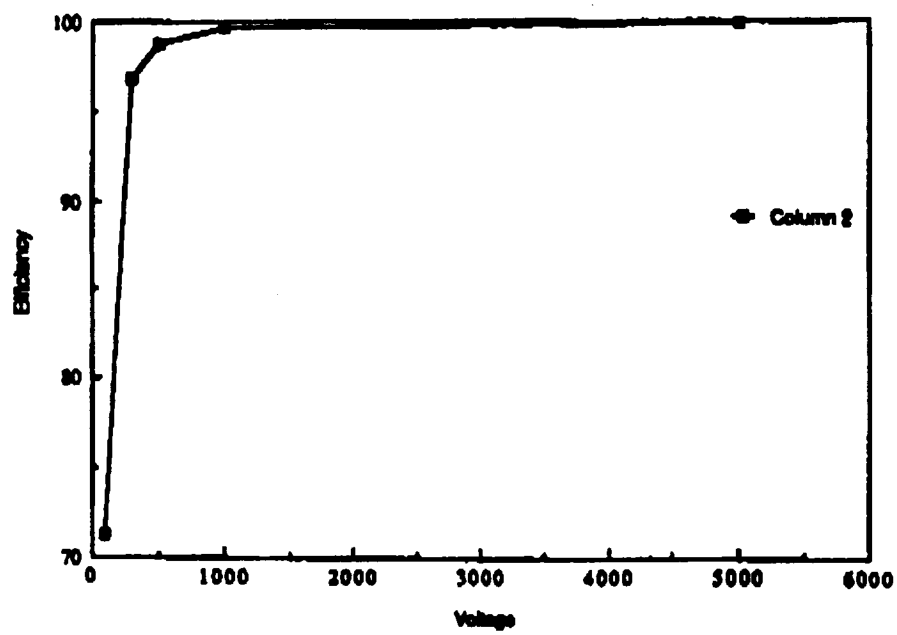

Fig. 1

V vs eff. for Conductor Radius = .05 m

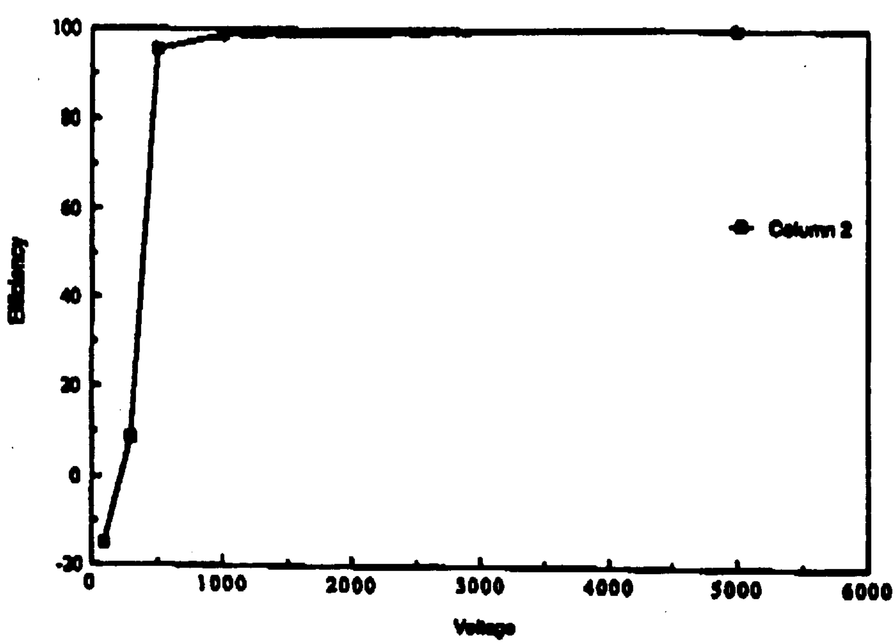

Fig. 2

3.5

The voltage and current will also effect other power issues. For example, high voltages not only increase the effects of ac impedence, but also increase the required insulator thickness. The voltage must be selected to trade off thermal losses with electromagnetic losses. As shown in Fig. 1 and Fig. 2, optimum voltage ranges for minimize power losses can be found knowing the power required, the transmission distances and selecting a conductor thickness.

Power Waveform (AC or DC)

Three phase ac current flow is the primary method of transmission on Earth mainly because of economical considerations. However dc current flow's advantages are strong enough that it should be seriously considered for lunar applications. For the same amount of power transmitted over the same size conductors, line losses (neglecting skin effect) for ac is 33% greater than for dc.(1) The dc line will have two conductors instead of 3 (if 3 phase is used) and need less insulation, if any (see dielectric section: Dc is preferred when vacuum insulation is used because ac will create inductances and capacitances and lower the breakdown voltage). Ac resistances are greater than dc due to skin effect (the tendency of current to "gather" near the surface of a conductor thus decreasing efficiency). Switching surges on dc lines are lower than on ac lines. And lastly, dc transmission will not interfere with radio communications.

The difficulties of dc are found mainly in the required transformers and converters for voltage conversion. Both devices are costly and have their operating drawbacks. The converters have little overload capacity and produce harmonics on both sides, thus requiring filters to limit the interference. Voltage transformation is difficult which may require the load operating voltage to be less than the line voltage. Other disadvantages include a lack of dc circuit breakers, and finally induction motors are simpler, more rugged and cheaper than dc motors.

The increase in cost of the devices can be partially offset by the decrease in weight and power losses. The cost and weight will also be

decreased by the fact that the SP-100 thermoelectric and PV/RFC are both dc sources; a converter would therefore only be needed at the load end (if it required ac current or a voltage transformation). As interest in dc transmission grows improvements in converter, transformer, and dc motors have already begun. (Thyristor convertors for high voltage and high current applications, and prototype circuit breakers have been developed).

The merits of dc transmission make it a viable option for the lunar base.

<u>Frequency</u>

While dc transmission is attractive three phase ac (single phase is not economical for high power transmission but may be used for lower power systems) is still a contender because of its low cost conversion devices and ease of transformation. Therefore, an optimum operating frequency range must be found. An increase in frequency will result in an increase in the power/weight ratio, because of the decrease in size of the components (conversion/inversion equipment). A general scaling technique for components is an increase in frequency of N^2 corresponds to a reduction in linear dimension by a factor of N.(2)

However, an increase in frequency also results in a number of negative effects. Such as an increase in leakage reactances for both machines and transmission lines, increased hysterisis loses, an increase in radiation, an increase in thermal losses, and greater stress on the insulating solid dielectrics.

An estimate of an optimum frequency cannot be made at this time.

<u>Conductor Material</u>

The two most commonly used conductors are copper and aluminum. Despite coppers larger electrical conductivity the weight advantage of aluminum (approximately 30% lighter than copper) makes it the preferred choice for lunar applications.

<u>Dielectric Insulation</u>

Air is used to insulate the transmission lines on earth; therefore, another form of insulation will be needed on the moon. Because of

possible leaking of gas and liquid dielectrics , and because of the extra weight of a liquid dielectric, a solid dielectric would appear to be the only other option. However, using the vacuum environment as the method of insulation should also be considered. A good vacuum (less that 10E-04 Torr) has been shown to be an effective method of insulation. The moons extreme vacuum of
10E-09 Torr day and 10E-12 Torr night is well within the required pressure levels to prevent breakdown. This method of insulation would eliminate insulator weight and increase thermal dissipation.

Underground Cable

Burying the transmission line would appear to be a safe and convenient option. However, the thermal conductivity of the lunar soil is low enough that it may cause the line to overheat.(3) Therefore it would be better at this time to plan for above ground transmission until a detailed analysis can be done.

Thermal Considerations

While the previous discussions have delt exclussively with electrical energy, thermal energy collection/distribution should not be overlooked. Waste heat can be used directly or stored for future use. Possible methods of thermal transmission include heat pipes and closed loop thermal fluid flow.

As an example, a SP-100 thermoelectric reactor operates at .051 efficiency. Therefore, for a thermal input of 2.21 Mw the rejected power is 2.10 Mw.(4) Most of this power is in the form of thermal energy. This energy can be collected and then transported to zones which require thermal power in addition to electrical (such as the habitat and ISRU). The required thermal input for hydrogen reduction of illeminte is approximately 1000 C (5)while the radiator temperatures of the SP-100 thermoelectric reactor fall between 800 C and 1000 C.(4) Therefore because of radiative losses electrical energy will have to be converted to thermal or perhaps solar thermal collectors can be placed near the ISRU. It should also be noted that the stirling power plant (operating at 50%

efficiency) will reject only 1.1 Mw of power, therefore it will be less effective as a heat source. This may influence the placement of the thermoelectric lander to make the best of its inefficiencies.

Design Example #1

type	dc
geometry	2 parallel plates, 30 cm wide by 2 cm thick
conductor material	aluminum
insulation	vacuum
voltage	2000 V
current	400 A
thermal loss	460 W per km
line efficiency	99.7 %
location	on or suspended slightly above the ground

Design Example #2

An example of a possible ac transmission line design would be a three phase line insulated with solid dielectrics. The operating voltages and frequencies can not be determined at this point. Information necessary for their selection, such as inductances of the line, dielectric material, and conductor geometry are not know and are not easily determined without further evaluation of the power losses. Also details of ac design must wait for further load and transmission characteristics to be determined.

Conclusion

A number of parameters of electrical power/energy transmission have been discussed and when appropriate recommendations have been made. A summery of these guidelines is illustrated in the form of two design examples. They should be considered very preliminary. They were made with limited detailed information and therefore represent a significant amount of speculation. Any of them would likely change as more indepth research continues.

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2.1.6

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SP-100 REACTOR SCALABILITY STUDY
FINAL BRIEFING PACKAGE

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APPLIED TECHNOLOGY

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SP-100 REACTOR SCALABILITY STUDY

The purpose of the SP-100 Reactor Scalability Study was to show that SP-100 reactor technology can be scaled to thermal power levels required for Nuclear Electric Propulsion. A power range of 10 to 50 MWt was to be evaluated for a 10 year mission lifetime. Control of the reactor was to be accomplished by the motion of hinged or sliding reflector elements. Fuel burnup was limited to 10 atom percent. Other reactor safety and performance criteria were established as appropriate for this application.

The tools normally used in such a reactor concept development and evaluation study are COROPT, TWODANT and MCNP. COROPT is a SP-100 nuclear subsystems optimization code. COROPT uses first principles to estimate the size and mass of the reactor, shield, primary heat transport and reactor I&C subsystems. TWODANT is a multi-group transport code used to confirm COROPT design and performance predictions. In this study, Keff, reflector control worth and shield performance values were computed with TWODANT. MCNP is a Monte Carlo Neutron/Photon code which is used, as necessary, to confirm key TWODANT values. MCNP was not needed in this study.

The main features of the reactor are similar to SP-100. The Li coolant enters and exits the reactor vessel in a coolant inlet/outlet plenum at the aft end of the vessel. Inlet flow travels forward in a bypass flow region between the reactor core and vessel until reaching the forward coolant plenum. At this point the Li coolant flows aft through the reactor core cooling the fuel pins until again reaching the coolant inlet/outlet plenum and exiting the reactor vessel. The radial reflector is located outside the reactor vessel.

The approach taken in this study was to establish the main features of a 10 MWt low mass reactor and shield, consistent with the study guidelines, and identify how those features change as the reactor power is scaled up to 50 MWt. As expected, the reflector control worth decreases as the reactor

size and power increases. Hinged reflector control does not appear to satisfy the control requirements even with thick radial reflectors at 10 MWt. However, a relatively thin radial reflector (~7.6 cm) can be used at 10 MWt if sliding reflector panels are used for control. This thin reflector appears to satisfy control requirements up to about 25 MWt. Between 25 and about 40 MWt reflector thickness needs to increase to meet control requirements. At 40 MWt a thick (15.2 cm) radial reflector is needed. Increasing reflector thickness above 15.2 cm does not significantly improve control worth since infinite effective reflector thickness is being approached. Thus, to scale the reactor design above 40 MWt using reflector control, the control requirement must be reduced. This is accomplished by increasing the mass of fuel to reduce burnup and, hence, the burnup reactivity requirement. At 50 MWt, a reduction of nearly 1 atom percent on the average fuel burnup is needed to sufficiently lower control requirements. This results in an increase of the combined reactor shield mass of about 1500 Kgs.

The design approach imposed by the study requirement of reflector control leads to a scaling situation such that, for a 10 year mission, two 25 MWt reactors have lower mass than one 50 MWt reactor. The use of in-core control would allow higher fuel burnup and a thinner reflector resulting in substantial mass savings (1500-3000 Kgs) at 50 MWt.

A key SP-100 design feature examined in the scalability study was passive cooling of the radiation shield. TWODANT analysis of the 50 MWt shield design showed that passive cooling is possible if the Be conduction plate thickness is increased to ~12 cms (which moves the forward edge of the LiH further back into the shield) and the forward LiH is made of depleted Li^7H. These changes to the shield design reduce the peak heating in the forward LiH to ~0.1 watts/cm^3 (~60% of the peak SP-100 heating), assuring that passive cooling is possible.

The SP-100 reactor scalability study shows that the SP-100 technology can be scaled up to 50 MWt meeting all study constraints. The resulting base case designs yield a combined reactor/shield mass of 3,650 Kgs at 10 MWt and 16,150 Kgs at 50 MWt.

Several sensitivity studies were performed to evaluate the mass impact of key study constraints, i.e., reactor outlet temperature, shield half cone angle, mission length, and safety requirements. The 50 MWt design for a 3 year mission is much smaller (lower mass) than the corresponding 10 year design (7,725 vs. 16,150 Kgs). The significantly lower mass results from lower burnup and control requirements of the 3 year mission which permits substantially increased core performance. The mass impact of designing for launch of a completely assembled reactor system (containing in-core safety rods and a re-entry cone) was estimated to be less than 4,000 Kgs at 50 MWt. The mass impacts of 50 MWt designs containing 7 and 13 in-core safety rods were estimated at 2,650 and 4,280 Kgs, respectively. Nine or 10 safety rods would probably be required to satisfy re-entry safety requirements.

The conclusion of this study is that SP-100 reactor and shield technology is scalable to 50 MWt for long life (10 yr) missions needed for Nuclear Electric Propulsion. While additional key feature tests would be needed for scaled-up or modified equipment and for extension of the fuel/materials data base, no technology development is required beyond the scope of the SP-100 Ground Engineering System Program.

FINAL BRIEFING

SP-100 REACTOR SCALABILITY STUDY

CONTRACT NO. DE-AC03-89SF17787
TASK ASSIGNMENT NO. 89-1

AUGUST 2, 1989
(REVISED)

APPLIED TECHNOLOGY

AGENDA

- OBJECTIVE AND APPROACH
- REQUIREMENTS
- ANALYTICAL TOOLS
- DESIGN DESCRIPTION
- PERFORMANCE ANALYSIS
- SENSITIVITY ANALYSIS
- DEVELOPMENT
- CONCLUSIONS
- FOLLOW ON WORK

SP-100 REACTOR SCALABILITY STUDY

OBJECTIVE AND APPROACH

OBJECTIVE	-	SCALE SP-100 TYPE REACTOR TO 10-50 MW_T POWER RANGE FOR NUCLEAR ELECTRIC PROPULSION MISSION
APPROACH	-	ESTABLISH 10 MW_T MINIMUM MASS DESIGN WITHIN STUDY REQUIREMENTS
	-	SCALE TO 50 MW_T
	-	EVALUATE MASS AND PERFORMANCE

REQUIREMENTS

CONCEPT	1	2
MISSION REQUIREMENTS		
LIFETIME, YEARS	10	10
REACTOR POWER LEVEL, MW_T	10	50
REACTOR ASSEMBLY	IN SPACE STATION ORBIT	
REACTOR STARTUP	IN "NUCLEAR SAFE ORBIT"	
MISSION DEFINITION	REUSABLE/RETURN TO HIGH EARTH ORBIT	
CONFIGURATION	MINIMUM MASS	
SAFETY REQUIREMENTS		
END OF MISSION SHUTDOWN	NO	NO
INTACT REENTRY	NO	NO
LOSS OF COOLANT MITIGATION	NO	NO
ARMORING FOR SPACE DEBRIS	NO	NO
PAYLOAD DEFINITION		
DIAMETER, M	20	20
SEPARATION DISTANCE, M	100	100
NEUTRON DOSE LIMIT, 10^{13} NVT	1	1
GAMMA DOSE LIMIT, 10^5 RAD	5	5

REQUIREMENTS

CONCEPT	1	2
REACTOR		
SCOPE OF DEFINITION	MASS/VOLUME SCALING	
MAX. COOLANT TEMPERATURE, K	1350	1350
COOLANT TEMPERATURE RISE, K	110	110
FUEL BURNUP, A/O	<10	<10
PRESSURE DROP, MPA	WILL BE CALCULATED	
SHIELD		
SCOPE OF DEFINITION	MASS SCALING	
PRIMARY HEAT TRANSPORT		
SCOPE OF DEFINITION	PARTIAL INTERFACE DESIGN DATA ONLY	
NUMBER OF LOOPS	2	2
FLOW RATE, KG/SEC	WILL BE CALCULATED	
REACTOR I&C		
SCOPE OF DEFINITION	PARTIAL INTERFACE DESIGN DATA ONLY	
REACTIVITY CONTROL/SHUTDOWN	EX-CORE REFLECTOR	
ACTUATOR DRIVES	WILL BE INCLUDED IN MASS SCALING	

REACTOR CONCEPT ANALYSIS METHODS

COROPT - REACTOR SYSTEM OPTIMIZATION CODE

- PRELIMINARY ESTIMATE OF REACTOR/SHIELD DESIGN PARAMETERS
- ESTIMATE OF REACTOR/SHIELD MASS

TWODANT - TWO DIMENSIONAL MULTI-GROUP TRANSPORT CODE

- CONFIRMATION ANALYSIS OF COROPT RESULTS

MCNP - MONTE CARLO NEUTRON/PHOTON CODE

- CONFIRMATION ANALYSIS OF TWODANT AND COROPT RESULTS AS APPROPRIATE

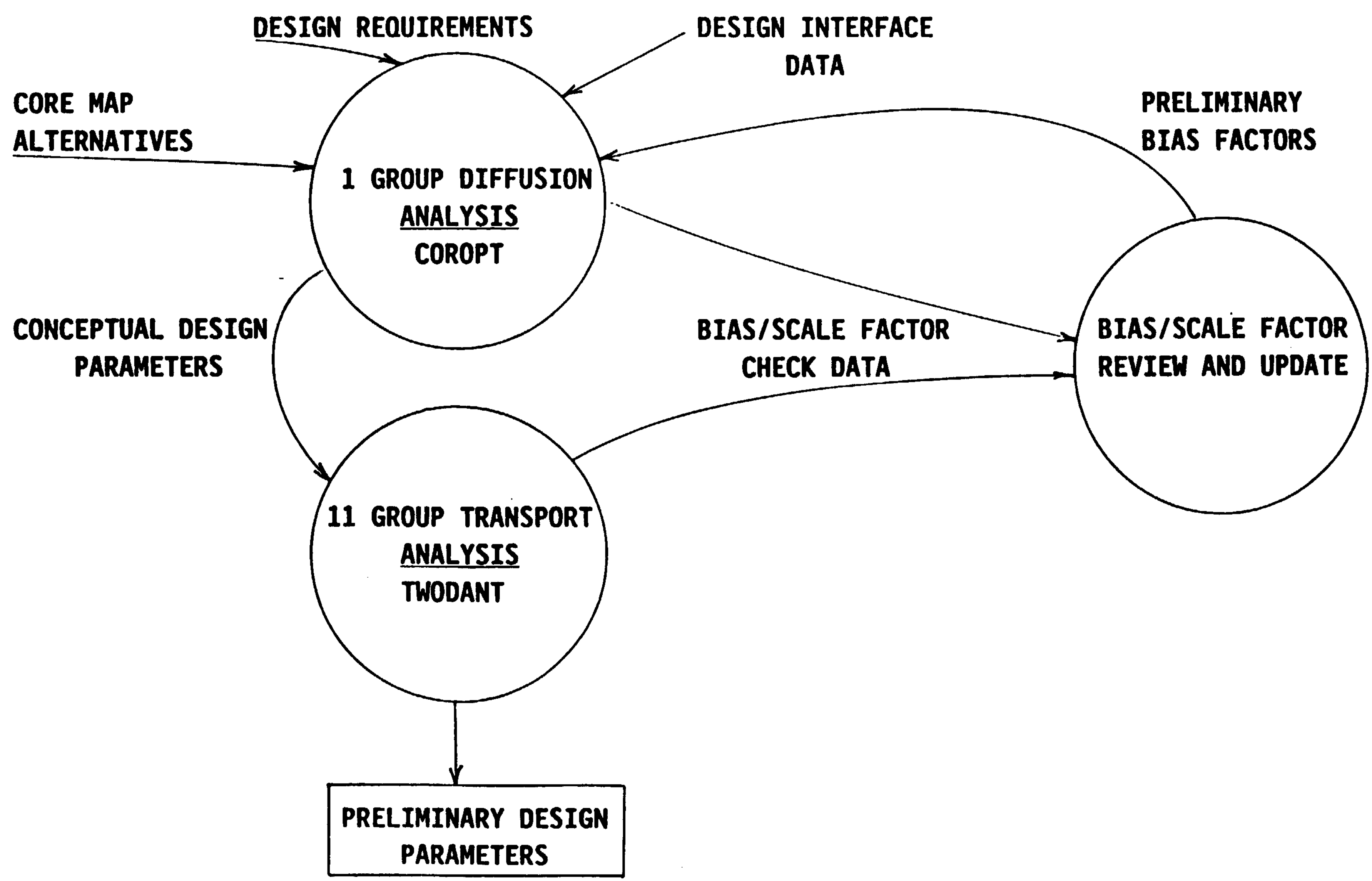
COROPT-S CODE CAPABILITIES

- SP-100 NUCLEAR SUBSYSTEMS OPTIMIZATION CODE
- USES FIRST PRINCIPLES TO SIZE REACTOR, SHIELD, PRIMARY HEAT TRANSPORT AND REACTOR I&C SUBSYSTEMS
- DEMONSTRATED CAPABILITY TO SCALE SP-100 CONCEPT FOR A WIDE RANGE OF REQUIREMENTS AND CONCEPTS

TYPICAL OPTIMIZED PARAMETERS	CONSTRAINTS
CORE HEIGHT	PEAK BURNUP
PIN DIAMETER	F.G. PRESSURE STRAIN
PIN P/D RATIO	FCMI+FG PRESSURE STRAIN
CLADDING THICKNESS	MAX FUEL TEMP
PLENUM LENGTH	REACTOR ΔP
REFLECTOR THICKNESS	REFLECTOR WORTH
FUEL PELLET/CLADDING GAP	

- SIZES AND PROVIDES MASS ESTIMATES IN SENSITIVITY OR OPTIMIZATION MODE

REACTOR AND SHIELD DESIGN STUDY APPROACH

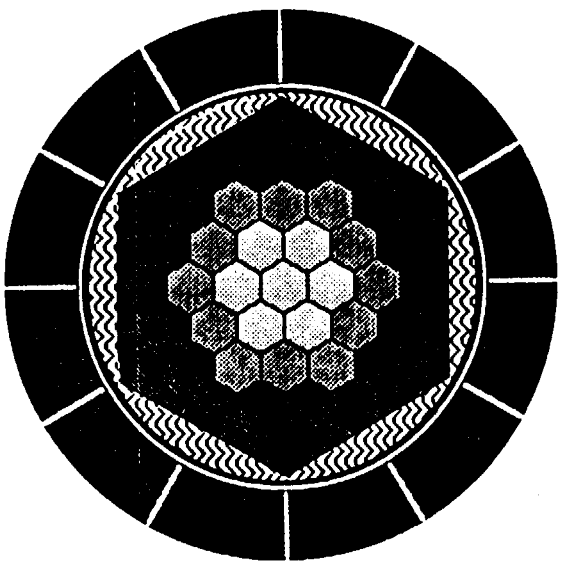


SP-100 REACTOR SCALABILITY STUDY

CORE CROSS SECTION SCHEMATIC

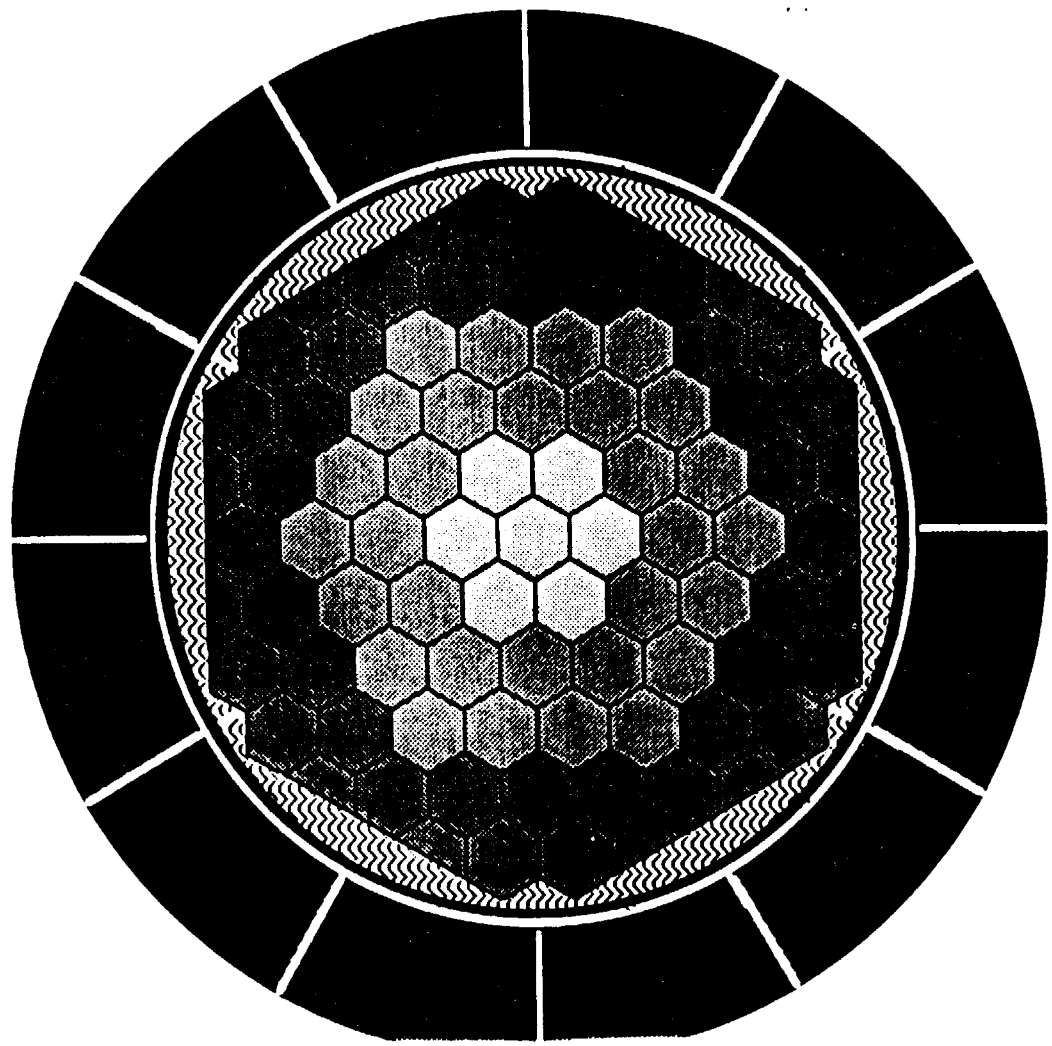
10MW

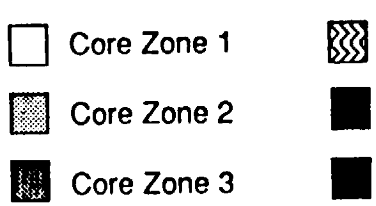
Number of full assemblies = 43
Number of partial assemblies = 12
Assembly pitch = 5.18 cm
Vessel outer radius = 21.6 cm
Reflector outer radius = 29.4 cm



50MW

Number of full assemblies = 73
Number of partial assemblies = 12
Assembly pitch = 8.09 cm
Vessel outer radius = 40.9 cm
Reflector outer radius = 56.4 cm

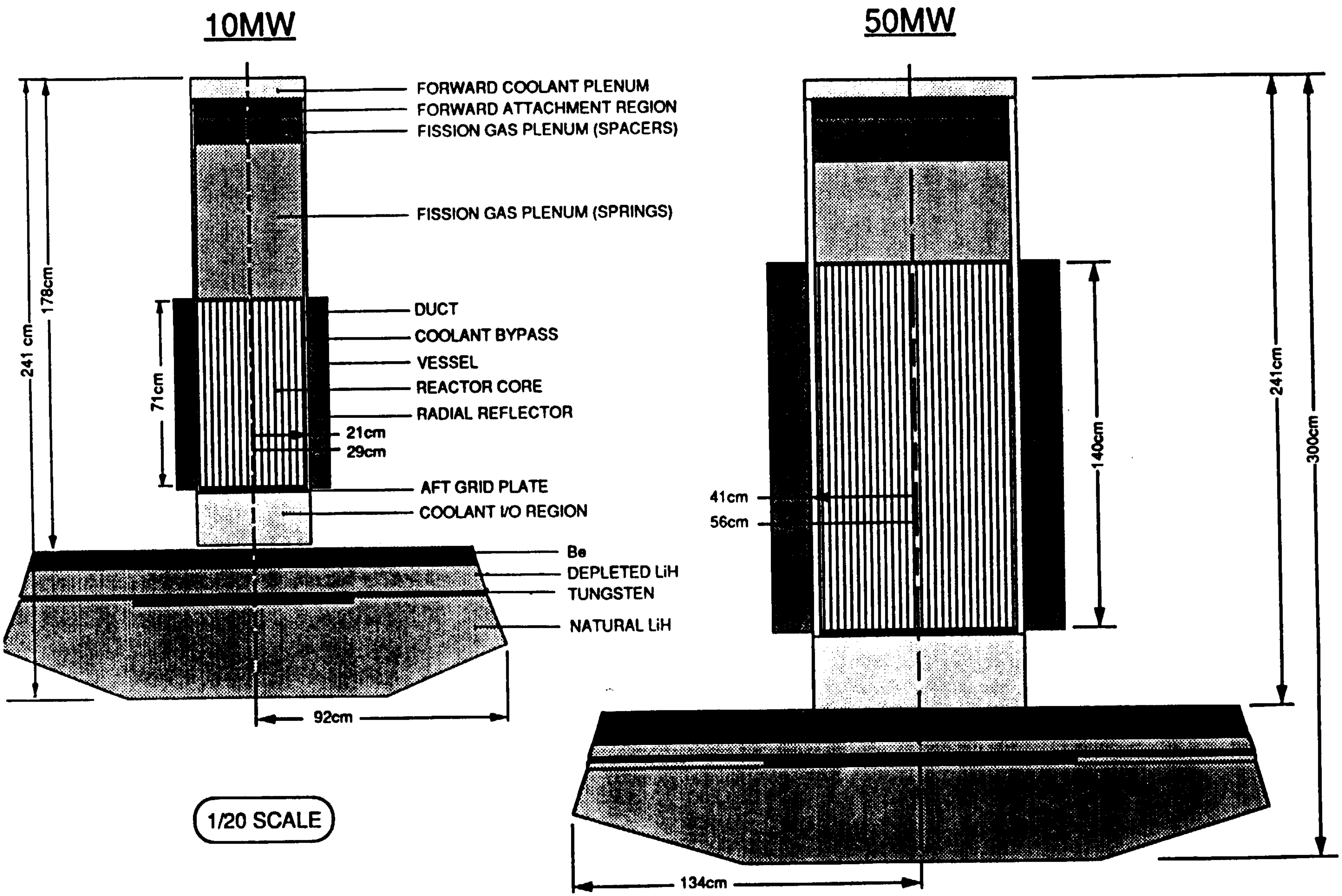




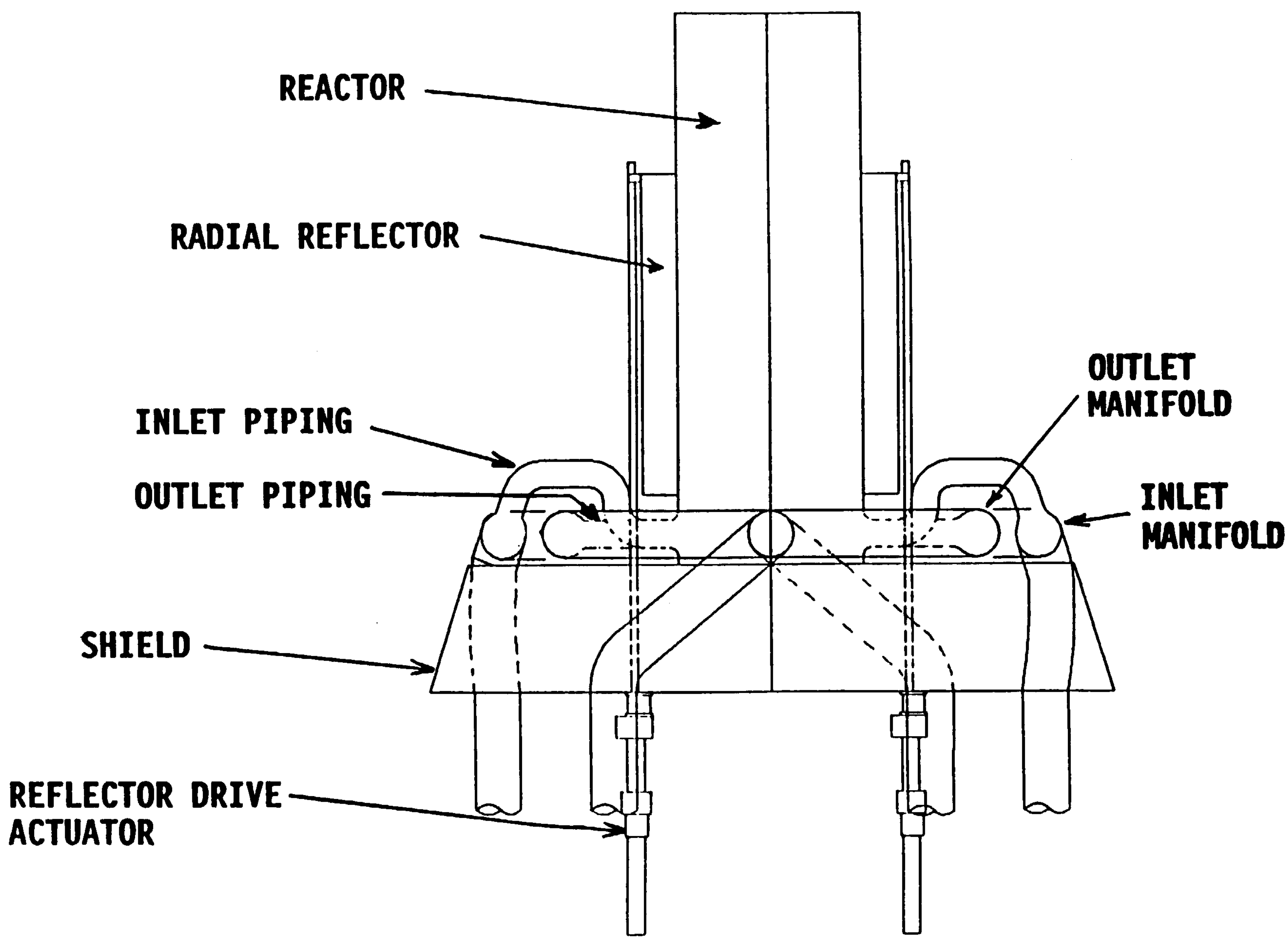
1/8 SCALE

SP-100 REACTOR SCALABILITY STUDY

REACTOR R-Z SCHEMATIC

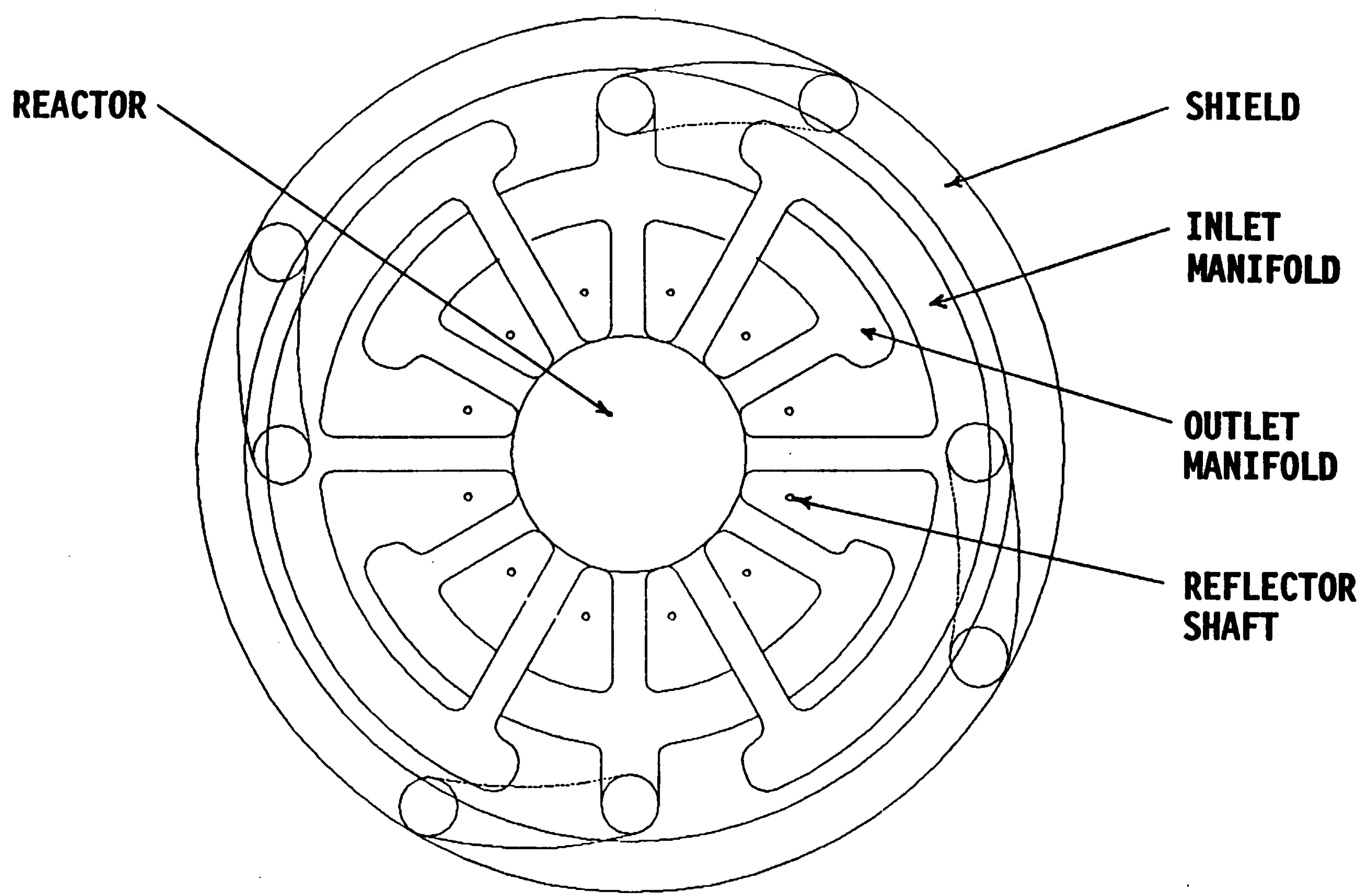


50 MW_T REACTOR ARRANGEMENT ELEVATION VIEW



SP-100 REACTOR SCALABILITY STUDY

50 MW_T REACTOR ARRANGEMENT PLANVIEW



SP-100 REACTOR SCALABILITY STUDY

KEY REACTOR DESIGN FEATURES

	10 MWT	50 MWT
REACTOR VESSEL OUTER DIAMETER (IN)/(CM)	17.0/43.2	32.2/81.8
REACTOR VESSEL LENGTH (IN)/(CM)	69.9/177	95.1/241
REACTOR REFLECTOR DIAMETER (IN)/(CM)	23.2/58.8	44.4/113
VESSEL WALL THICKNESS (IN)/(CM)	.120/.305	.328/.833
REFLECTOR THICKNESS (IN)/(CM)	3.0/7.62	6.0/15.2
REACTOR MATERIAL	PWC-11	PWC-11
REFLECTOR CONTROL CONCEPT	SLIDING	SLIDING
REFLECTOR MATERIAL	BeO	BeO

KEY CORE DESIGN PARAMETERS

		10 MW_T	50 MW_T
CORE HEIGHT (IN)/(CM)		28.1/71.4	55.0/140
ASSEMBLY HEXAGONAL PITCH (IN)/(CM)		2.04/5.18	3.18/8.09
NUMBER FUEL ASSEMBLIES	FULL	43	73
	PARTIAL	12	12
	TOTAL	55	85
NUMBER FUEL PINS/ASSEMBLY	FULL	37	91
	PARTIAL	22	51
TOTAL NUMBER OF PINS		1855	7255
CORE ENVELOPE DIAMETER (IN)/(CM)		16.6/42.2	31.4/79.8
CORE HEIGHT TO EQUIVALENT DIAMETER RATIO		1.85	1.84
FUEL PIN PITCH TO DIAMETER RATIO		1.06	1.061
FISSION GAS PLENUM LENGTH (IN)/(CM)		27.5/69.8	22.0/55.9
CORE STRUCTURAL MATERIAL		PWC-11	PWC-11

KEY PIN DESIGN PARAMETERS

	10 MWT	50 MWT
PIN DIAMETER (IN)/(CM)	.301/.765	.305/.775
CLADDING THICKNESS (IN)/(CM)	.013/.033	.013/.033
CLADDING MATERIAL	PWC-11	PWC-11
CLADDING LINER THICKNESS (IN)/(CM)	.004/.010	.004/.010
LINER MATERIAL	RE	RE
WIRE WRAP DIAMETER (IN)/(CM)	.015/.038	.015/.038
WIRE WRAP PITCH (IN)/(CM)	5.4/13.7	5.4/13.7
WIRE WRAP MATERIAL	PWC-11	PWC-11
FUEL	UN	UN

KEY SHIELD DESIGN PARAMETERS

	10 MWT	50 MWT
SHIELD LAYER SEQUENCE/MATERIAL	Be/LiH/W/LiH	Be/LiH/W/LiH
CONE HALF ANGLE (DEGREES)/(RADIANS)	18/31.4	18/31.4
SEPARATION DISTANCE (FT/M)	328/100	328/100
SHIELD STATION DISTANCE (IN)/(CM)	98.4/249	162/411
LAYER THICKNESSES		
Be (IN)/(CM)	2.8/7.1	4.8/12.1
LiH (FORE) (DEPLETED) (IN)/(CM)	3.9/10.0	2.0/5.0
W (IN)/(CM)	1.3/3.3	1.8/4.6
LiH (AFT) (NATURAL) (IN)/(CM)	16.9/42.8	14.5/36.9
TOTAL THICKNESS (IN)/(CM)	24.9/63.2	23.1/58.6

PRIMARY HEAT TRANSPORT SYSTEM INTERFACE DATA

		10 MWT	50 MWT
PRIMARY SYSTEM MASS FLOW RATE (LBM/HR)/(KG/SEC)		$1.745 \times 10^{+5}$/21.98	$8.72 \times 10^{+5}$/109.9
PRIMARY SYSTEM PRESSURE* (PSI)/(MPA)	BOL	18.5/.13	22.5/.16
	EOL	51.4/.35	81.3/.56
REACTOR PRESSURE DROP (PSI)/(MPA)	BOL	4.60/.032	10.4/.072
	EOL	4.65/.032	10.4/.072
NUMBER OF PIPES	INLET	2	2
	OUTLET	2	2
PIPE INSIDE DIAMETER(IN)/(CM)		4.0/10.2	8.0/20.3
PIPE WALL THICKNESSES(IN)/(CM)		.040/.102	.121/.307
COOLANT		LI	LI

*ASSUMED

REACTOR I&C INTERFACE DATA

	10 MW_T	50 MW_T
NUMBER OF REFLECTOR CONTROL ELEMENTS	12	12
NUMBER OF CONTROL DRIVES	12	12
CONTROL DRIVE ENVELOPE, CM		
DIAMETER	11	13
LENGTH	64	77

CORE PERFORMANCE CHARACTERISTICS

		$10\ MW_T$	$50\ MW_T$
BURNUP (AT/O)	AVG	7.0	4.3
	PEAK	9.7	6.2
EOL LINEAR POWER (kW/FT)/(W/CM)	AVG	2.3/75	1.5/49
	PEAK	3.2/105	2.1/69
EXCESS REACTIVITY REQUIREMENT (ΔK)			
	TEMP. EFFECT	.022	.022
	FUEL BURNUP	.081	.049
	FUEL SWELLING	.027	.020
	EOL MARGIN	.000	.000
	TOTAL	.130	.091
REFLECTOR WORTH (ΔK)		.197	.109
PEAK FAST FLUENCE (N/CM^2)		1.22×10^{23}	1.66×10^{23}

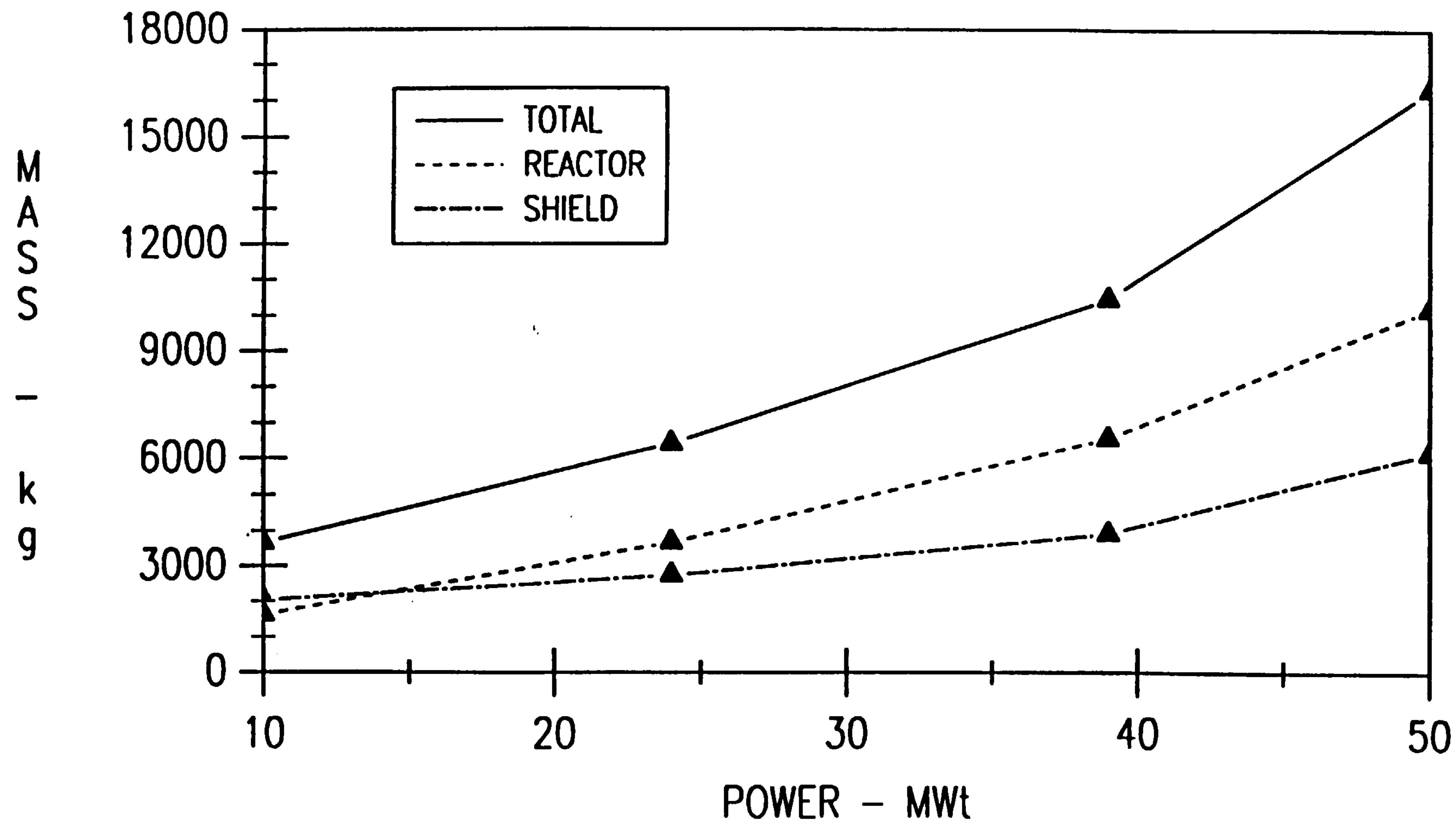
SP-100 REACTOR SCALABILITY STUDY

MASS (KG) BREAKDOWN SUMMARY

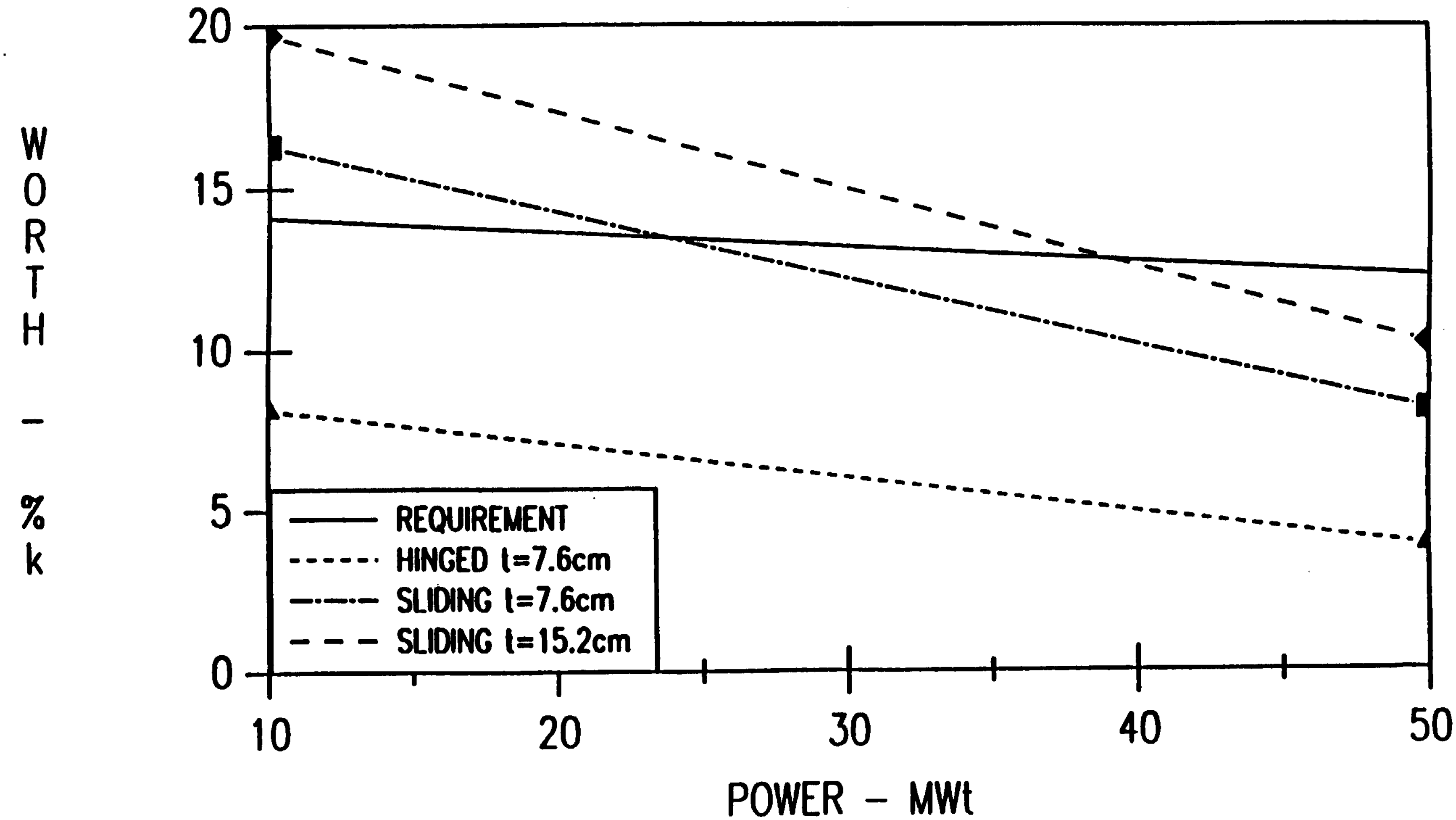
		10 MWT	50 MWT
REACTOR			
	FUEL	580	4,700
	STRUCTURES	980	5,220
	COOLANT	60	250
	SUBTOTAL REACTOR	1,620	10,170
SHIELD			
	STRUCTURE	250	630
	BE	580	1,640
	LIH (DEPLETED - COMPOSITE)	140	200
	LIH (NATURAL)	520	1,230
	W	540	2,280
	SUBTOTAL SHIELD	2,030	5,980
	SUBTOTAL REACTOR + SHIELD	3,650	16,150
PHTS (PARTIAL)			
	MANIFOLDS	20	170
	PIPING THRU SHIELD	10	130
	COOLANT	50	280
	SUBTOTAL PHTS	80	580
RI&C (PARTIAL)			
	REFLECTOR DRIVE UNITS	90	160
	REFLECTOR DRIVE LINES	10	10
	SUBTOTAL RI&C	100	170

SP-100 REACTOR SCALABILITY STUDY

REACTOR AND SHIELD MASS

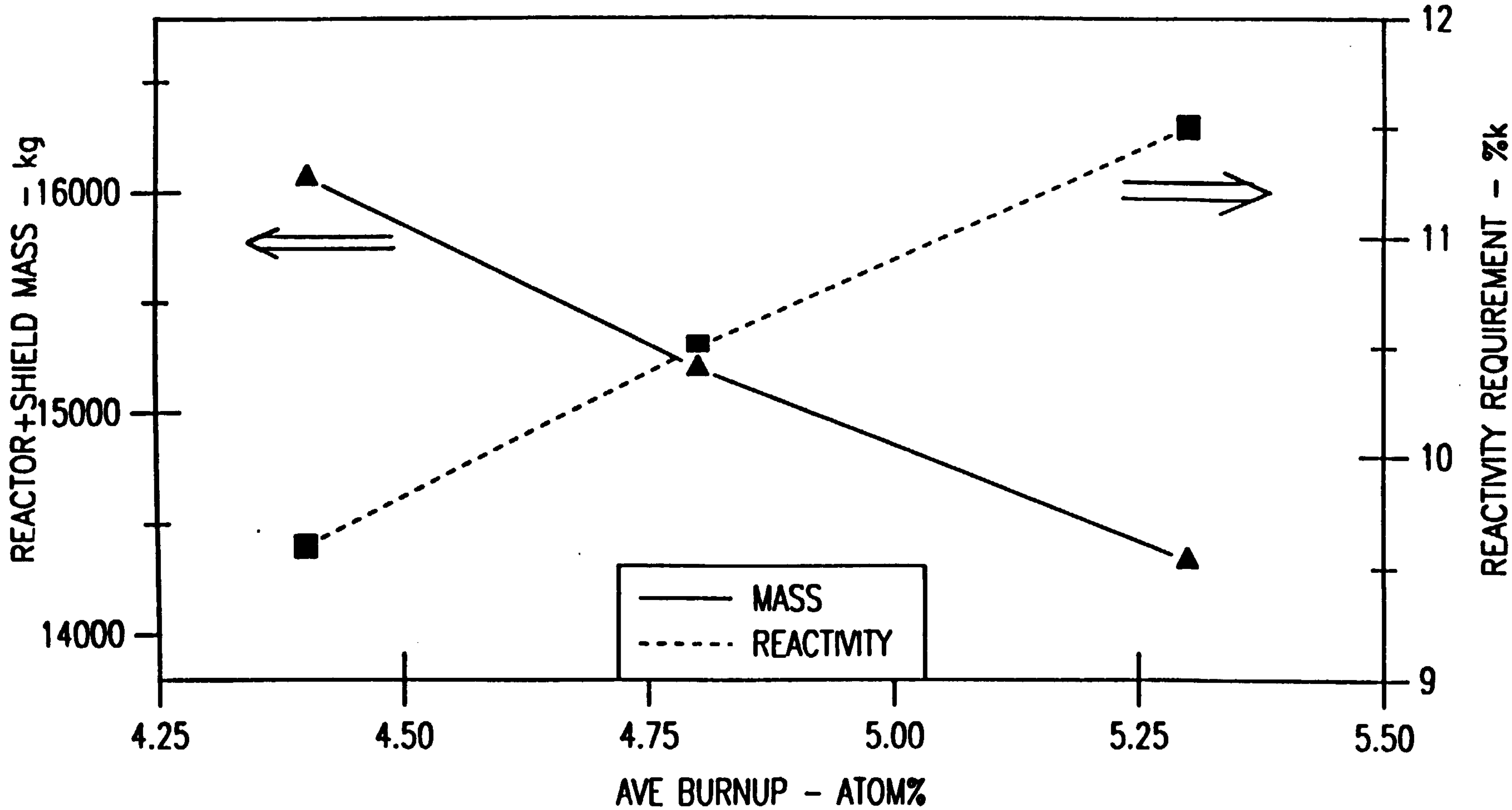


REFLECTOR WORTH



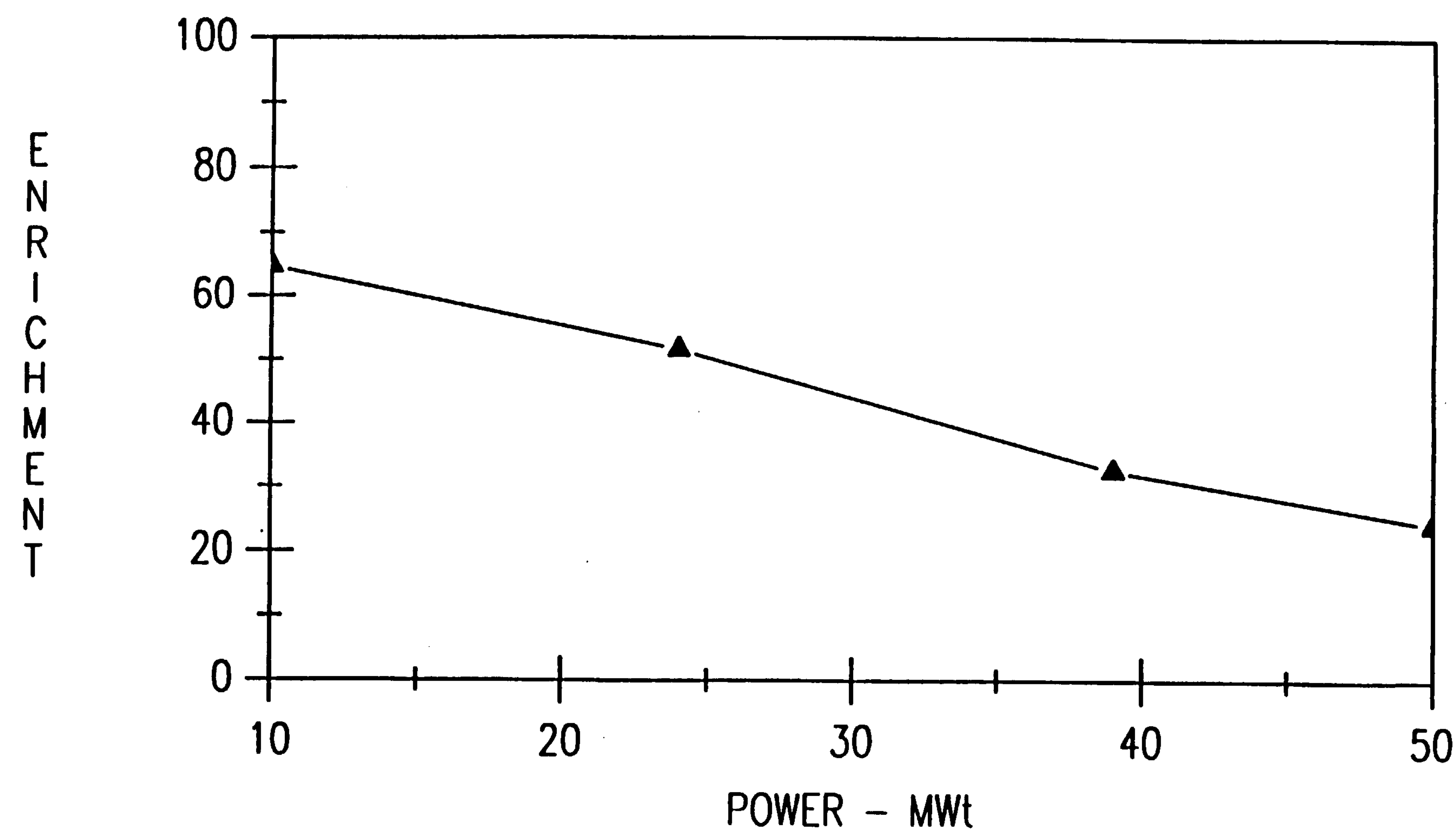
SP-100 REACTOR SCALABILITY STUDY

50 MW_T MASS AND REACTIVITY VS. BURNUP

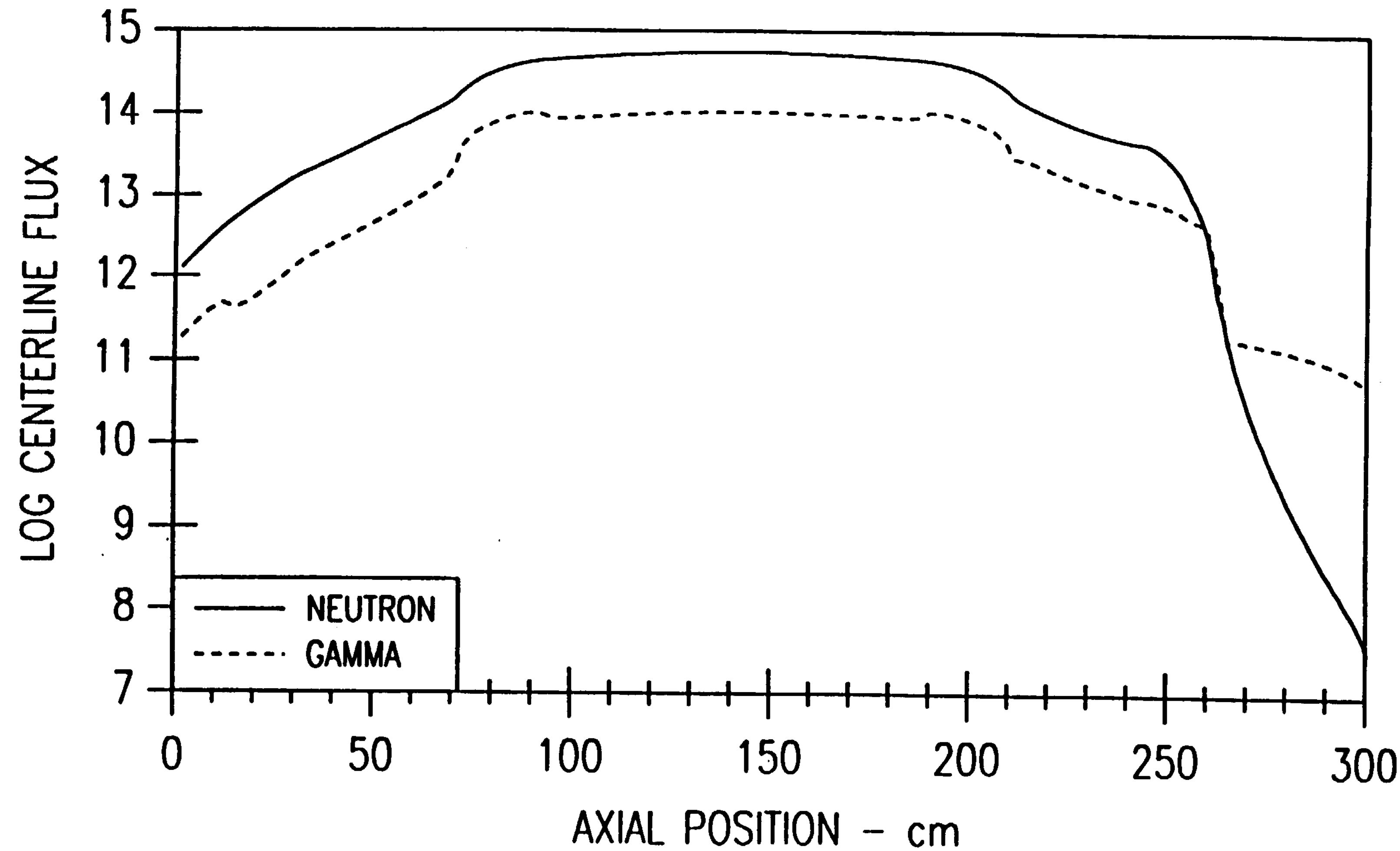


SP-100 REACTOR SCALABILITY STUDY

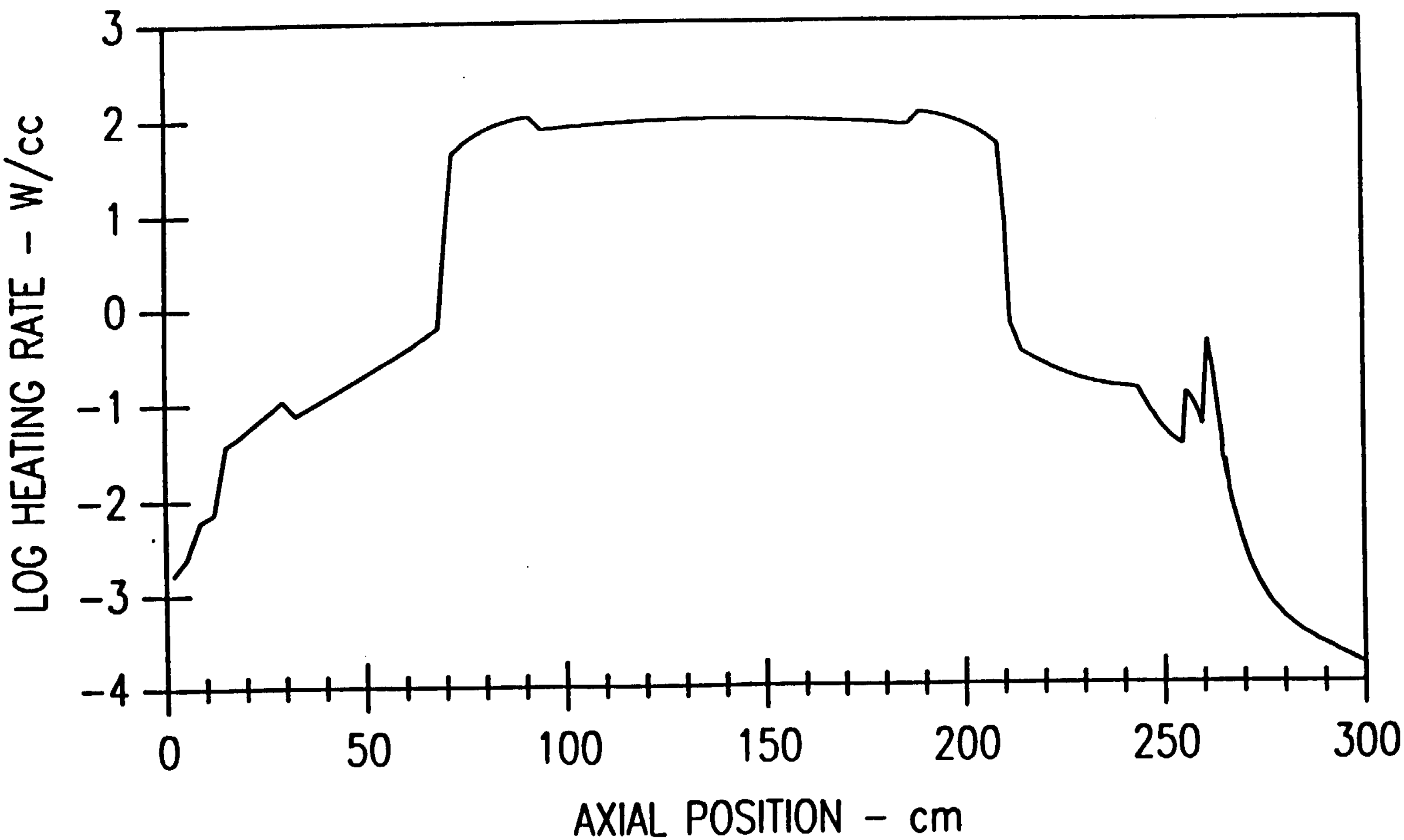
AVERAGE ENRICHMENT



50 MW_T CENTERLINE FLUX DISTRIBUTION



50 MWT CENTERLINE HEATING DISTRIBUTION



SP-100 REACTOR SCALABILITY STUDY

PERFORMANCE SUMMARY

REACTOR PERFORMANCE

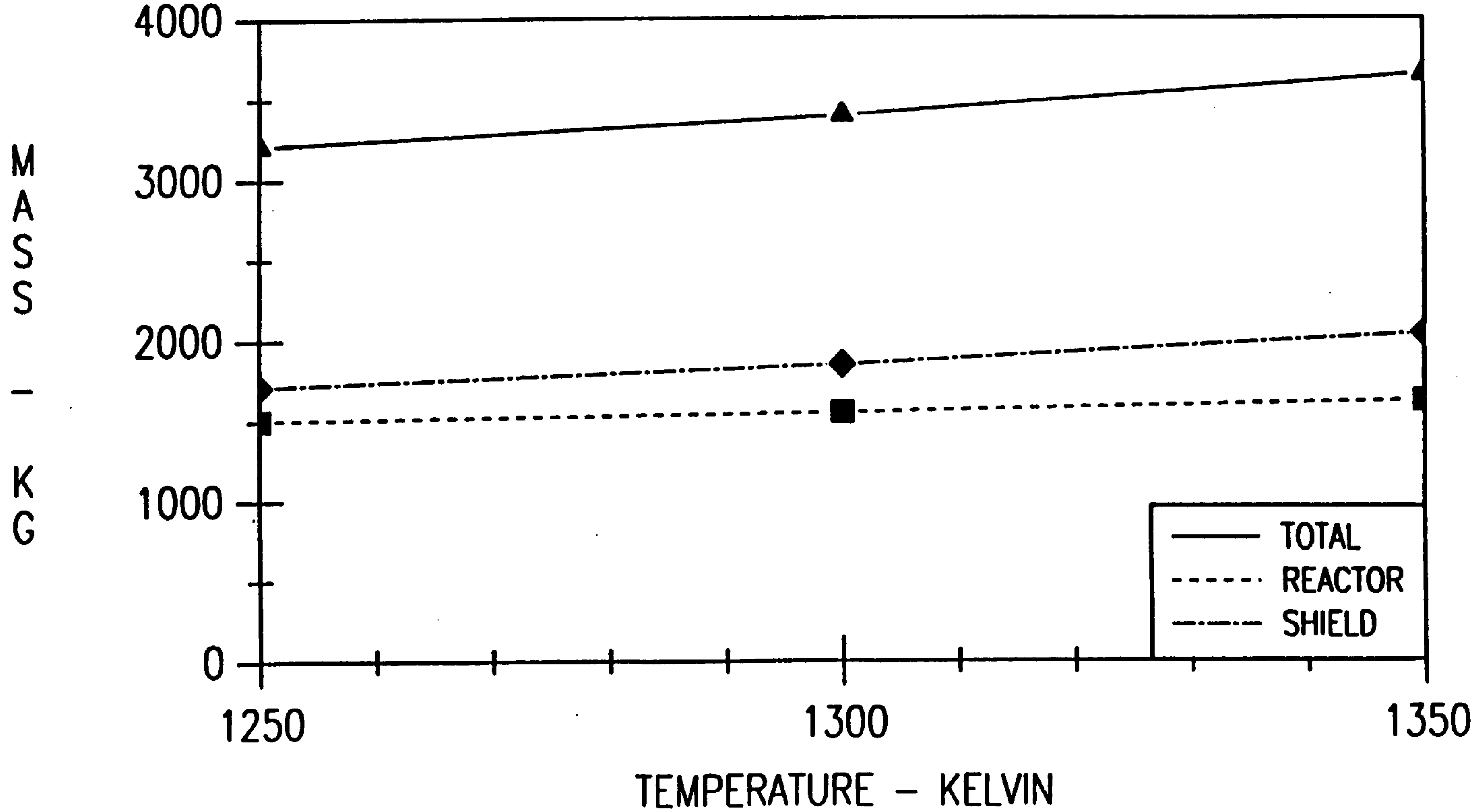
- SLIDING REFLECTORS (LIKE SNAP 10A) ARE NECESSARY TO MEET 10 YEAR MISSION CONTROL REQUIREMENTS WITH EX-CORE CONTROL
- THIN REFLECTORS (7.6 CM) ARE ADEQUATE UP TO ~25 MW_T
- THICKER REFLECTORS (UP TO 15.2 CM) ARE NEEDED UP TO ~40 MW_T
- COMBINATION OF THICKER REFLECTORS (UP TO 15.2 CM) AND LOWER FUEL BURNUP NEEDED TO MEET CONTROL REQUIREMENTS AT 50 MW_T

SHIELD PERFORMANCE

- NEUTRON AND GAMMA FLUENCES MET
- USE OF THICK (~12 CM) BE CONDUCTOR PLATE AND DEPLETED Li^7H REDUCES PEAK LiH HEATING TO ~60% OF SP-100 VALUE
- PASSIVE COOLING @ 50 MW_T IS POSSIBLE

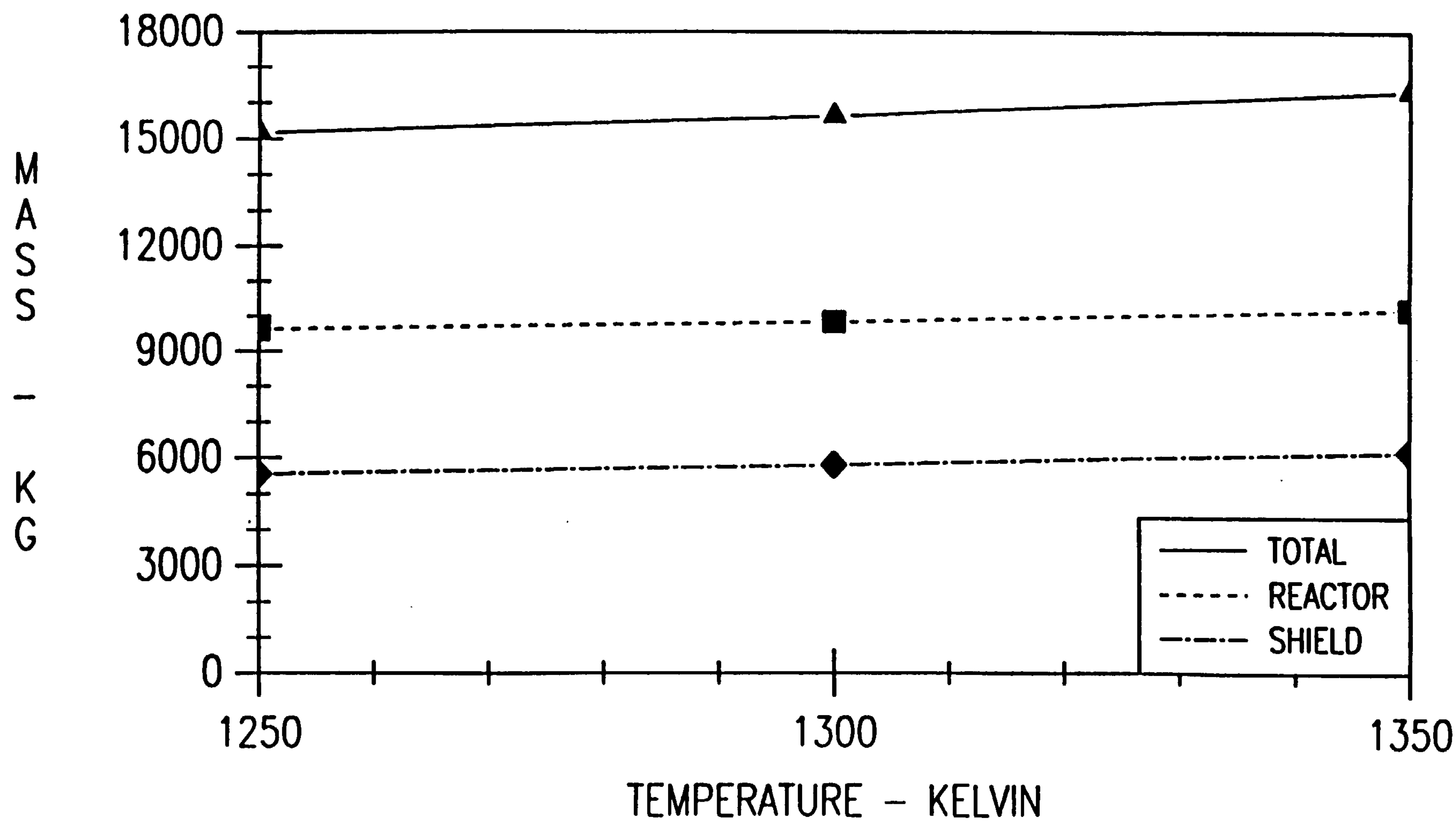
SP-100 REACTOR SCALABILITY STUDY

10 MW_T OUTLET TEMPERATURE SENSITIVITY



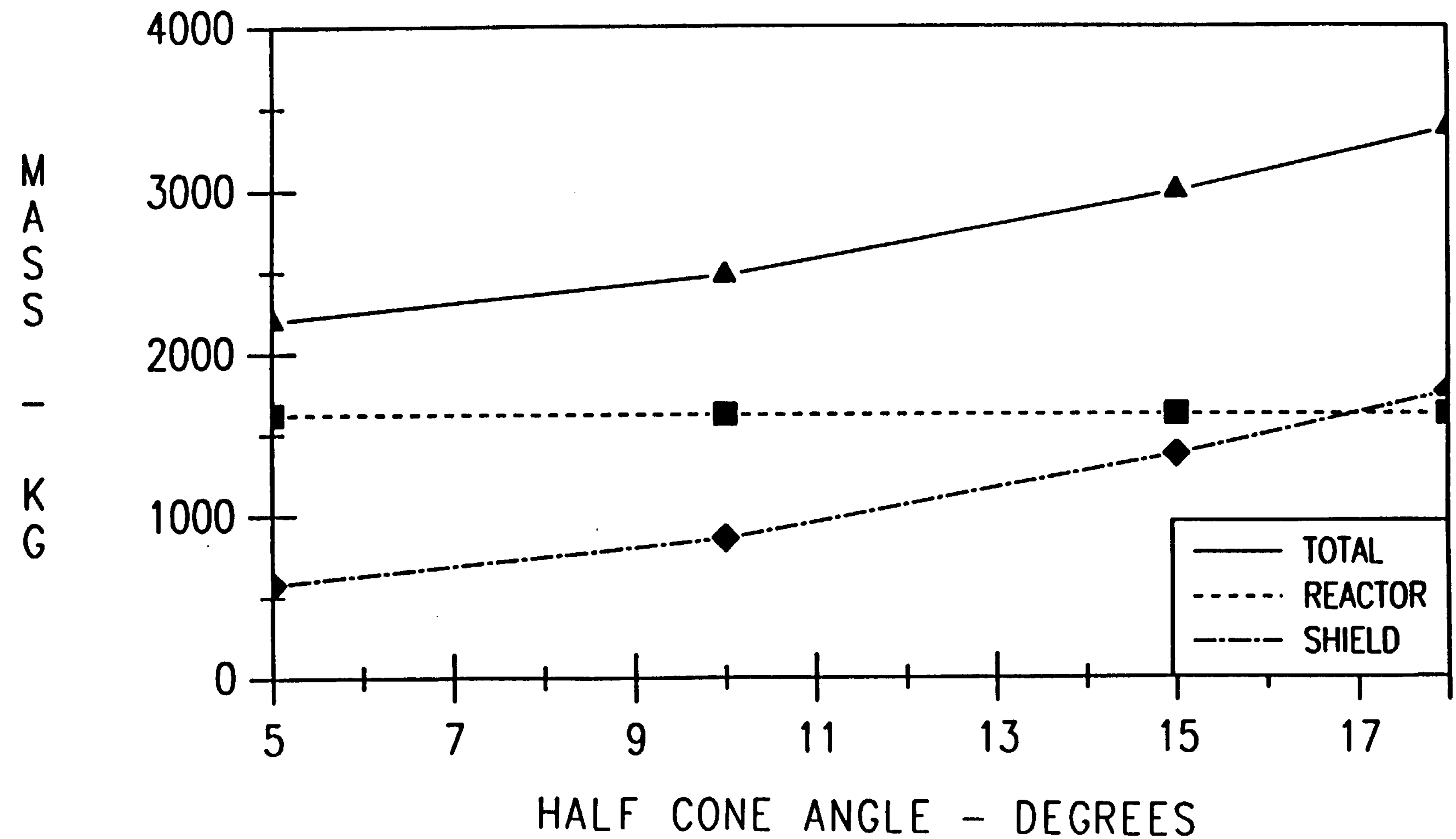
SP-100 REACTOR SCALABILITY STUDY

50 MWT OUTLET TEMPERATURE SENSITIVITY



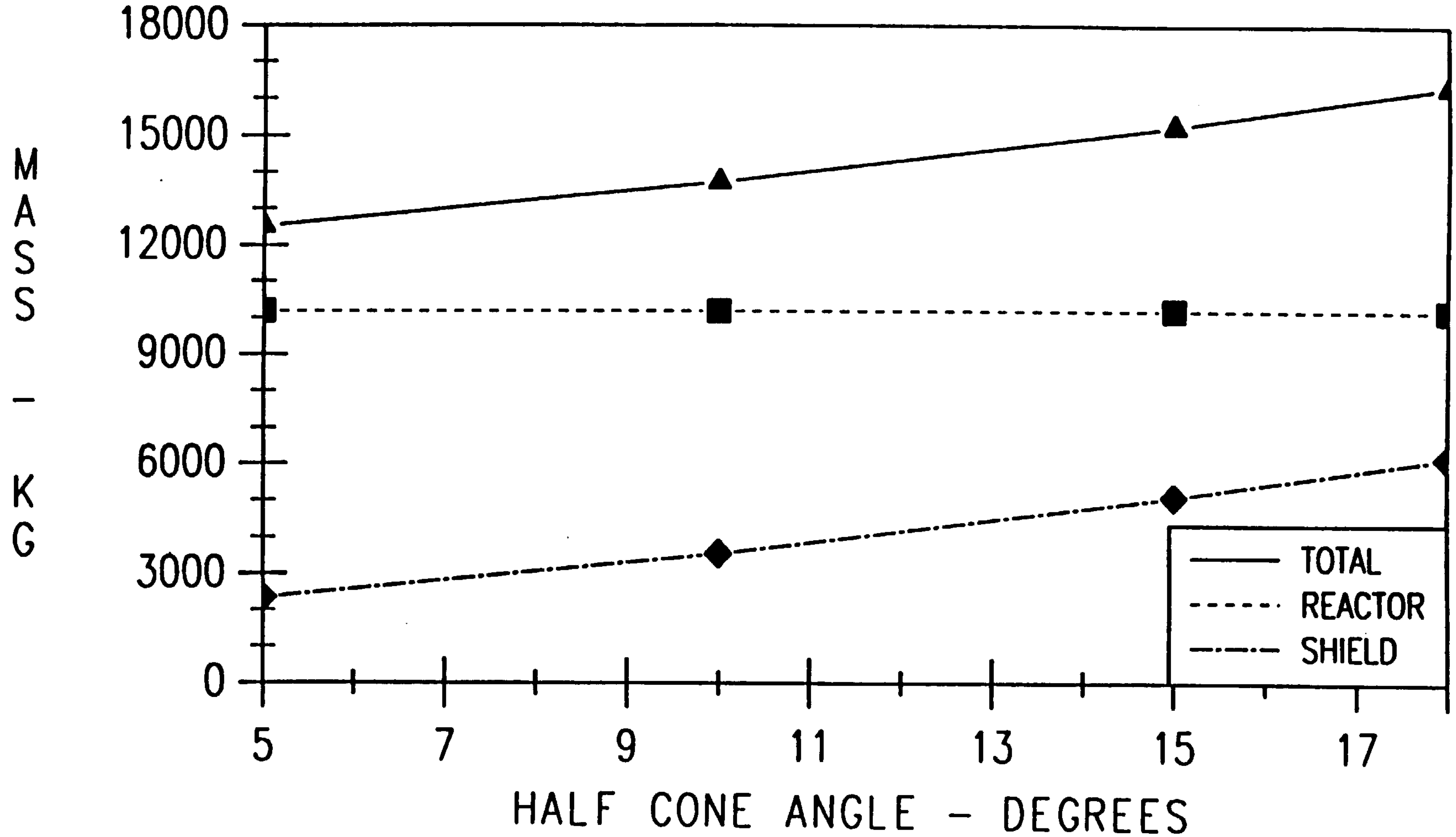
SP-100 REACTOR SCALABILITY STUDY

10 MWT CONE ANGLE SENSITIVITY



SP-100 REACTOR SCALABILITY STUDY

50 MW_T CONE ANGLE SENSITIVITY



C-6

SP-100 REACTOR SCALABILITY STUDY

MISSION LENGTH SENSITIVITY

	50 MW_T (3 YEAR)	50 MW_T (10 YEAR)
REACTOR	3,730 KG	10,170 KG
SHIELD	3,995 KG	5,980 KG
TOTAL	7,725 KG	16,150 KG

- MUCH SMALLER MASS IMPACT EXPECTED AT 10 MW_T POWER LEVEL

MASS IMPACT FOR SAFE LAUNCH OF ASSEMBLED REACTOR SYSTEM

50 MW_T DESIGN

	7 SAFETY RODS, MASS, KG	13 SAFETY RODS, MASS, KG
REACTOR	1,160	2,390
SHIELD	550	1,270
RE-ENTRY CONE	340	340
SAFETY ROD DRIVES	150	280
TOTAL	2,650	4,280
*BOL BURIED, K_{EFF}	0.98	0.77

*COROPT ESTIMATE

DEVELOPMENT NEEDS

- o NO TECHNOLOGY DEVELOPMENT ITEMS BEYOND GES
- o KEY FEATURE TESTS NEEDED
  - CRITICAL EXPERIMENT
  - SLIDING REFLECTOR
  - SCALED-UP CONTROL DRIVE
  - SCALED-UP PUMP
  - FLOW TEST
  - Li^7H NUCLEAR HEATING TEST
  - EXTEND FUEL AND MATERIALS IRRADIATION DATA BASE

SP-100 REACTOR SCALABILITY STUDY

CONCLUSIONS

- SP-100 REACTOR TECHNOLOGY IS SCALABLE TO 50 MW_T
  - EX-CORE CONTROL (SLIDING REFLECTORS) IS FEASIBLE
  - USE OF IN-CORE CONTROL WOULD REDUCE MASS ABOVE ~25 MW_T
- SP-100 SHIELD TECHNOLOGY IS SCALABLE TO 50 MW_T
  - PASSIVE COOLING IS FEASIBLE
- AN ATTRACTIVE, LOW MASS DESIGN CONCEPT HAS BEEN DEVELOPED TO MEET STUDY REQUIREMENTS (TEN YEAR MISSION):
  - 3,650 KG AT 10 MW_T
  - 16,150 KG AT 50 MW_T
- SIGNIFICANTLY LOWER MASSES ARE ACHIEVABLE FOR SHORTER MISSIONS
  - 7,700 KG AT 50 MW_T (THREE YEAR MISSION)

SP-100 REACTOR TECHNOLOGY CAN BE SCALED TO THERMAL POWER LEVELS REQUIRED FOR NUCLEAR ELECTRIC PROPULSION

POTENTIAL FOLLOW-ON WORK

- **INTEGRATE REACTOR AND SHIELD WITH OVERALL SYSTEM**
  - REACTOR/POWER CONVERSION SYSTEMS OPTIMIZATION/INTEGRATION
  - SHIELD/SPACECRAFT/PAYLOAD INTEGRATION
  - SENSITIVITY TO MISSION REQUIREMENTS
  - COMPLETE DEFINITION OF PRIMARY HEAT TRANSPORT AND REACTOR I&C SYSTEMS

- **OPTIMIZE REACTOR**
  - REACTOR ARRANGEMENT
  - ADVANCED TECHNOLOGY (E.G., VENTED FUEL AND/OR REACTOR, IN-CORE CONTROL, ALTERNATE MATERIALS)

- **OPTIMIZE SHIELD**
  - SHIELD ARRANGEMENT
  - ALTERNATE MATERIALS

- **EVALUATE IMPLICATIONS OF "ASSEMBLY IN SPACE"**

2.1.7

SOLAR ELECTIC PROPULSION

POWER SYSTEM OPTIONS ASSESSMENT

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Solar Electric Propulsion (SEP) Power System Options Assessment

Introduction - This is the combined report of three interrelated studies: The Mars SEP Radiation Damage Tolerance Study, The Mars SEP Power System Options Study, and The Mars SEP Conceptual Design Study.

Objectives - The objectives of these studies were the following: to evaluate the survivability against radiation damage of solar photovoltaic cell technologies and power systems for a Mars SEP cargo vehicle, to compare applicable solar power options, to calculate the performance, size, and layout of a SEP power system, and to integrate the various SEP power system components into a system concept.

Rationale - SEP vehicle designs have advantages over both nuclear and chemical systems which could make SEP vehicles viable alternatives to those other systems. A SEP power system neither requires radioactive fuel nor produces radioactive waste, and thus could never impose a radiation hazard to cargo, rendezvous sites, or human personnel. The SEP vehicle has reduced propellant requirements when compared to chemical propulsion systems. This results in a higher payload fraction: 50-60% for a SEP vehicle compared to 10-25% for chemical vehicles.

There are other reasons as well that would lead to consideration of SEP vehicles over other systems. Photovoltaic power systems are flight-proven systems. The Solar Electric Propulsion Stage (SEPS) concept was taken through the design stage and an engineering model was built (OAST Technology Readiness Level 6). The Solar Power Satellite (SPS) blanket design had a specific mass competitive with chemical and nuclear systems. Also, Space Station Freedom will use large photovoltaic arrays generating a large data base and practical experience. All the technologies studied could be ready as early as 2000-2005.

It is possible the power system could be further used once the vehicle has reached its Mars destination. The distance of the PV power system from the sun notwithstanding, the PV arrays will still be generating several megawatts. Beamed power to the Martian surface would be possible. The power from the arrays could also be transported to Phobos via a conducting tether. The possibility also exists that the blankets could be repackaged and landed on the Martian surface for use there.

Power System Groundrules - Photovoltaic (PV) and solar dynamic (SD) power systems were initially considered. SD systems were determined to be too massive and were subsequently dropped from the study. To provide some parity with a similar NEP Conceptual Design Study, the power system was baselined to generate 5 MWe at earth (1 A.U.). Because the vehicle would be constructed and launched from low earth orbit (LEO), the power system would have to be oversized such that after passing through the Van Allen Belts the power output would degrade down to the rated 5 MWe

level. The vehicle would spiral out from 408 km around earth to escape altitude, then rendezvous and come to a circular orbit 17033 km above Mars (areosynchronous circular orbit). The vehicle would be able to deliver 400 metric tonnes (MT) of payload. Because of the severe degradation of power output due to radiation damage, only one trip through the Van Allen belts is reasonable for multiple trips to Mars. Vehicles returning to earth should stay in high earth orbit, beyond the radiation belts.

It was determined that no storage would be included in the power system. Storage would be necessary if constant thrust through Earth's umbra is required. However, storage is extremely massive and unnecessary for most of the trip to Mars. By not having storage, the vehicle would lose altitude as it passes through the shadow region. A trade study is necessary to optimize the mass penalty of energy storage versus extra trip time in low earth orbit.

Propulsion System Groundrules - Self-radiating argon ion thrusters were chosen for the reference electric propulsion (EP) configuration. Existing experimental thrusters are 10-50 cm in diameter and use 1-30 kWe of power. Rather than use hundreds of such existing low power devices, thereby introducing complexities of propellant and power distribution, development of advanced, megawatt-level thrusters is preferred. These advanced thrusters would require technology advances in grid manufacture and materials to support the large grid area required at high power levels. Thruster efficiency at these power levels and with large grids is assumed to be comparable to results observed in low power devices. This behavior is expected based on first order physical principles governing ion thruster performance; empirical data in higher power devices is required to verify this assumption. Thruster dissipated power is rejected by the thruster grids themselves, which provide sufficient surface area to radiate the waste heat.

For a SEP vehicle, ion engine throttling with respect to power is required due to the variation in power with distance from the sun. The preferred throttling method uses a third decelerator grid in addition to the screen and accelerator grids. This third grid allows the thrust to be throttled down while the specific impulse (Isp) remains constant without requiring changes in conditions within the discharge chamber. By avoiding the limitations inherent in the two-grid system, the three-grid system would be capable of a broader operating range of Isp.

Of the thruster configurations considered, it was decided that the SEP cargo vehicle would operate 8 1-by-5 meter thrusters. The lifetime of these thrusters is approximately 10,000 hours, therefore 2 sets (16 thrusters total) would be required for a one-way trip to Mars. The power into each thruster is 0.625 MWe at 2000 volts-dc. The beam current leaving the thruster is 679 amperes with an ion production cost of 150 W/A. This thrust yields a specific impulse of 5000 seconds at an efficiency of

0.67. The specific mass per thruster is 2.3 kg/kWe. Other thruster options produced higher Isp or greater beam current but required only 4 thrusters. With only 4 thrusters, a loss of a single thruster due to structural failure or meteoroid impact could be catastrophic or at the very least crippling to the mission. It was therefore decided that the 8 thruster option would be selected as keeping the power and propellant management minimal while providing greater assurance against mission failure due to single thruster loss.

Radiation Damage - Sources of radiation damage to the solar arrays are solar flares and the Van Allen belts consisting of both electrons and protons, each with a spectrum of energy levels. Of the two sources of radiation, the Van Allen belts are the predominant source of array damage.

Since there are wide spectrums of energies for both electrons and protons in the space environment, the space PV community has been using the concept of 1 MeV electron equivalence for all radiation degradation projections. For example, the actual damage to silicon cells by electrons of any energy is related to the damage produced by 1 MeV electrons. In a similar manner, all proton damage is related to the damage caused by 10 MeV protons. A factor of 3000 relates the 10 MeV proton damage to 1 MeV electron damage (i.e., one 10-MeV proton causes the same damage as 3000 1-MeV electrons). This allows degradation to be calculated using the readily available 1 MeV experimental data on solar cell damage.

The JPL Radiation Handbook has several tables of 1 MeV annual equivalent fluences. Each table is for a particular orbit inclination and incident particle (electrons or protons). In each table, the annual equivalent fluence is given at 34 different altitudes (LEO to GEO) and eight different quartz coverglass thicknesses (0 to 60 mils). These tables are the result of extensive integration of electron and proton energies at different orbits along with available radiation damage experimental data. This concept has been of great value to array designers using silicon cells. There have been a large number of actual flights where preditions have been very close to the actual degradation. For gallium arsenide (GaAs) cells, one major change is the ratio between 1 MeV electrons and 10 MeV protons. A value of only 1000 is considered a much better correlation. For the calculations in this study, a ratio of 1000 was used for GaAs type cells (i.e., all III-V cells).

Figure 1 shows the annual 1 MeV electron equivalent fluence as a function of altitude for three different coverglass thicknesses. This particular set of curves is the equivalent fluence for proton caused damage. There is significant reduction in fluence due to the coverglass. The reduction is different at various altitudes due to the changing proton energy spectrum as a function of altitude.

Figure 2 shows the SEP vehicle altitude schedule and the

radiation flux as a function of days in orbit. The graph shows that the SEP vehicle power system will sustain most of the radiation damage from approximately day 90 to day 170, with peak fluence from day 127 to day 151. Figure 3 plots the time in orbit and the radiation flux as functions of orbital altitude. This graph indicates that the radiation damage will occur in the region from 6500 km to 20400 km altitude, with the maximum damage occurring from 11100 km to 16100 km.

Due to the amount of time spent in the proton-dominated radiation belts, the largest component (greater than 96%) of the equivalent fluence is due to the protons. For the portion of the mission beyond GEO, solar flares are the only significant contribution. Hence, to calculate the total equivalent fluence for the LEO to GEO portion of the SEP vehicle orbit, the fluence levels were summed for each day at a particular altitude over the range of altitudes. This was done for several coverglass thicknesses. Contributions from both front and back irradiation were included since the SEP array will not be a body mounted array. It was assumed that the orbit was at a 30 degree inclination.

The results of the above calculations give equivalent 1-MeV electron fluences for the LEO to GEO portion of the SEP vehicle orbit for any combination of front and back shielding thicknesses. With the data for 1-MeV solar cell degradation, one can plot the loss in maximum power to GEO as a function of total thickness for GaAs, InP, and silicon solar cells (figure 4). This data can then be used to determine array sizes using various cell types and shielding thicknesses.

Array Technologies - Two basic array configurations were included in this analysis: planar and concentrator. Planar silicon arrays have been used for over 30 years. The current trend in planar arrays is toward thinner cells on lightweight substrates such as Kapton (to be used on Space Station Freedom). NASA is funding an effort to optimize lightweight arrays through the Advanced Photovoltaic Solar Array (APSA) program. The work is being done by TRW through JPL. The goals of the program include achieving an array specific power of greater than 130 W/kg (BOL) for array sizes in the 10 to 25 kW range. Since this design is the most advanced of any planar array design, it was decided to use the APSA design as the baseline for the SEP array with the following assumptions: 1) the specific power of the APSA array is constant to multi-megawatt levels, 2) different cells can replace the current baseline 2.5 mil silicon cells with the proper changes in cell efficiency and weight, and 3) extra shielding beyond the 2 mils provided by APSA on both front and back sides can be added with the proper weight additions. There is probably a limit to how much shielding can be added without changing the array structure. Using these assumptions, the APSA array can be configured with any cell type and with additional shielding for the trip through the radiation belts.

There is no space flight experience for concentrator arrays. Several arrays are being designed for military purposes such as the TRW cassegrainian array, but they would be too heavy due to survivability requirements. NASA Lewis has a SBIR contract with ENTECH to develop designs for lightweight, high efficiency concentrator systems. These systems use domed fresnel lenses with high optical throughput (greater than 90%) and lightweight structures. Currently, materials are being selected that will enhance the array's stability in a space environment.

Cell Technologies - There are several different cell types which could be used for the SEP array. Since the lead time for final selection is probably about 10 years, many research cells are available. Brief discussions of several cell types follow:

Single Crystal Silicon - These cells have been used almost exclusively in the United States space program for the last 30 years. Their size has grown to the 8x8 cm cell to be used on Space Station Freedom (SSF). Silicon cells have an efficiency up to 13.5% at the expected operating temperatures of the SEP array. Since other cell types have higher efficiencies and better radiation resistance, the silicon cells were not considered further.

Amorphous Silicon - These are thin film cells with low efficiencies (under 10%). They are lightweight, however, due to their thinness (less than a mil). Amorphous silicon cells have been used in terrestrial markets for several years. Some preliminary data indicates that amorphous silicon cells may be radiation resistant. This, coupled with low mass, makes them good candidates for missions where area is not a factor. For the SEP array assembled in LEO, they cannot be considered due to their low efficiency resulting in atmospheric drag.

Gallium Arsenide - Gallium Arsenide (GaAs) is a III-V cell with a bandgap near the optimum for sunlight conversion. Hence, its one-sun efficiency can be near 20%. For the SEP array, a conservative assumption was made of 18% efficiency at temperature for production line cells. This is about the highest efficiency for single-junction cells. GaAs is also slightly more radiation resistant than silicon (see figure 4). A major disadvantage to a GaAs SEP array is its mass. GaAs weighs twice as much as silicon, and GaAs cannot be thinned to 2.5 mils like silicon because it is too brittle. Current GaAs cells can be no thinner than 8 mils. A very promising solution to the mass problem is to grow the active GaAs cell layers (which are less than 1 mil) on 3 mil germanium substrates, cutting the cell mass by half. This technology is in the development stage under an Air Force manufacturing technology program. Currently, GaAs cells are the power source for a few military missions about to be flown.

For the SEP mission, GaAs on germanium (GaAs/Ge) is a viable candidate for both planar and concentrator options. Late in the analysis it was shown that GaAs/Ge, albeit a viable choice, was

not competitive with other cell technologies. Therefore, a GaAs/Ge SEP array is not among the final selected cell technologies. The same performance parameters were assumed for GaAs/Ge as for 8 mil GaAs.

Indium Phosphide - Indium Phosphide (InP) is also a III-V cell with high efficiency (17% for production line cells). The main advantage of InP is its high radiation resistance. NASA is researching the InP radiation resistance. This work includes both electron and proton damage at a variety of energy levels, the potential for illuminated annealing, and the applicability of InP cells to concentrator arrays. Currently, InP substrates are expensive, and for this study, InP cells are assumed to be grown on other substrates such as silicon, currently the subject of a NASA contract. Due to its radiation resistance and fairly high efficiency, InP was selected for study as a prime candidate for the SEP mission.

III-V Multijunction Cells - Multijunction cells use the solar spectrum more effectively by having different spectral bands absorbed by different bandgap cells. The higher bandgap cell is placed above the lower bandgap cell. The short wavelength light is therefore absorbed in the upper cell with the longer wavelength light passing through to the lower cell. The major advantage is much higher efficiency, with performance levels of between 25 and 30 percent possible. The two junctions may be series connected with required matching currents or mechanically stacked with separate wiring harnesses. In either case, there are potential problem areas which are the subject of current work.

Power Generation Systems - The two array and two cell technologies selected were combined into four power generation systems. Radiation damage factors were calculated for each of these systems. This factor, along with a power processing efficiency, was then used to scale the power systems to the mass and area required to achieve 5 MWe after radiation degradation (figure 5). The level to which each system is oversized can be seen in figure 6. The InP concentrator array power system is the least oversized, but with a mass of over 100 MT, its power system specific mass was the greatest at just over 20 kg/kWe (evaluated at the assumed 5 MWe at 1 A. U. level). The power system that exhibited the least power system mass, area, and specific mass was the multijunction concentrator array system.

Power Processing - To meet the 2000 Vdc input required by the thrusters, the array blanket circuits could be connected to provide that voltage directly, thus eliminating the need for power processing. There are two problems with this technique, however. The first problem is that engine throttling would not be possible. To maintain constant Isp, the input voltage must vary. This cannot be done without a power processing unit. The second problem is that solar arrays generating high voltage current may very well experience arcing in the charged plasma of

the Van Allen belts. Arcing could damage the cells and circuitry of the array blankets, possibly causing total failure of the array wing.

To eliminate these problems, a power processing unit (PPU) is necessary. One possible PPU is represented in figure 7. This PPU first converts low voltage dc current to low voltage ac current. This ac current is then transformed to high voltage, high frequency ac current which is then converted back to dc current, now at 2000 Vdc, available for use by the thrusters. An overall PPU efficiency of 90% and a specific mass of 3.3 kg/kWe (including the PPU heat rejection system) was assumed.

The PPU mass is a significant fraction of the entire power system mass (figure 8). For the four power systems considered for the SEP vehicle the PPU masses range from just under 20 MT for the InP concentrator array to just over 30 MT for the multijuction planar array. These masses are 20 to 35 percent of the total power system mass, respectively. For the light weight multijuction concentrator array, the PPU contributes over 41 percent of the 61 MT power system mass, while the array contributes 45.5 percent, the remainder consisting of the thrusters and masts.

<u>Vehicle Design</u> - Figure 9 depicts a possible configuration for a SEP vehicle. The octagonal array area is composed of eight keystone-shaped split blanket arrays on eight radial masts. Two thrusters are on the extended ends of each of the masts for a total of 16 thrusters (the number required for the one-way trip to Mars). The argon propellant tanks and 400 MT payload would be positioned in the center.

The octagonal shape provides redundancy and a relatively short distance for power and distribution (PMAD) and for the propellant feed lines. By positioning the thrusters at the end of mast extensions, they are spaced apart, increasing reliability. A severed propellant line or mast from a meteoroid impact could take at most two thrusters out of service in this configuration. Clustering thrusters, on the other hand, leads to the possibility of the entire cluster being disabled. The octagonal shape also provides symmetry and a logical structural framework for the split blanket arrays.

The SEP vehicle mast design is the two-wing center mast design used for both the Space Station Freedom and APSA designs. The main differences between their designs and that of the SEP vehicle is that the SEP vehicle design is much larger, and the top cross member is longer than the bottom cross member (non-rectangular blankets). It does not appear that either of these differences will present complications, although the design of the cross members and boom probably will differ (e.g., continuous beams for the APSA versus post tensioned truss structures for the SEP vehicle) from the previous designs because of the large dimensional dissimilarities. It presently is not clear exactly how the structural components will look but

it is expected that the mast and boom cross sectional areas will increase and the other components may remain relatively similar in comparison to the smaller Space Station design. The overall design will probably be constrained by a lower stiffness limit (e.g., 0.01 Hz) and a demand to withstand a maximum acceleration level (e.g., 0.1 g). For a structure of this large size, post tensioned guide wires possibly could be used for providing stiffness and a means for vibration suppression. Cables also may be used to minimize the blanket substrate thickness (and mass) by relieving the substrate from having to carry all of the blanket tension loading. In the future, a study should be made to investigate the trade-off between mast length, weight, and thruster requirements.

Mission Performance Analysis - The mission was analyzed using CHEBYOPT, a trajectory program which optimizes thruster steering, switching points, and launch/arrival dates to minimize the propellant required to accomplish the mission. The specific mass of the power system was varied parametrically to obtain figure 10. This graph shows the effect specific mass has on initial mass in LEO for various trip times. Analytic spirals were used for the escape and capture portions of the trip. This method did not include shadowing effects, nor the effect of degradation due to radiation damage. The overall trip times were altered by introducing coasting periods of varying durations.

The effect of degradation due to the radiation damage was analyzed separately. Spiral escape trajectories were generated and used in conjunction with a detailed radiation model to ascertain the level of cell degradation. This information was used to oversize the arrays to allow for a final power level of 5 MWe. This power level was then used to develop the final results. A computer code developed at the Lewis Advanced Space Analysis Office was used to generate the escape trajectory and was verified using the more detailed geocentric trajectory program, SECKSPOT.

The vertical lines in figure 10 represent the four power systems examined. This figure indicates that a SEP vehicle using multijunction concentrator arrays could be designed to reach Mars in 1000 days with a total vehicle initial mass in low earth orbit (IMLEO) of 650 MT (60% payload fraction). By following an all propulsive trajectory, this same vehicle could reach Mars in 870 days with a mass penalty of 30 MT.

Observations and Implications - Atmospheric drag and Van Allen belt radiation present challenges to assembly and launch of SEP systems at LEO. Figure 11 shows preliminary estimates of the time it could take for a SEP vehicle to descend from 400 km to 100 km orbital altitude assuming no orbit raising thrust capability. Two decay times are shown for each viable power system identified above. The worst case decay time is calculated assuming that the plane of the vehicle arrays is perpendicular to the direction of motion through the

atmosphere. The best case decay time assumes minimum drag, that is, the plane of the arrays is parallel to the direction of motion. For the InP planar array system, the SEP vehicle could descend to a 100 km altitude in as few as 14 days, assuming the worst case condition. With feathering of the arrays, the vehicle could have hundreds of days without thrust before re-entry.

The other problem with launch from LEO is that the vehicle will have to go through the Van Allen belts. The degradation of the cells and therefore the power output due to radiation damage requires that the arrays be oversized to compensate for the power loss and extra shielding mass (i.e., thicker coverglasses) be added. For the four systems examined, the power systems were oversized 18 to 82 percent, increasing both the mass and array area.

If the SEP vehicle were assembled and launched from GEO, both of the difficulties above would be eliminated. There would be no atmospheric drag problem lengthening total trip time to Mars. Nor would there be the magnitude of radiation damage incident upon the arrays from GEO to Mars as there would be from LEO to GEO. This alone could reduce the power system mass and area from 6 to 36 percent compared to beginning at LEO. The difficulty is that transportation to GEO is necessary.

<u>Conclusion</u> - A preliminary characterization of an unmanned solar electric propulsion cargo vehicle was made. Solar photovoltaic cell and array technologies were evaluated on the basis of maximum radiation tolerance and minimum mass and area. The two cell technologies selected for further study were indium phosphide and III-V multijunction solar cells. The two array technologies selected for further study were the APSA planar array and the ENTECH concentrator array. Of the four power generation systems created by the combination of the above technologies, the multijunction concentrator system had the least mass, area, and power specific mass.

Mission analysis was performed calculating trip times to Mars. Trip times through the radiation belts were also calculated to determine the radiation damage used in the power systems analysis. From the mission analysis, total trip times to Mars should fall in the range of 870 to 1000 days with a corresponding initial mass in low earth orbit of 680 to 650 metric tonnes.

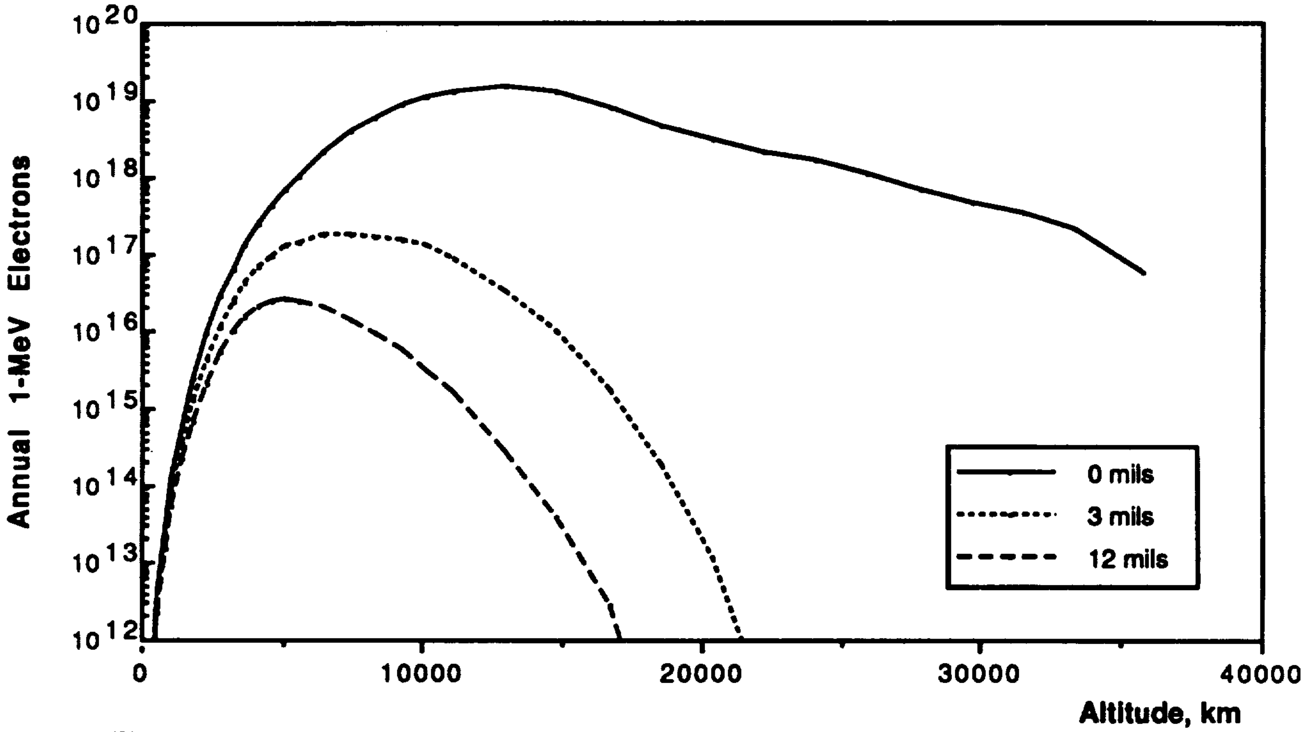


Figure 1 Proton fluence (1-MeV equivalent electron) at 30 degrees inclination

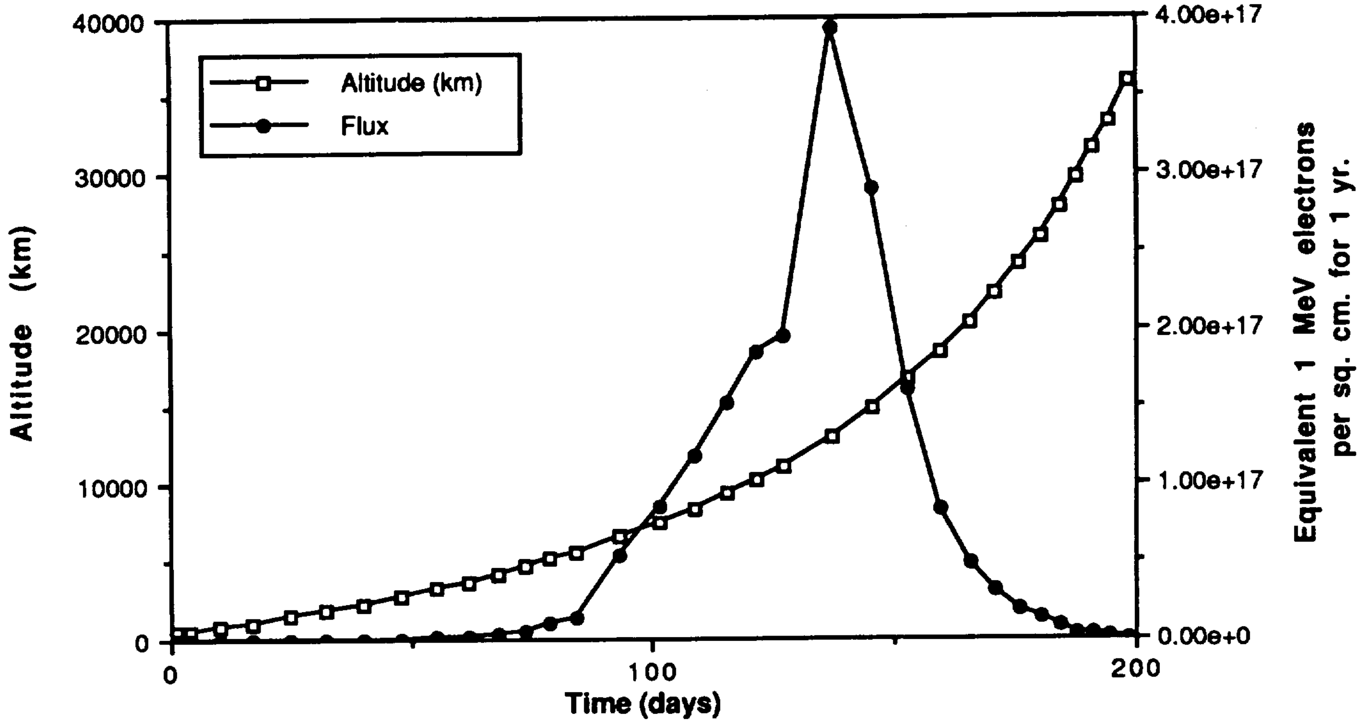


Figure 2 SEP Orbit Altitude and Flux Schedule, Zero Shielding, 30 Degree Inclination

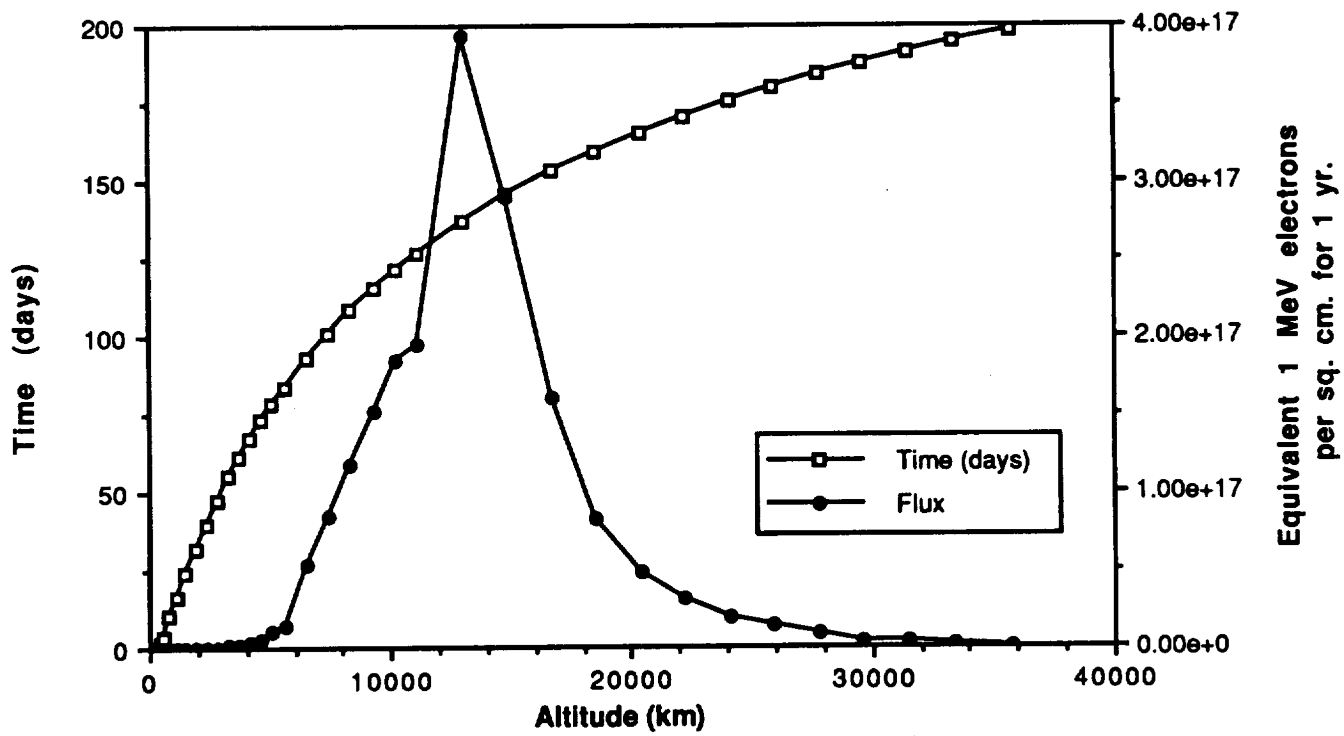


Figure 3 SEP days in orbit & flux vs. altitude, LEO to GEO, zero shielding, 30 degree inclination

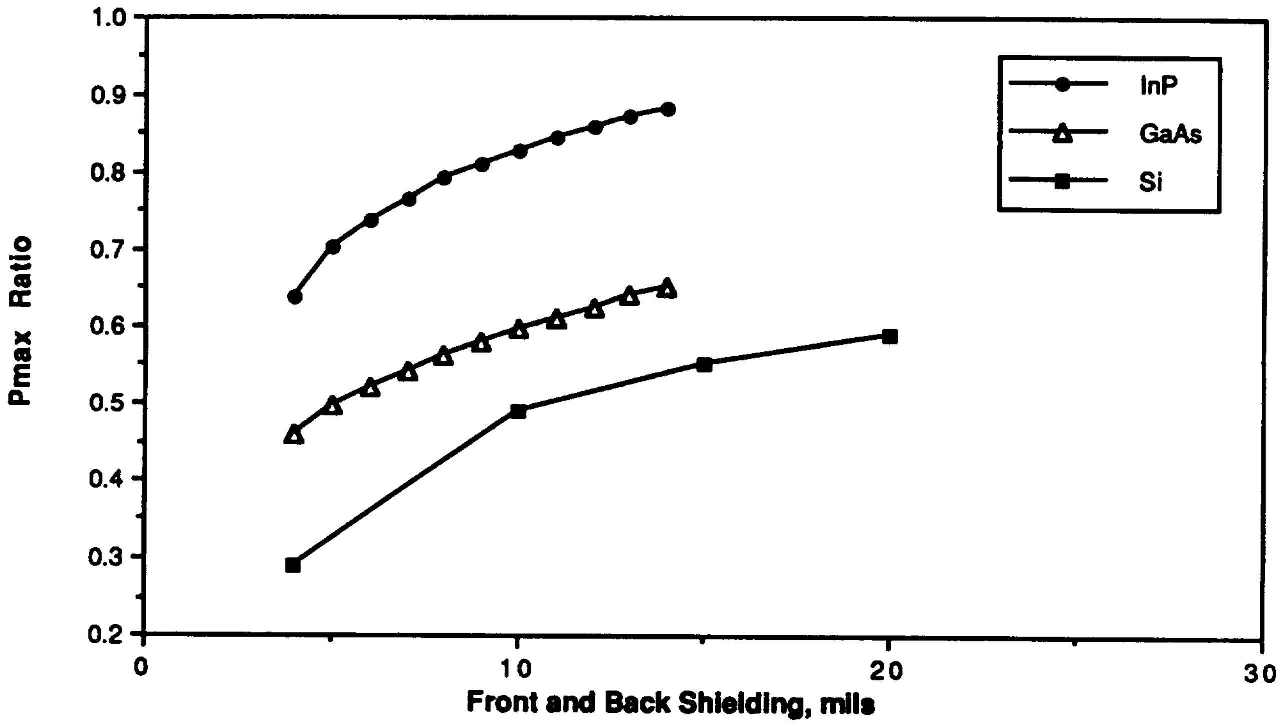


Figure 4 Pmax ratio vs. total shielding

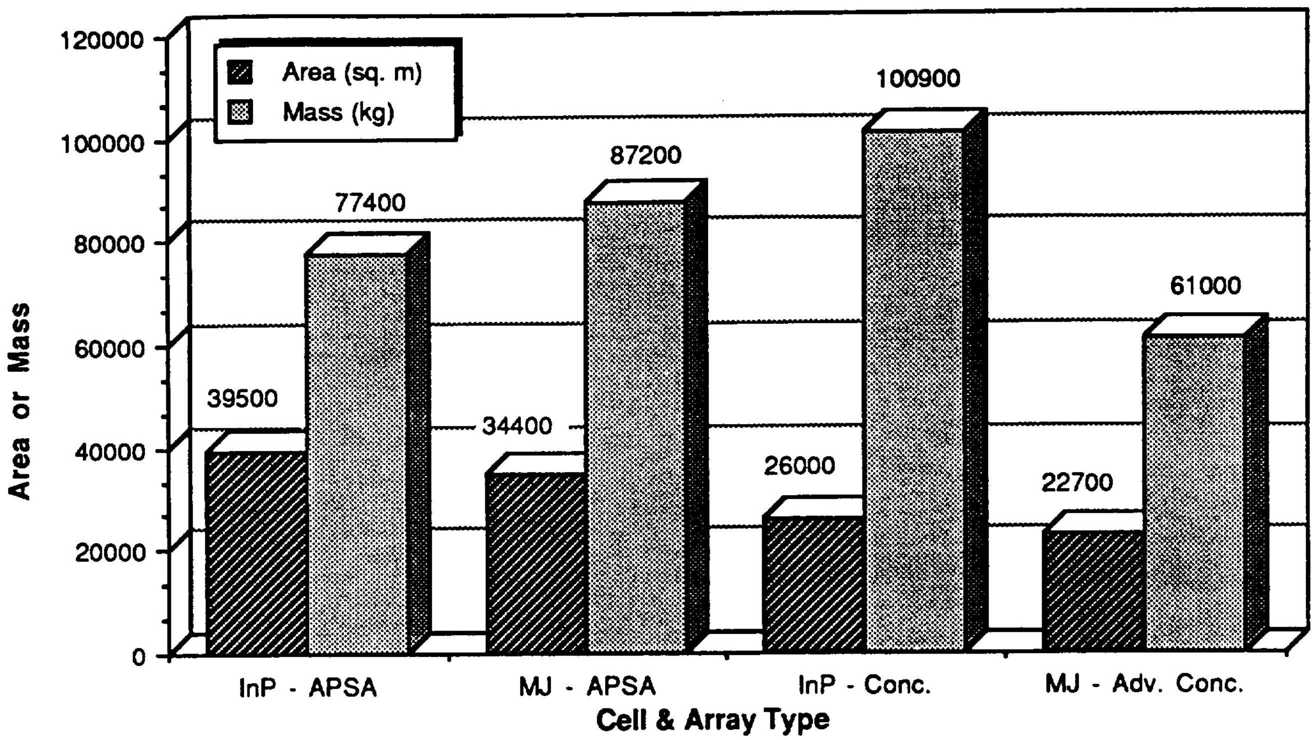


Figure 5 Comparison of mass and area for the selected power systems

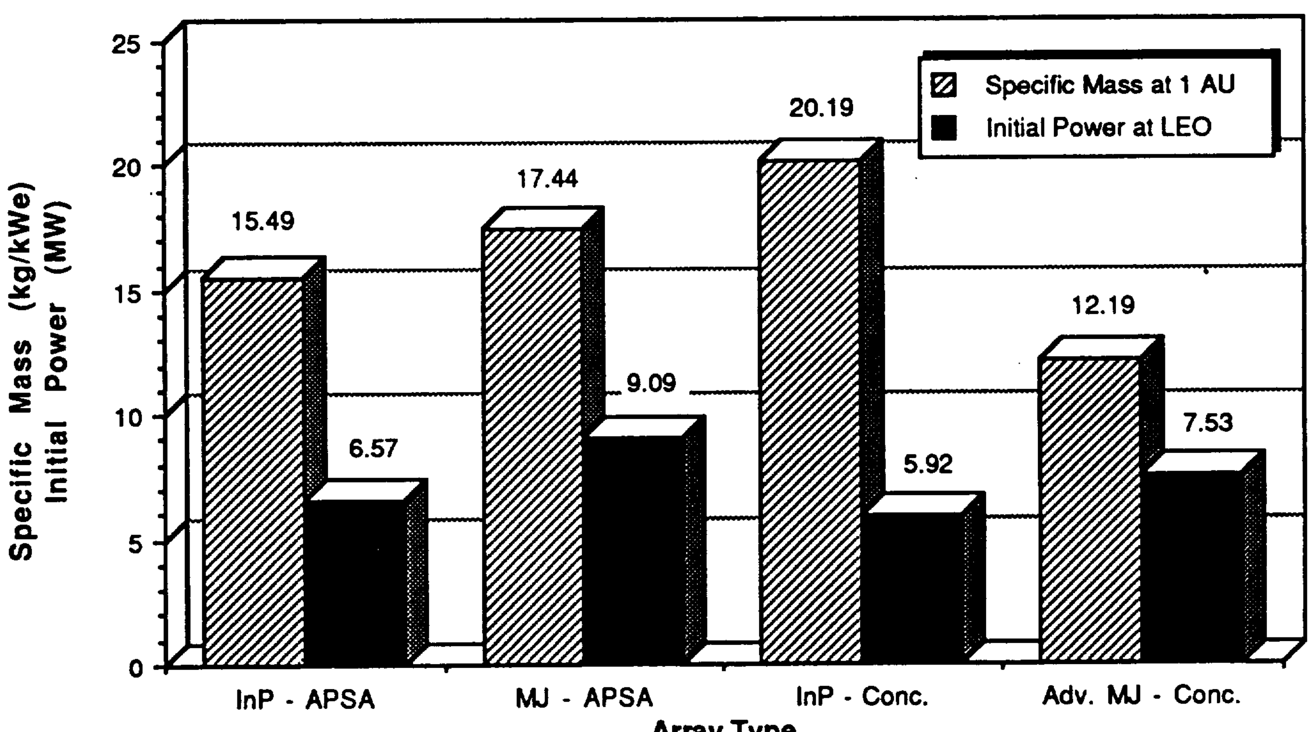


Figure 6 Comparison of specific mass and initial power for various SEP power system designs

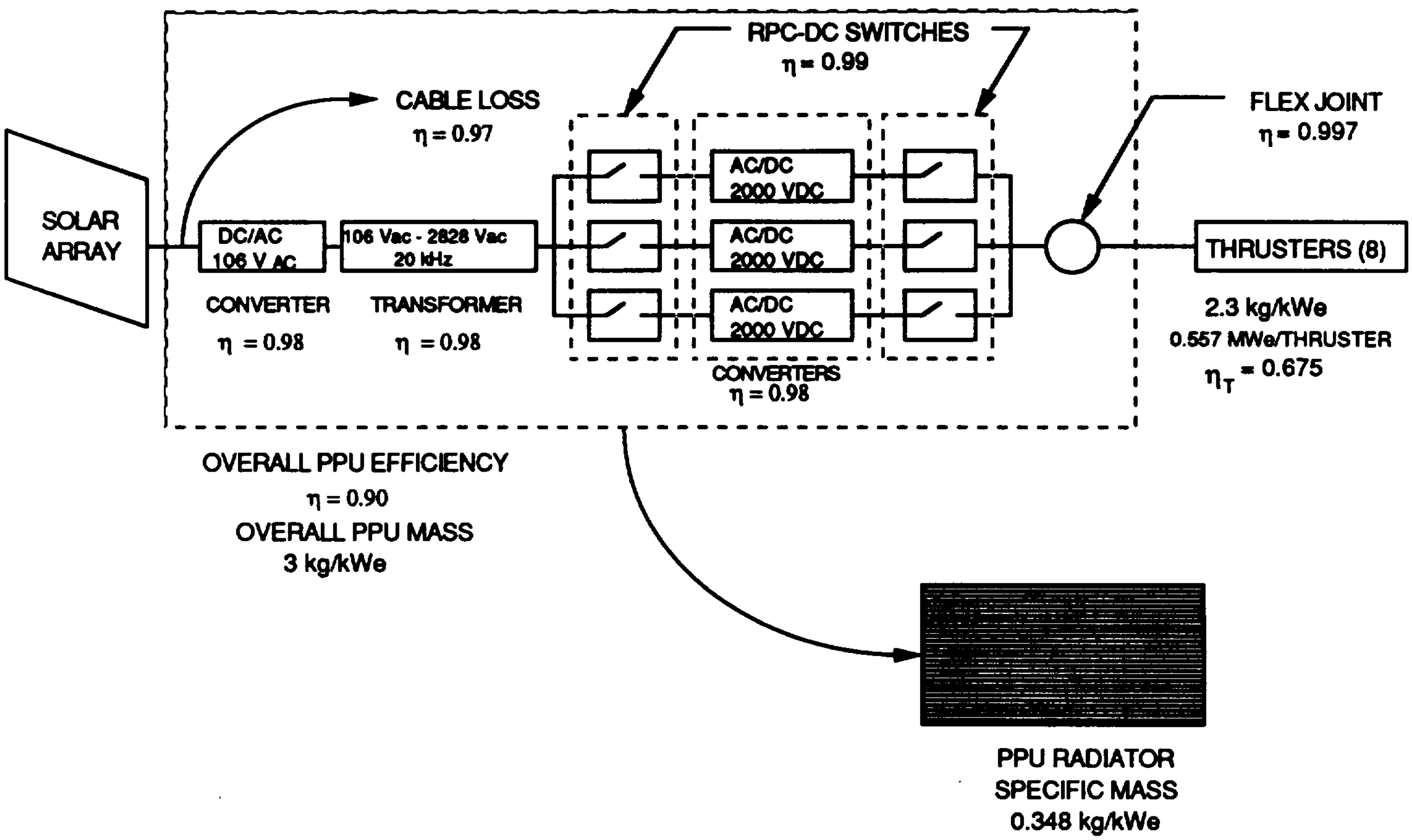


Figure 7 Solar electric propulsion vehicle power processing unit schematic

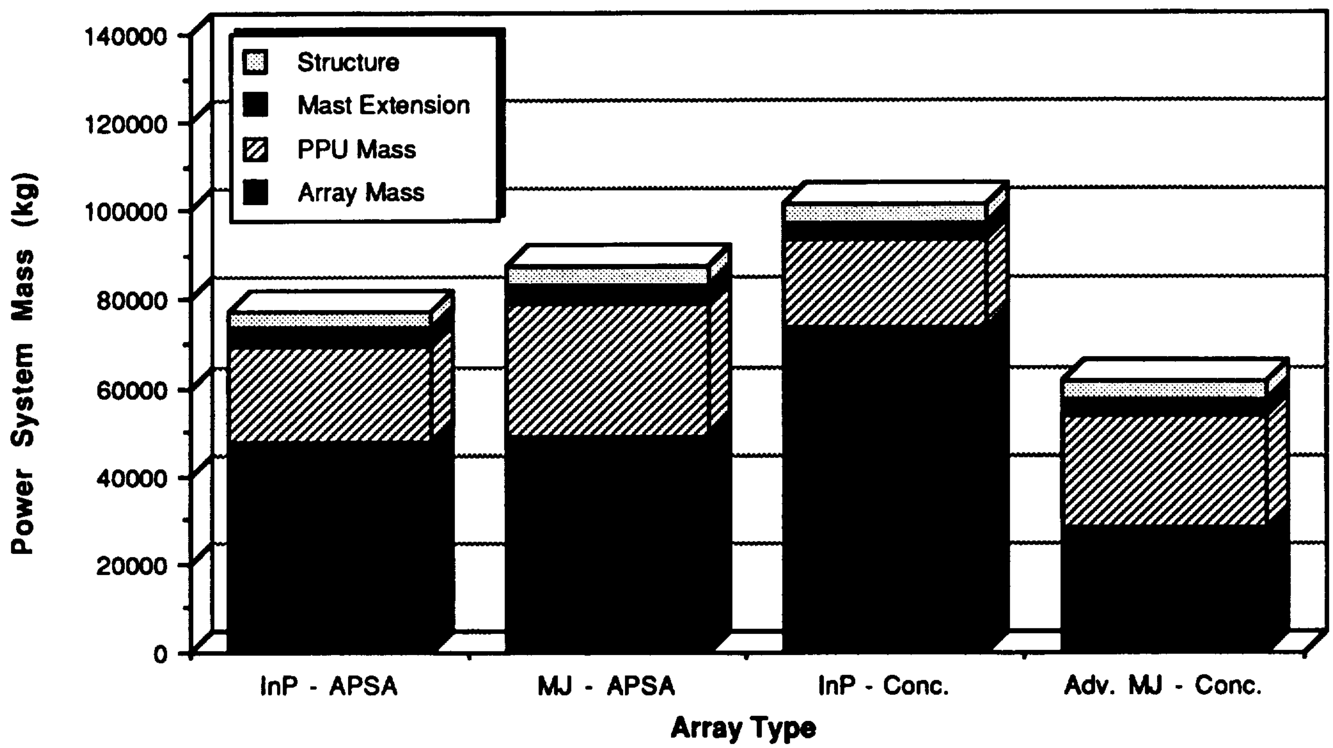
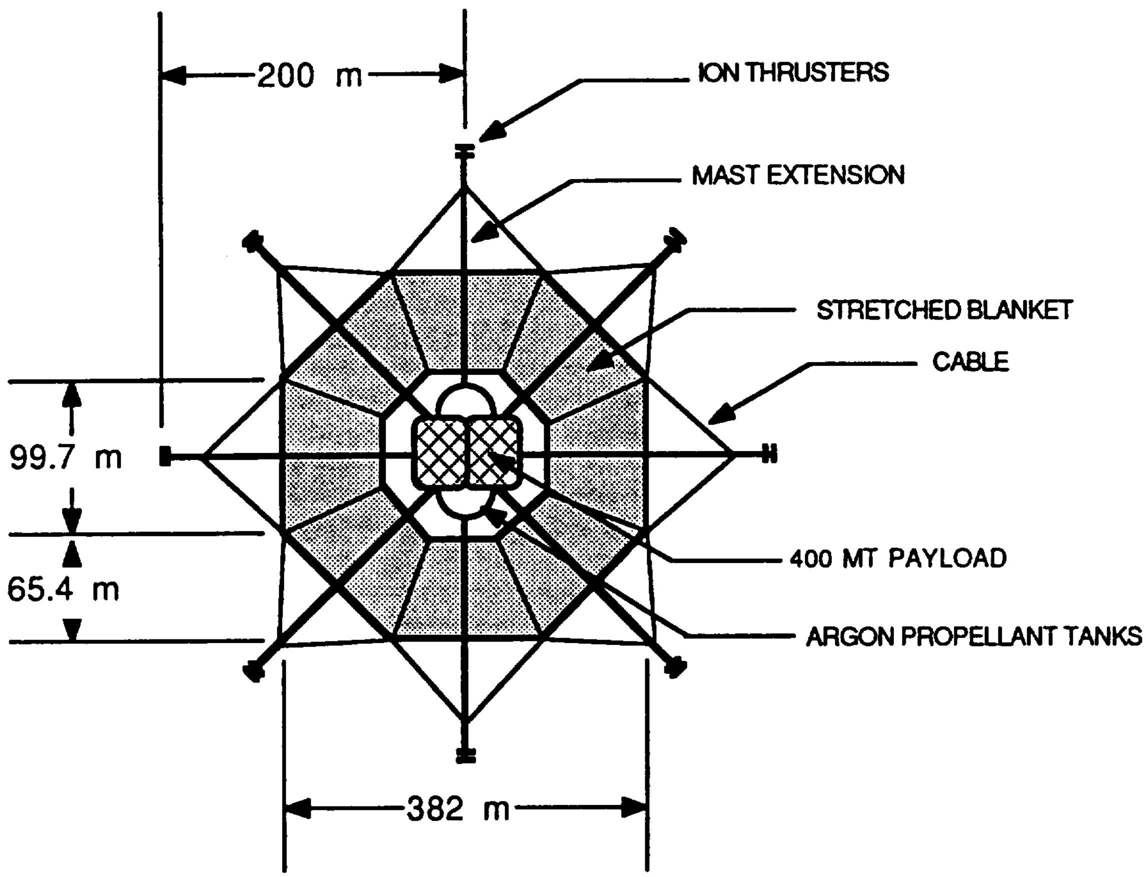
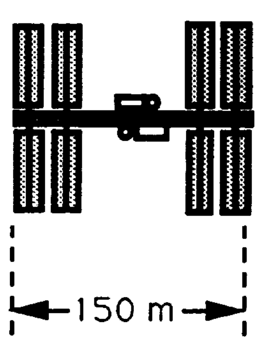


Figure 8 Breakdown of power system components

SEP VEHICLE

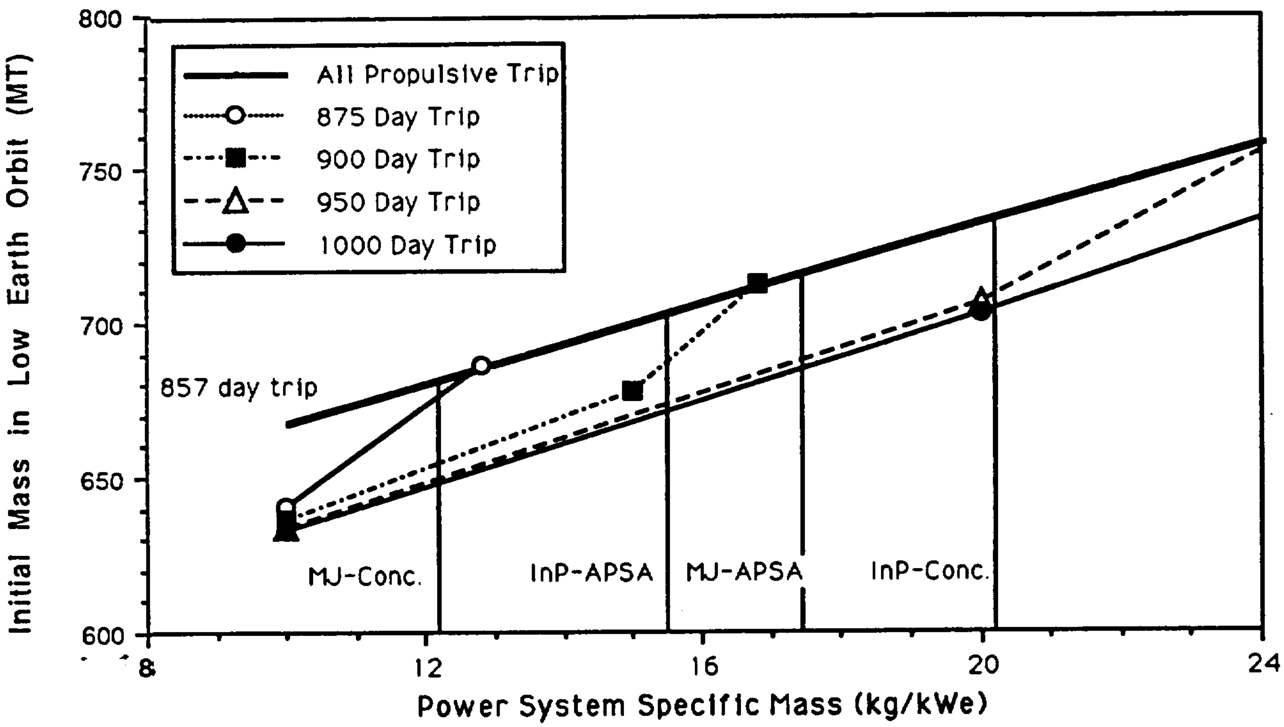


SPACE STATION FREEDOM IOC



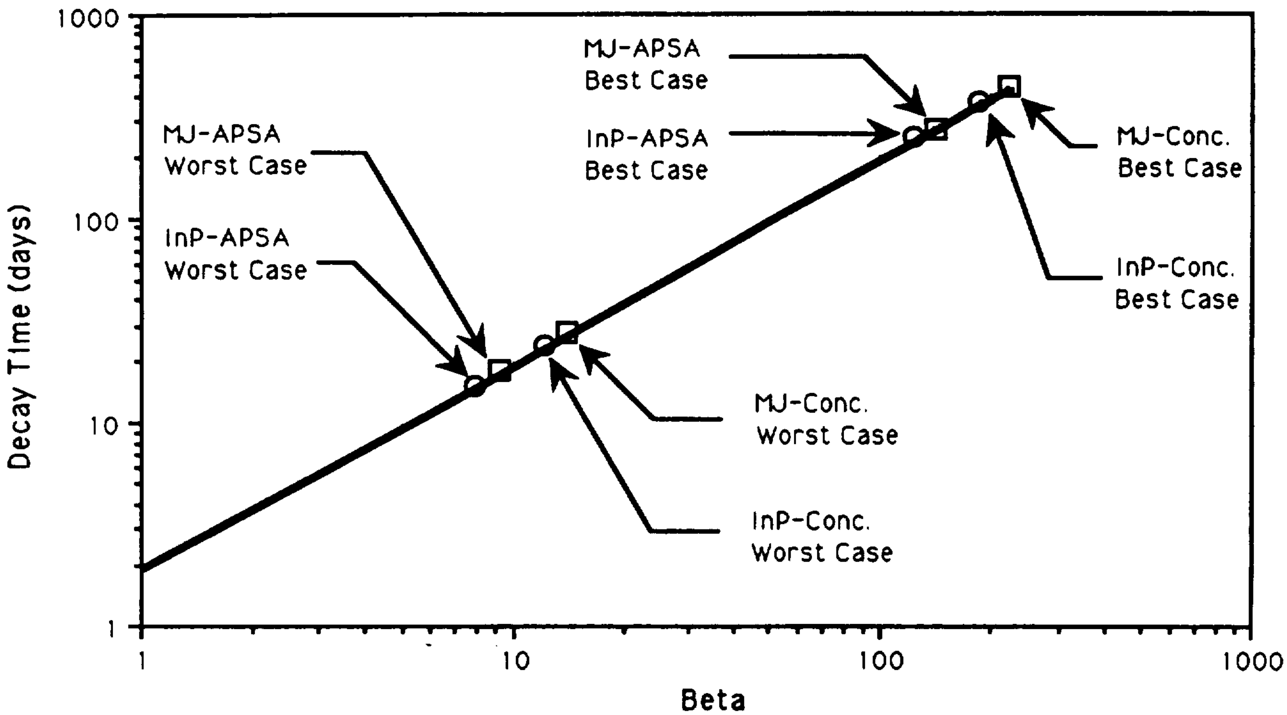
SEP VEHICLE PERFORMANCE	
IMEO	700-670 MT
TRIP TIME	897-1000 DAYS

Figure 9 SEP vehicle configuration for a 38000 square meter array design.



Propulsion System Specific Mass = 2.34
Orbits: Earth-408 Km, Mars-Areosynchronous Circular
Isp = 5000, Payload Delivered = 400 Mt
One-way Trip

Figure 10 SEP cargo vehicle performance



Beta = Mass/(Area*CD)

Best case CD = 0.136: Plane of SEP vehicle parallel to the direction of motion through the atmosphere	Worst case CD = 2.11: Plane of SEP vehicle normal to direction of motion through the atmosphere

Figure 11 Orbit decay time for SEP vehicle concepts from 400 km to 100 km orbital altitude (U. S. Standard Atmosphere, 1962)

2.1.8

HUMAN-NUCLEAR RADIATION ISSUES FOR EXPLORATION MISSIONS

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Study prepared for
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the NASA Space Station Evolutionary Working Group

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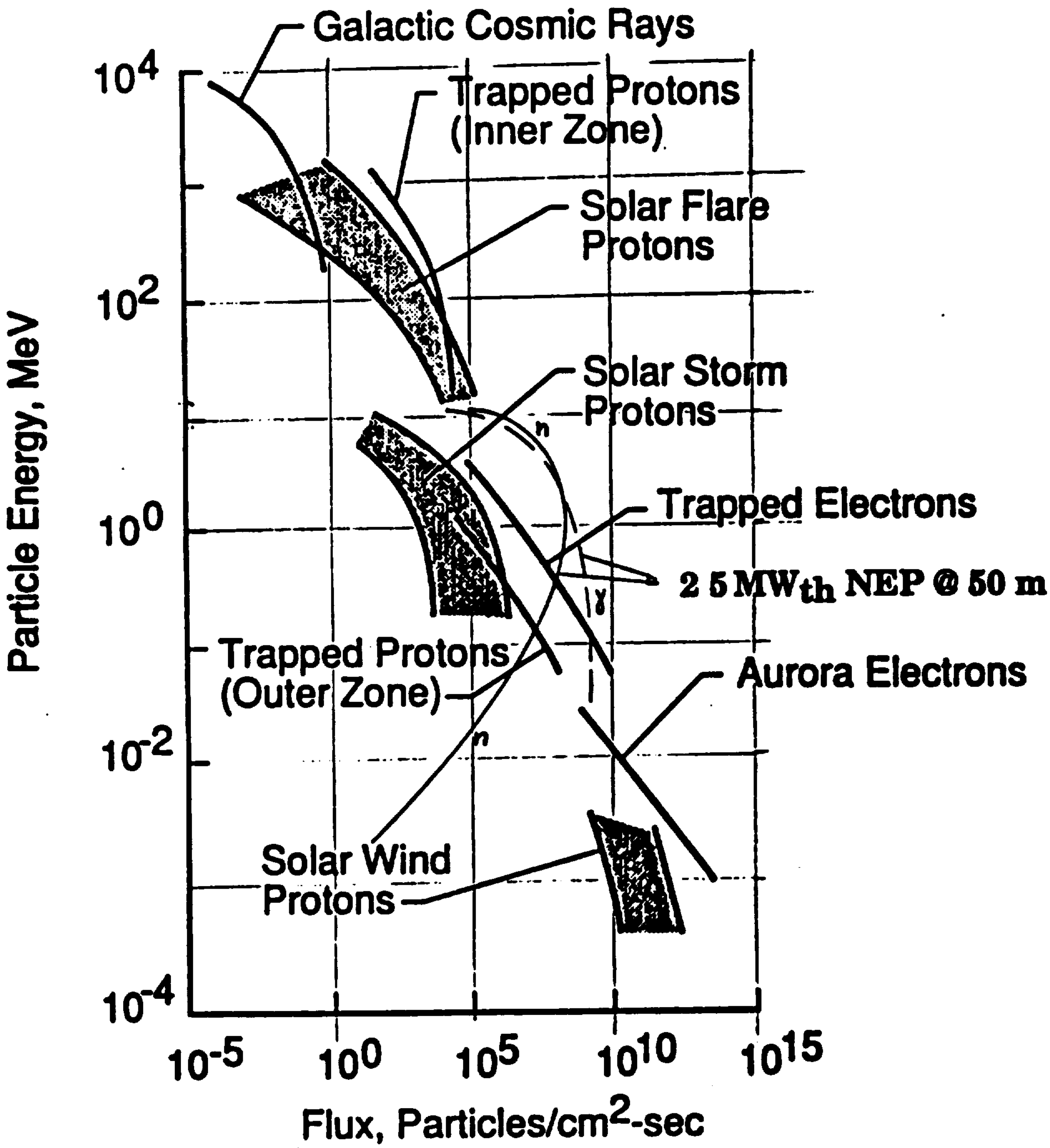
Objective: The objective of this study was to identify the issues and ramifications of human proximity to nuclear systems, in the context of space exploration and its natural radiation hazards. The scope was the entire spectrum of human presence across a complete range of man-made nuclear sources.

Conclusions: We have reached the conclusion that humans can safely function in every nuclear situation examined, provided that sensible safety measures are taken. Further, these safety measures are reasonably achieved for all cases of current interest in the exploration of the Moon and Mars. In many cases, the natural radiation hazards appear more menacing than do the man-made hazards, yet both demand the utmost respect. Rather than being part of the radiation problem, nuclear systems can be an important part of systems oriented solutions to overall human radiation issues.

The Radiation Hazards: Space is far from empty. It abounds with energetic particles. These natural nuclear radiations span an enormous range of energies and number densities. Man-made radiation hazards to astronauts are put in their most accurate perspective by examining them in their number-energy context as done in the first figure, Space Radiation Environment. The natural environments have been plotted by Drs. Townsend and

SPACE RADIATION ENVIRONMENT



Wilson of Langley Research Center (reference 1), while the reactor curves (labeled "25 MW_{th} NEP @ 50 m") were computed from General Electric data to represent a nuclear electric vehicle (NEP) which uses a ten scale SP-100 class reactor.

This figure can help us understand many things. The worst problems would be in the upper right, very high fluxes of very energetic particles. Fortunately, such radiation does not exist within our solar system. The highest energy nuclei, that is the galactic cosmic rays (GCR), are extremely penetrating and present the most difficult shielding problems. As the cosmic rays interact with the spacecraft, they create copious secondary radiations which also cause great biological concern.

It can also be observed in this figure that the inner belt of trapped protons is worse than a continuous bombardment of worst case solar flare protons. Radiations in the lower right of the figure are of no real concern for humans, since they can be easily shielded. Reactors fall in the intermediate range, not the toughest of our shielding challenges nor the easiest.

Some distinctions between the natural and reactor radiations are noteworthy. Reactor neutrons and gammas are neutral, therefore are more penetrating and require more shielding material than equally energetic charged particles. But, reactor environments are uniquely directional and are the only ones we are free to adjust before shielding, by changing our distance from the source. And finally, the gammas are less readily absorbed in the human body than high energy charged particles and they do relatively less biological damage per unit energy absorbed per unit mass of tissue.

When radiations collide with matter, whether in a shield, in equipment or in a person, they lose energy and create a variety of secondary radiations. The orderliness shown in the figure is replaced by a complexity of different radiations at a spread of energies, each with different biological effectiveness factors.

Natural and man-made radiations do increase in commonality, however, as they go through matter. For example, GCR's unleash many high energy neutrons and cause gamma events. Similarly, the bremsstrahlung caused by trapped electrons are photons, just lower in energy than the gamma rays from reactors. These observations suggest the useful idea that the optimum shielding materials for nuclear reactors in space might also be useful for decreasing the natural radiation hazards in space. It further suggests that the optimum shielding location, even for nuclear systems, might be near the crew where it will do multiple good, rather than near the reactor.

Standards for Human Exposure: As biological research advances and the human consequences of radiation are better understood, the acceptable limits for human radiation exposure have gotten more stringent and more specific. The National Council on Radiation Protection (NCRP) has proposed new standards shown in Tables 1.1 and 1.2 (reference 2), which are expected to be adopted by NASA. Career whole body dose limits protect against cancer. They vary from 1.0 - 4.0 Sieverts (Sv), or 100 to 400 rem, being the most stringent for younger females. Cancer is a stochastic effect, i.e. its probability is a function of dose. Shorter term dose limits protect against nonstochastic effects, i.e. effects for which the severity of damage is dose related, and apply to astronauts of any age and gender. Since radiations in the space environment are deeply penetrating, dose to the bone marrow and blood forming organs is of most concern. The NCRP limit for this dose is 50 rem annually, with an included maximum of 25 rem in any 30 day period.

Table 1.1 Career Whole Body Dose Equivalent Limits Based on a Lifetime Excess Risk of Cancer Mortality of 3×10^{-2}.

Age (years)	Female (Sv)[a]	Male (Sv)[a]
25	1.0	1.5
35	1.75	2.5
45	2.5	3.2
55	3.0	4.0

Table 1.2 Short Term Dose Equivalent Limits and Career Limits for Protection Against Nonstochastic Effects (Sv)[a].

Time Period	BFO[b]	Lens of the Eye	Skin
30 day	0.25	1.0	1.5
Annual	0.5	2.0	3.0
Career	See Table 1.1	4.0	6.0

[a] 1 Sv = 100 rem.
[b] Blood forming organs. Denotes dose at a depth of 5 cm.

To develop usable benchmarks for problem identification, we combined these NCRP monthly and annual limits with the expected background radiation at Space Station Freedom (SSF) orbits where much activity could take place. The worst case (WC) radiation background is about five (5) rems/month, resulting from flying at a constant 250 nm altitude during solar minimum. The best case (BC) background dose is only one (1) rem/month, achieved by following a constant drag philosophy throughout the solar cycle (reference 3). The primary source of these doses are trapped protons at the lower edge of the lower Van Allen belt.

We developed an upper benchmark by subtracting the worst case backgrounds at SSF orbits from the NCRP limits. This yielded a lower bound of available dose (LBAD) for either the short term (-st) of anything less than 30 days, or the long term (-lt) for times up to a year. The LBAD is a limit of man-made dose we could safely allow to astronauts. By exceeding the LBAD with man-made sources, we would risk exceeding the NCRP limits for total radiation. LBAD-st becomes 20 rem (25 rem/mo - 5 rem/mo) for exposures from a brief instant up to a month. LBAD-lt was derived assuming an extended crew rotation of six months at the SSF. LBAD-lt becomes 20 rems (coincidentally) over the six months (50 rem - 6 mo x 5 rem/mo), or about 3.3 rem/mo.

Our lower benchmark, one rem per exposure, is the same exposure as one month's natural background (best case) at SSF. Doses from man-made sources below this threshold become lost in the total radiation received by even the best protected astronauts. It makes little sense to reduce man-made radiation below this

level at the cost of other mission risks or penalties.

The ALARA (As Low as Reasonably Achievable) principle is still strongly advocated by the NCRP. Our benchmarks attempt to define "reasonable" within a space operations context which involves a high amount of natural radiation unparalleled in terrestrial situations. Exceeding any LBAD dose would be unreasonably permissive. Similarly, driving the man-made dose below some low level can be unreasonably stringent, if it imposes mission penalties. If no unreasonable penalties are involved then of course man-made doses should be driven as near to zero as possible. The specific values above apply only to space operations in low earth orbit. For deep space astronauts where background doses are even higher, the ALARA principle must be applied to total radiation in each specific context. These LBAD's are smaller, therefore our range of options is smaller.

We might envision each astronaut as having a radiation budget. When it is exceeded he or she must return home. If the background radiation is kept low through prudent operations or with well protected habitation structures, then radiation is hardly a factor in crew rotation. If the maximum SSF background doses are allowed, then the combined radiations can impact crew rotation. Sources causing an LBAD-lt dose would limit rotations to six months, by definition. Any additional dose, such as incurred by servicing a nuclear system, would further limit time on orbit. In essence all radiation sources compete for the same dose budget, and solutions should be directed at all sources including natural radiation.

<u>Man-made Sources Versus Human Presence:</u> The table of results on the next page summarizes the situation across the spectrum of human nuclear issues. The dimensions of the problem are the complete range of nuclear power (shown vertically) and the span of increasing human presence (shown horizontally). Reactor operating assumptions are designated in the corner of each box, either as "O" for operating or "SD" for shut down.

NASA — LEWIS RESEARCH CENTER

HUMAN-NUCLEAR RADIATION ISSUES - RESULTS

Increasing Human Presence →

Radiation Source	Brief Service (hrs.) in Proximity	Extended Activity (wks.) in Vicinity	Habitation in Vicinity	Habitation on Board
Radioisotopes	O	O	O	O
Small Reactors < 1 MW_{th}	SD	O/SD	O	O
Large Reactors ≥ 1 MW_{th}	Node: SD	Node: O/SD	Node: O	Node: O
	Surface: SD	Surface: O/SD	Surface: O	Surface: N/A
Nuclear Electric Vehicle	SD	SD	O/SD	O
Nuclear Thermal Vehicle	SD	SD	O/SD	O

Safe under standard practices.

Safe in and when special precautions are taken.

Hazards to be avoided.

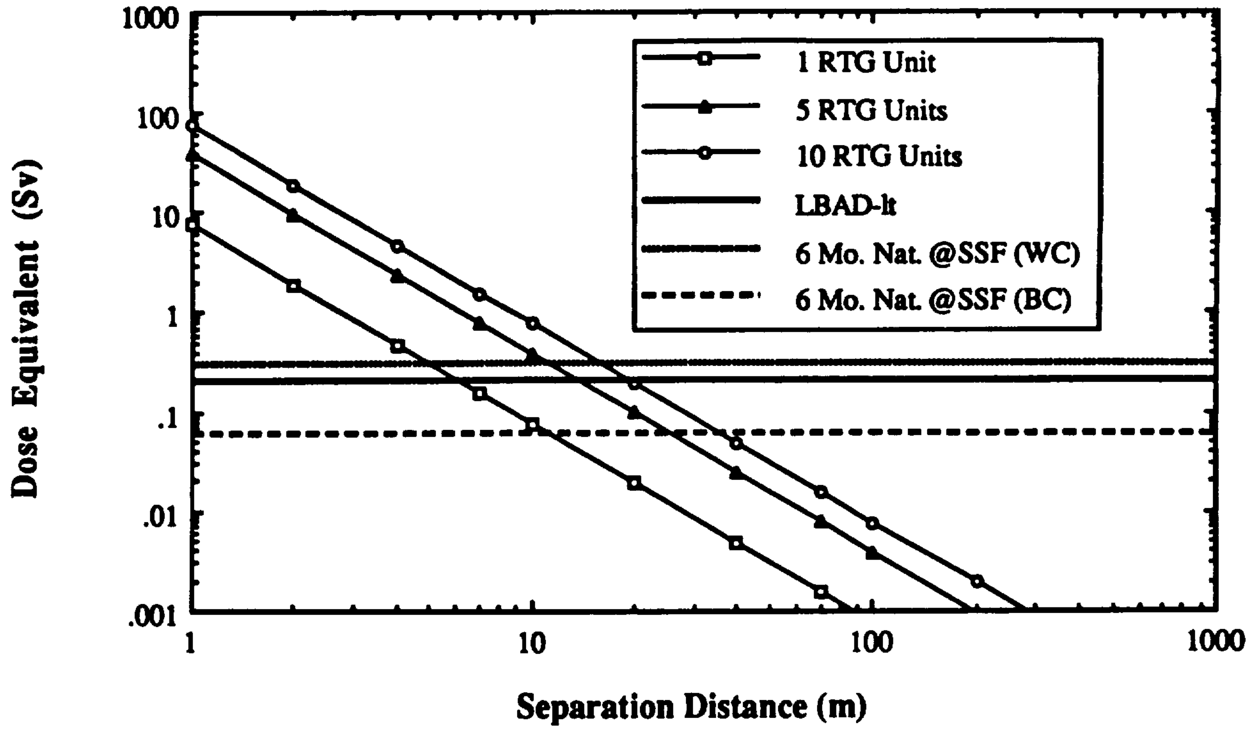
Special Concerns

Only two of these circumstances are inherently radiation safe to the operators without special precautions. These are ironically at opposite ends of the power-presence spectrum, short term involvement with radioisotopes and riding the very high power nuclear thermal rockets (NTR). This is not to imply that safety is automatic. Designers and fabricators must still assure that such hardware is built to its full safety potential. In all other situations, the operators must adhere to specific safety guidelines. The "special concerns" backshading in the chart is somewhat subjective, but it refers to situations which may impose stricter design and operational constraints.

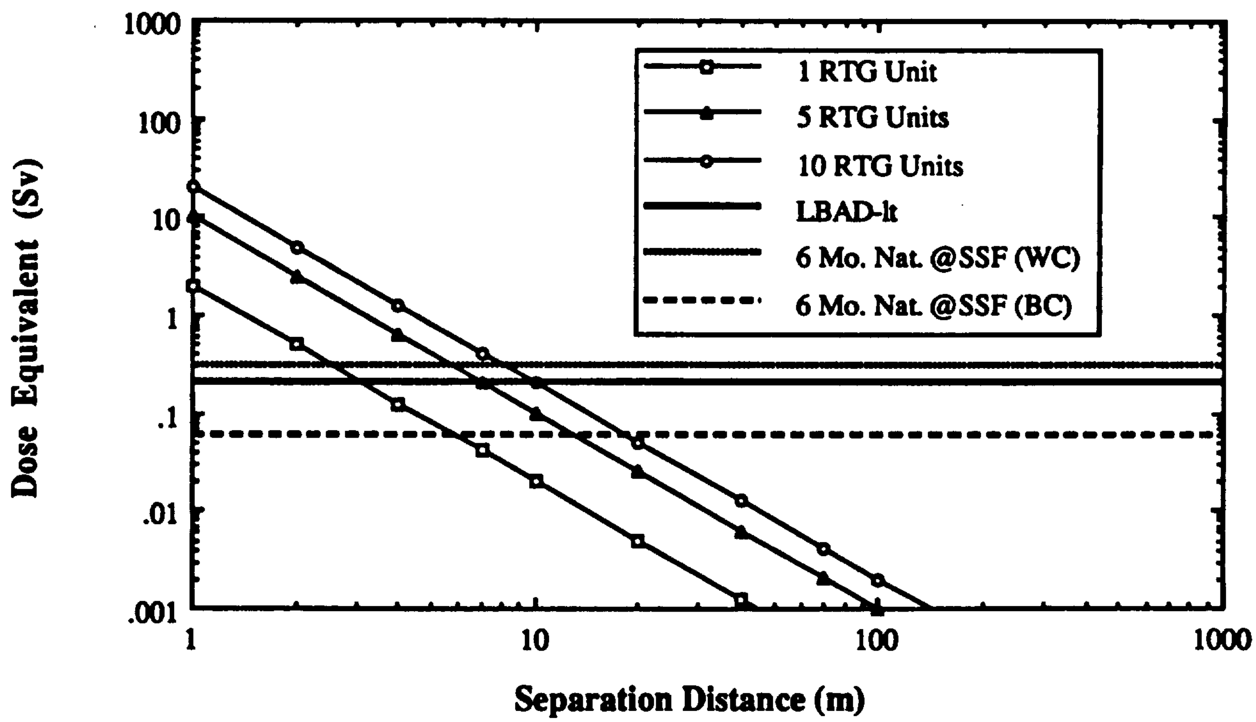
Radioisotopes: The radioisotope human interaction issues can be understood with help from the figures on the next page. Strict standards of cladding integrity (to withstand the rigors of launch and reentry) assure that they can easily be handled without radiation risk. The alpha and beta radiations from these isotopes are completely shielded. However, low levels of neutrons and gammas are also emitted and represent a potential external radiation hazard. The plotted data for 5 kW_{th} radioisotope thermionic generators (RTG) units is derived from dose rates from Normand, et al (reference 4).

Multiple RTG's within a few meters of people for several months would constitute an operational concern. Long term storage or use of RTG powered devices in or near an inhabited area could be unacceptable. On interplanetary transports this imposes a shielding or distance requirement if numerous RTG's are needed. On the space station structure, however, a safe storage site could be found 50 meters or more away, much further than would be needed. Allowable quantities of RTG's and duration limits, as shown in the figures, are based on commonly used Pu-238 isotope systems. We need different limits for other isotope systems, or for Pu-238 with different amounts of impurities, which have different neutron and gamma emission rates. Under changed circumstances other isotopes may be considered, and they may be more or less compatible with human presence.

180-Day Dose: Dose vs. Distance from RTG Centerline in the Radial Direction
[Source: Normand et. al (1989), 6th Symposium on SNPS]



180-Day Dose: Dose vs. Distance from RTG End in the Axial Direction
[Source: Normand et. al (1989), 6th Symposium on SNPS]



Power reactors: Power reactors of various sizes would be extremely enabling on lunar or planetary surfaces, on space transports or at orbital nodes (either attached or as free flyers). As we shall see, there are a variety of very workable options for shielding them.

Surface nuclear power: Small reactors for lunar and planetary surfaces were investigated at NASA Lewis Research Center in reference 5. Four shielding options were considered: distance, earth-assembled shields, stacked lunar/planetary soil and excavations. The primary reference dose rate was 5 rem/mo. This 20% of the NCRP one month dose limit, apportioned to the "controllable" man-made sources, was chosen in anticipation of a preliminary NCRP guideline. (The NCRP later decided that it was unwise to make any distinction with respect to dose source.) This 5 rem/mo dose rate would not be unreasonable for short missions, but would be too high for long missions. The results show it will be easy to achieve much lower dose rates than this.

Shielding trades can be understood by comparing the alternatives for a single reactor. A 500 kW_{th} unshielded reactor can meet the 5 rem/mo level at 5.6 km. This result will scale inversely proportional to distance squared and approximately linearly with power. If earth manufactured shadow shields are used, the shield mass will depend on height and angle to be protected. At only 100 m away, a mere 300 kg shield can protect an area 40 m wide x 5 m high from this same reactor. But closer distances, larger areas and lower doses quickly increase shield masses to a few tonnes, a burden on launch capability. More lunar or martian material is needed, by comparison, to give the same benefit. A 5 m thickness of local materials would reduce this same reactor's dose rate to the same 5 rem/mo level at the same 100 m distance. Or, a 7 m thickness would give the same dose rate directly adjacent to the shield. These modest amounts of very available material should not be difficult to move into place. But the simplest solution may be to dig a hole. A hole merely one meter deep and two meters wide would give similar protection only 7 meters away!

Larger reactors of the SP-100 size (2.5 MW_{th}) for lunar bases were also considered by NASA Lewis personnel in reference 6. Circular shields of a full 360 degrees were designed to meet the same 5 rem/mo dose level for the same rationale. The shields surround the reactor, but leave the power conversion hardware and radiators outside where they can be serviced. The 5 rem/mo dose rate, taken at the edge of the radiator nearest the reactor, is comfortably consistent with a servicing philosophy. Habitation would be at a greater distance with lesser dose.

The carefully engineered, earth manufactured, highly efficient, layered shield of lithium hydride and tungsten is only 55 cm thick, but has a mass of 22 tonnes. The 7 m thick indigenous shield requires the movement and stacking of 870 tonnes of lunar material. Both approaches are doable but have undesirable heat losses between the reactor and the power conversion. In comparison, the cylindrical hole, 4 m deep x 2 m wide, would only require the removal of 45 tonnes of material. As distance increases from the hole, not only would dose drop off as $1/r^2$ but increasing amounts of intervening material would greatly attenuate dose even further.

Pivotal issues are excavation technology and the compatibility of reactors with lunar and planetary excavations. These are engineering issues with human implications. Operating reactors in excavations involves special thermal design and the handling of backscatter neutrons. The heavier elements in the lunar regolith will reflect (scatter) neutrons back to the reactor and change its criticality and control. On Mars, neutron scatter would occur, but the variable presence of water permafrost would also moderate the neutron energies in a less predictable way. These moderated neutrons are more likely to affect criticality.

The Lewis study selected a boral (boron carbide strengthened by aluminum sheets) bulkhead to act as a neutron absorber, as well as to prevent lunar dirt and dust from falling onto the reactor. A supporting trade-off study (reference 7) had found this side

and bottom bulkhead would also reduce the amount of neutrons emerging from the top of the hole. This same study had also quantified the benefits of making the hole relatively deep and narrow. The effectiveness of this bulkhead for criticality control and the best means of heat rejection in such a tight volume are still subject to detailed engineering verification.

In contrast to the Lewis concept, a University of Washington study group adopted a drastically different approach to deal with neutron backscatter from lunar materials (reference 8). To insure their 10 MW_{th} reactor would operate both in space and on the lunar surface without changes to the control mechanism, they raised the entire reactor above the lunar surface and shielded it from below. This required a 15 m tall support tower and increased the radiation in the vicinity of the reactor. To use natural material to reduce radiation at greater distances, they required the system to be placed in a deep, preferably wide, existing crater. Their fifty tonne, minimally shielded system devotes about a third of its mass to shields and structure, even if an ideal crater can be found.

In summary, surface nuclear power can be safely implemented in a human environment. Even unshielded reactors are safe if large keep out zones are enforced. Earth manufactured shields to protect small areas are light. They become less economic if larger safe zones and/or shorter power transfer distances are desired. Indigenous material can make good shields. As soon as reactor-hole compatibility and hole construction issues are solved, excavations will be by far the best way of using this native material. Both near vicinity habitation and human servicing of the power system will become viable without reactor radiation being an issue. If reactors are placed on the surface, then earth delivered masses will be greater and/or keep out zones will be larger. If reactors are raised above the surface, then delivered masses will be even larger and maintenance much more restricted. But in all cases, nuclear power will still be an extraterrestrial energy bargain.

Collocated orbital nuclear power: Three nuclear power options for growth space station were considered in reference 9: an 8 MW_{th} reactor on a 60 m boom extension, a similar reactor on a 2 km conductive tether and two 4 MW_{th} reactors on 100 m boom extensions. Each would deliver 300 kW_e of electrical power even at low conversion efficiencies. With dynamic conversion they could deliver several megawatts. For each configuration dosages were computed for a mix of human activity: in the station, during extravehicular activities (EVA) and in the Shuttle during approach, docking and departure. Conservative (pessimistic) assumptions were made regarding EVA and flight activities. Ballast masses needed for the tether and single reactor concepts were postulated to be spare reactors, so redundancy was available in all concepts. Disposal options for the spent reactors were also included in the trades.

The design point for the tether length was taken where the dose from its instrument shield was equivalent to that from the other "man-rated" shields. The shield is less than 20% of the mass of the tether system. The hard boom concepts required shield masses on the order of 75% of the total mass.

The assessment confirmed the feasibility of installing, operating and disposing of all three concepts. The shielding and other hardware required to keep total reactor-attributable dose rates to a very safe 5 rem per three month period (one SSF crew rotation) was reasonable in each case, despite the fact that the reactor(s) are relatively large. Nuclear power can indeed be made human compatible on an orbital vehicle. It does place specific constraints on proximate operations, but these constraints do not seem to have any adverse effects.

Of the three options, the tethered mount was by far the most mass efficient. At 21 t, it was less than half the mass of the dual boom concept (45.3 t) and not even a third the mass of the single boom concept (69.1 t). Additionally, it has three to four times less disposable mass, only 10.5 t, which simplifies

the disposal of the spent reactor to any final destination. These dramatic mass benefits are the result of trading relatively heavy shielding for a small, instrument-rated shield and a light, long tether. The tether concept may also optimize even more favorably with a longer tether.

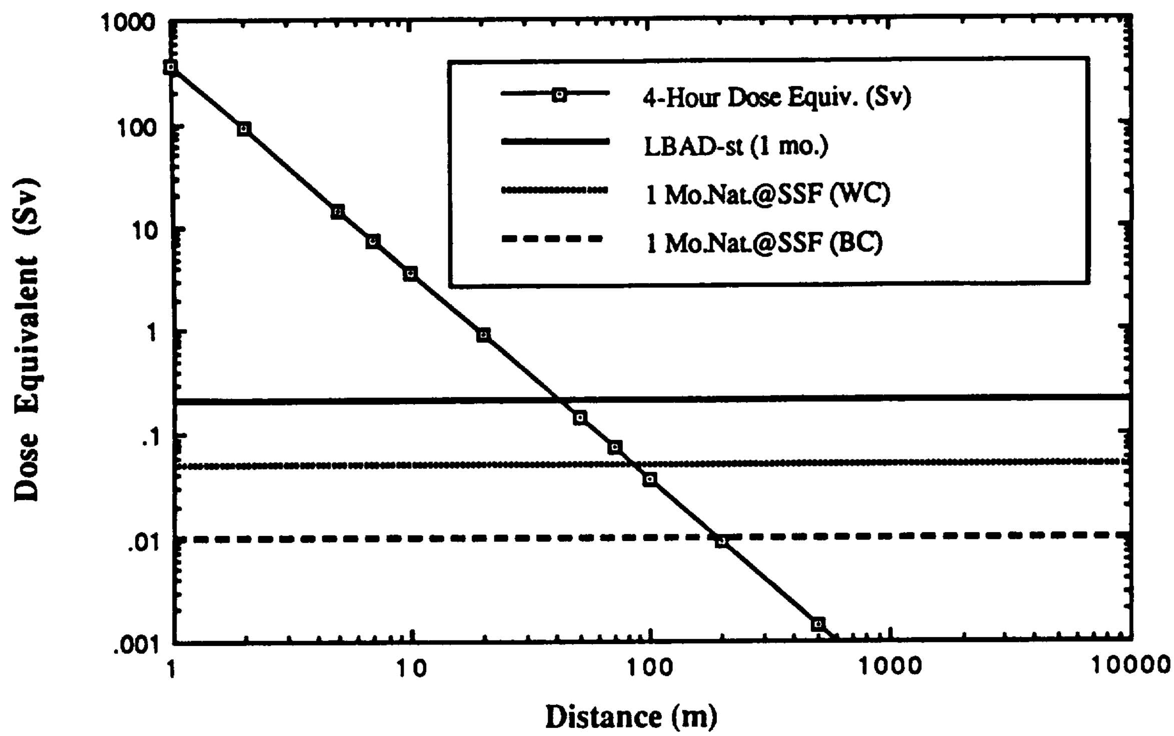
Radiation effects clearly do <u>not</u> preclude the use of nuclear power on inhabited orbital facilities. The overall feasibility of such power systems probably depends on other issues, such as the dynamics of long tethers or booms in interaction with station maneuvers. This study found it was easy to implement a case where the reactor dose was only one third of the relatively low background dose in low earth orbit. If a nuclear powered orbital facility were envisioned in lunar or martian space, the habitat would need to be shielded much more thoroughly from natural radiation. Hence, reactor dose would become even smaller, both absolutely and relatively.

<u>Free flyer nuclear power:</u> If a reactor collocated on an inhabited orbital facility is a viable option, then clearly a reactor on an uninhabited free flyer also gives acceptably low radiation to occupants of a neighboring space station. To get an idea of worst case constraints, we examined the radiation from a bare SP-100 reactor operating continuously at 2.4 MW_{th} near a space station. The two figures on the next page, based on source data from General Electric, show our results.

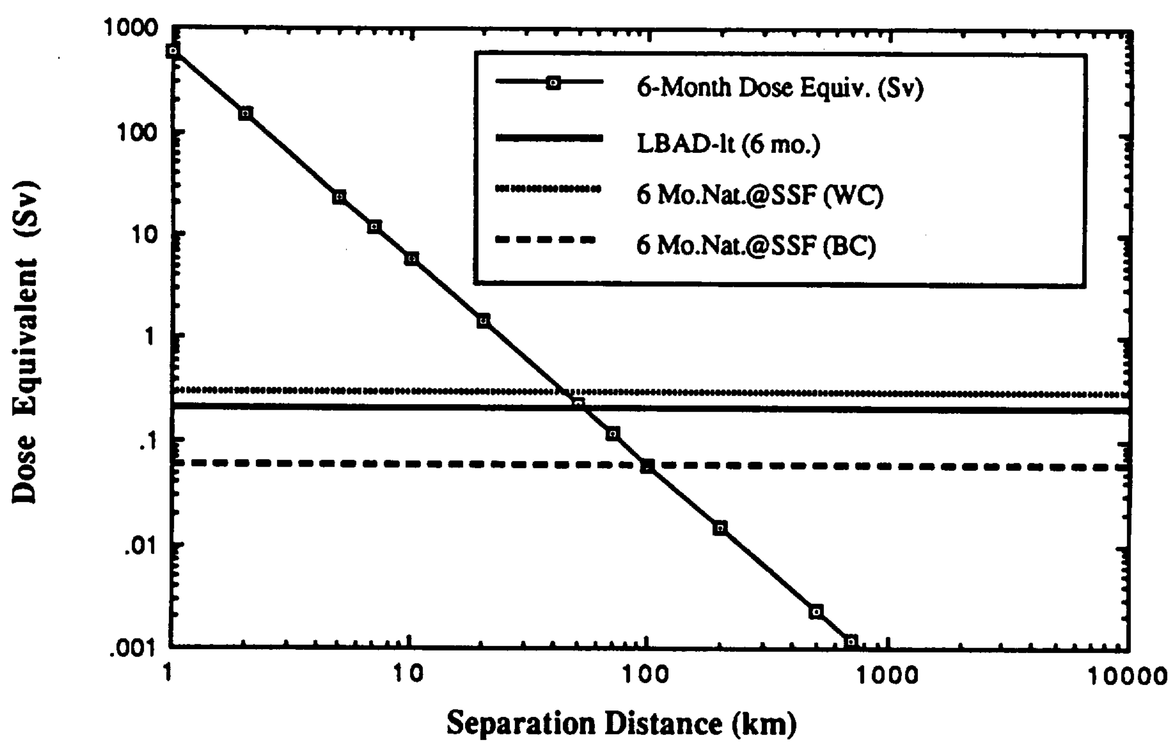
Can this free flying facility be safely visited by an EVA astronaut without shutting down the reactor? Yes, if the EVA takes place within the shadow of the instrument shield. The "4 Hour EVA: ..." figure shows that if the astronaut stays 50 m or further from the reactor he or she is within the NCRP limits. At 100 to 200 m, the dose received is only the equivalent of another month's background dose of living on the station.

Since it is not really desirable to approach the NCRP limits without good reason, is an additional shield warranted?

4 Hour EVA: Dose vs. Distance from Operating SP-100 Reactor Inside Shadow Shield, Reactor-Only Source Term



6 Month Dose: Dose vs. Distance from Operating SP-100 Reactor Outside Shadow Shield, Reactor-Only Source Term



Probably not a dedicated shield. Any real platform will have substantial mass. If it is a water electrolysis platform, for example, the hydrogen and oxygen are terrific neutron shields. The best solution is to let the inherent platform mass be the man-rated shield. This solution must, of course, be implemented on a case by case basis.

When does the free flyer pose a radiation hazard to its space station neighbors? We calculated the case in which it is not practical to keep the shadow of the instrument shield oriented toward the space station. The "6 Month Dose: ..." figure shows our results. A 50 km separation is required to meet the very restrictive long term LBAD. A 100 km separation is needed before the dose is as comfortably low as background dose.

These distances seem further than desirable, and at least three shielding solutions come to mind. Perhaps the free flyer should be designed to keep the shadow of the instrument shield oriented toward SSF. This would require a stable platform in a matched orbit directly leading or lagging the SSF. Or, a modest four pi shield could be used. Of course, this shield would be much lighter than for the attached reactor, since it is at a further distance. Or, perhaps best solution is still to shield with normal platform mass. Again, real solutions are case specific.

<u>Nuclear vehicles:</u> The largest reactors in space will be used to power and propel the lunar and planetary vehicles of the world leaders in space exploration. Do these vehicles pose a radiation hazard to people on a space station as they depart? Is radiation an issue to people riding these vehicles? When the vehicles return, do their hot shutdown reactors prevent them from being serviced? Are these shutdown reactors a radiation problem to their neighbors? These are the crucial questions addressed in this section.

For the original work in this study, two nuclear vehicles were defined and were later tailored to be consistent with other

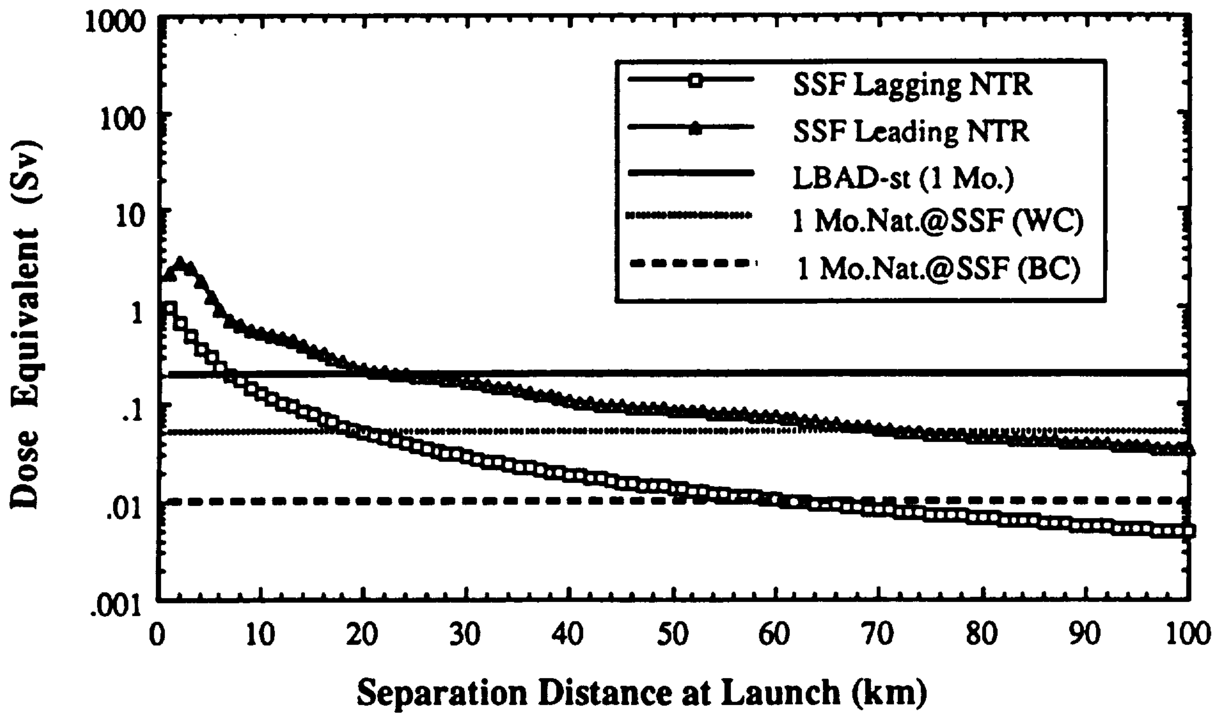
studies being done for NASA's Office of Exploration. The first vehicle was a two stage nuclear thermal personnel carrier. The first stage uses a 5000 MW_{th} Phoebus class engine, which is jettisoned after trans Mars insertion (TMI). The second stage uses a 1575 MW_{th} NERVA class engine. The other vehicle is a nuclear electric cargo vehicle which makes a round trip to Mars. It uses a ten scale SP-100 type reactor at 25 MW_{th}. With a 20% efficient dynamic conversion system it produces 5 MW_e to operate ion engines. This produces very low thrust at very good specific impulse. Mission details will be elaborated later as needed. Unless noted otherwise, all quantitative results refer to these vehicles.

<u>Departing vehicles:</u> The two figures on the next page summarize the radiation exposure to SSF personnel as either vehicle would depart from the vicinity of the SSF under a variety of conditions. These results show that there are both smart ways and dumb, hazardous ways to execute these launches.

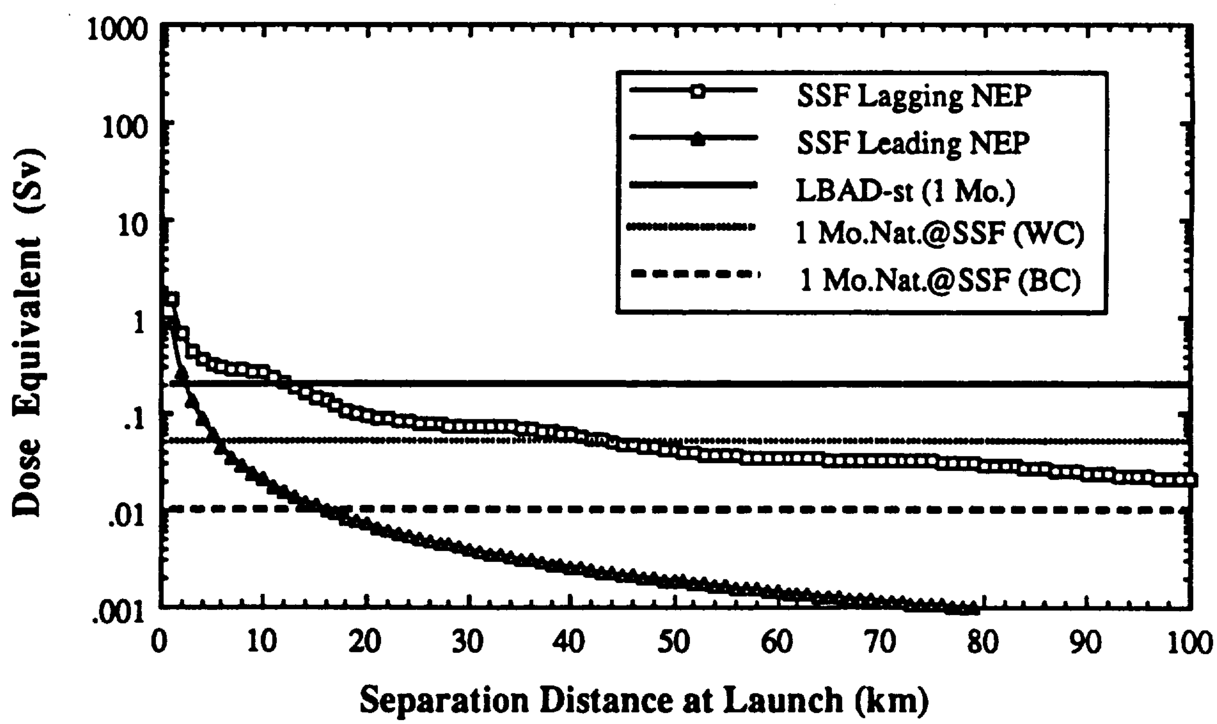
The NTR personnel vehicle uses a huge reactor and provides little shielding in a broad radial direction from the engine. The vacuum of space gives no protection either. We assumed conservatively that the line of sight to SSF was never quite sufficiently aligned to benefit from any vehicle shielding nor from any SSF self shielding. For maximum performance, this rocket aligns itself with, and thrusts along, its orbital velocity vector. If it starts behind the SSF, or any other orbiting node, it quickly flies by the node in a near miss. Radiation, collision risk and exhaust impingement are all potential problems. Launching from less than 20 km back, the radiation is clearly unacceptable, even if the trajectory were perfect. It could also be intolerable for a launch from much further back, if the trajectory were low.

The smart way to launch the NTR is with the rocket ahead of the SSF, shown by the "SSF lagging NTR" curve. Even though the exhaust goes towards the SSF and there is no shielding from the

22.5 Min Cumulative Dose: Dose vs. Separation Distance at Launch
NTR Personnel Vehicle, Outside Shadow Shield, Reactor-Only Source Term



10 Day Cumulative Dose: Dose vs. Separation Distance at Launch
NEP Cargo Vehicle, Outside Shadow Shield, Reactor-Only Source Term



vehicle, the separation is quick and continually increasing. Still, our initial separation distance must be wisely chosen from the lower curve in the graph. Eight (8) kilometers is barely tolerable, 20 km is okay, and 50 or 60 km is preferable. The ALARA principle can be implemented by launching the NTR with a lead of 60 km or greater ahead of the space station.

By contrast, the NEP cargo vehicle behaves quite differently and the smart launch strategy for it is just the opposite. The reactor is not nearly so large and it by no means zooms away. The low thrust causes a very gradual rise in the NEP vehicle's orbit. The higher orbit has a longer period, so it rises above and lags increasingly behind its original undisturbed position. If the SSF were initially lagging the NEP, the NEP would make a slow close pass over the SSF. Clearly, we must again assume unfavorable alignments and that we are afforded no free protection from either the vehicle or from the instrument-rated shadow shield. The distance of closest approach, and hence the dose, would depend only mildly on the starting separation.

The smart way to launch the NEP cargo vehicle is with the SSF leading the NEP. This way the separation is always slowly increasing in the early time. By the time the NEP loses a full period and passes overhead a few days later, the altitude separation is sufficient to trivialize the dose. Though we looked at ten days exposure, most of the dose from a smart NEP launch is received in the first few hours. Less than 2 km initial separation gives unacceptable radiation, 5 km is okay, and 15 km makes the dose to SSF personnel trivial. The ALARA principle can be implemented by launching the NEP with a lag of 15 km or greater behind the space station.

<u>Riding nuclear vehicles:</u> The radiation issues, with respect to passengers on board nuclear vehicles, are as different between NTR and NEP as they were in the node situations. The NTR burns at very high power for brief periods only minutes long, while the NEP operates continuously for many

months, but at much lower powers. Most NTR's were originally defined as personnel vehicles, so an unsafe radiation dose is merely the result of a poor specific design. Our NEP was a cargo vehicle, so if it were adequately shielded for people it would probably be overweight as a cargo ship. Both vehicles, of course, can be designed in either personnel or cargo versions.

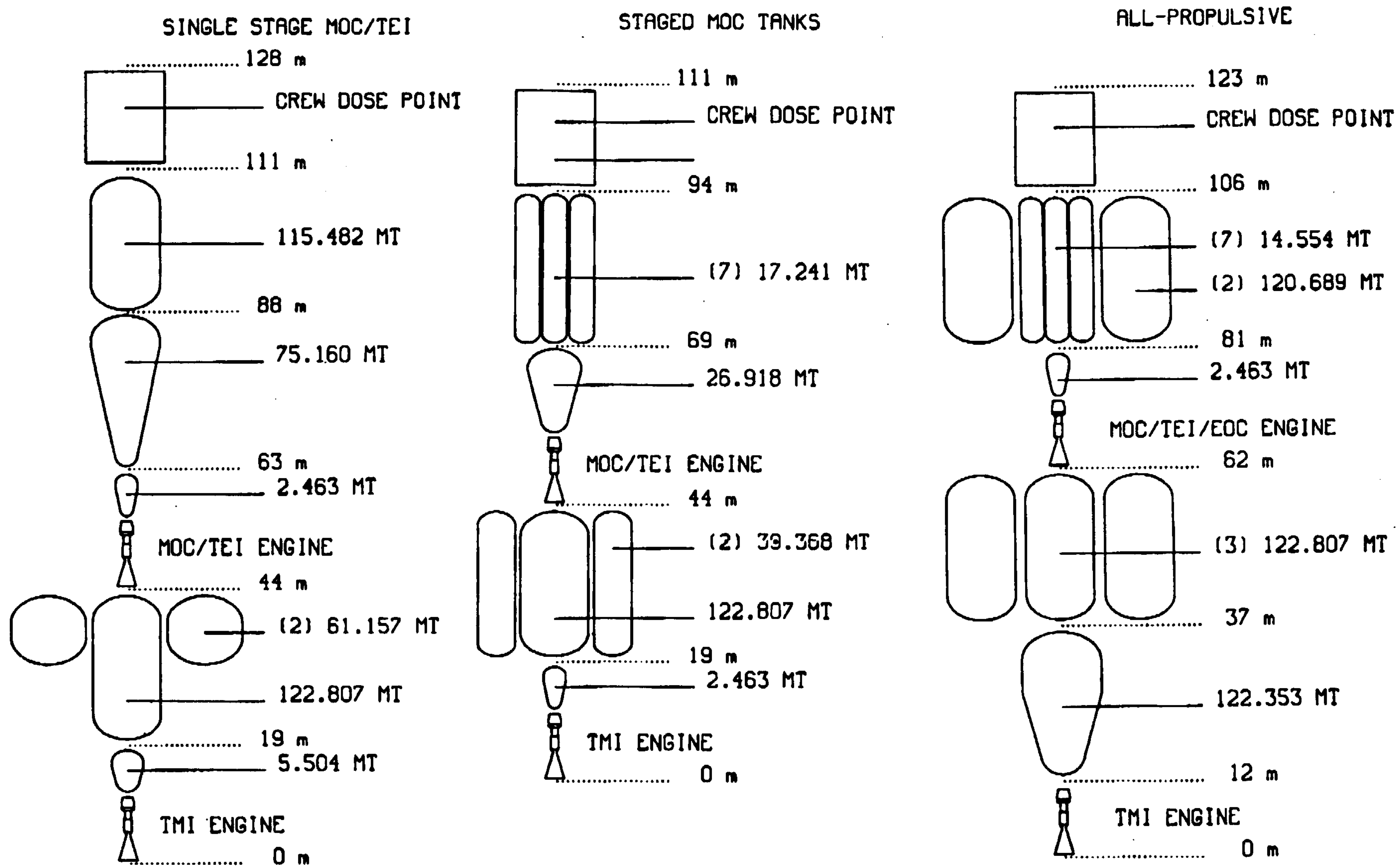
A well designed NTR personnel vehicle is safe to ride without any operator precautions, by definition. The built-in shields, tanks, propellant and its inherent length protect the crew. Three candidate configurations for the two stage NTR are shown on the next page. The table below shows the radiation doses from the NTR itself received by crew members at the specified dose points for each maneuver in a round trip to Mars. NASA Lewis personnel calculated the burn times and propellant requirements for selected Mars missions in the 1989 OEXP case studies. Idaho National Engineering Laboratory personnel used this information to define candidate configurations and to calculate the crew doses (reference 10).

Table 3. Mars Crew Doses from Nuclear Thermal Rockets.

Vehicle configuration	Trajectory class	Dose (rem) @TMI	@TMI*	@MOC	@TEI	@EOC
Single stage MOC/TEI	Sprint	<0.001	<0.01	0.03	1.0	**
Staged MOC tanks	Sprint	<0.001	<0.01	0.02	9.4	**
All propulsive	Opposition	<0.001	0.01	0.26	0.04	1.3

* Dose with disk shield removed from TMI engine.
** These configurations used an aerobrake rather than a nuclear engine for EOC.

The sprint mission is launched in September 2002, with Mars arrival in June 2003. After only a month's stay, TEI is in July 2003 and Earth return is in January 2004 for a 491 day round trip. The opposition mission launches in May 2004, arrives at Mars in April 2005 and, after a stay of up to 100 days, arrives back at Earth in December 2005. The round trip is 593 days. The burn history for the last mission will be specified later.


SINGLE STAGE MOC/TEI
128 m
CREW DOSE POINT
111 m
115.482 MT
88 m
75.160 MT
63 m
2.463 MT
MOC/TEI ENGINE
44 m
(2) 61.157 MT
122.807 MT
19 m
5.504 MT
TMI ENGINE
0 m
STAGED MOC TANKS
111 m
CREW DOSE POINT
94 m
(7) 17.241 MT
69 m
26.918 MT
MOC/TEI ENGINE
44 m
(2) 39.368 MT
122.807 MT
19 m
2.463 MT
TMI ENGINE
0 m
ALL-PROPULSIVE
123 m
CREW DOSE POINT
106 m
(7) 14.554 MT
(2) 120.689 MT
81 m
2.463 MT
MOC/TEI/EOC ENGINE
62 m
(3) 122.807 MT
37 m
122.353 MT
12 m
TMI ENGINE
0 m

Observe that only one NTR configuration gave a significant dose, and then only for one maneuver. The MOC tanks, which had been shed after the MOC burn, were really needed to shield the crew during the TEI burn. Although this ten rem dose may be tolerable, it is not reasonable in accord with the ALARA principle. Such an NTR configuration should not be selected. This case also shows how crucial the mass of the vehicle and the vehicle design are to overall NTR shielding.

Compare the trivial doses from the large TMI engines on the NTR to any natural dose in the space environment. Comparison shows that the disk shield on this stage is a needless weight penalty as far as humans are concerned. The people would be better off if the shield were traded for additional payload. Whether or not a disk shield is needed is only a propulsion system trade, since it does help protect the hydrogen propellant.

Do not be comforted by the "zero" doses at earth capture using aerobrakes. They only reflect the fact that there is no nuclear engine to give a man-made dose. The aerobraking maneuver may well deliver the crew into the Van Allen belts where they could indeed receive a substantial dose. The actual dose will always be very case specific. But in general, the propulsive braking is more flexible, can be performed above the Van Allen belts, and does not involve nearly so much radiation risk. The one rem from the rocket is a small price to pay to avoid the risk of many rem from Mother Nature.

What protection is needed to create a personnel version of the NEP vehicle? There are two approaches to protecting the crew, distance or shielding. Since forces are small, long tethers (more like fishing line than cable) can conceivably give us all the distance we need at very little mass. The dynamics of this approach need investigation. We have not quantified the shield mass needed to augment our cargo vehicle to make it into a personnel carrier of a more conventional rigid boom design. But based on size considerations, it should be less than for an NEP

manned vehicle defined by Los Alamos National Laboratory (LANL) in reference 11.

LANL's NEP vehicle was based on an in-core thermionic converter giving 11% conversion efficiency. Their 10 MW_e unit would need a reactor nearly four times as large (in thermal power) as our 25 MW_{th} cargo NEP. The shield for this reactor on a 50 m boom is only 13.3 t more for the manned version than for the unmanned. LANL's 10 rem/yr might be considered a little high, but an increase in boom length can reduce this easily. Between ten and twenty tonnes of additional shielding should convert most any cargo NEP of rigid boom design into a personnel vehicle. The more valuable trades, however, are related to the radiation balance between the reactor and natural sources. We may be able to get even more natural radiation protection from this same mass investment in shielding and boom.

Returning nuclear vehicles: Before reactors are started, they present no radiation risk at all. When nuclear vehicles arrive at or return to a human occupied facility, the risk is related to the total power produced and operational time history. We used the histories shown in Table 4 and 5. Our vehicles each completed a round trip to Mars and arrived back at space station. The first stage of the NTR had been jettisoned, so it was not a factor. A dual mode NTR was also considered in which the upper stage reactor also generated mission electrical power, since this could affect shut down radiation rates.

Table 4. Mission Profile for NTR Personnel Vehicle.

Mission phase	Duration	Power (MW_{th}) (propulsion only)	Power (MW_{th}) (Dual mode)
Coast to Mars	286 days	0.0	0.2
Mars orbital capture	40 min	1575.	1575.
Mars operations	30 days	0.0	0.4
Trans Earth insertion	35 min	1575.	1575.
Coast to Earth	170 days	0.0	0.2
Earth orbital capture	23 min	1575.	1575.
Earth orbital arrival	variable	0.0	0.0

Table 5. Mission Profile for NEP Cargo Vehicle.

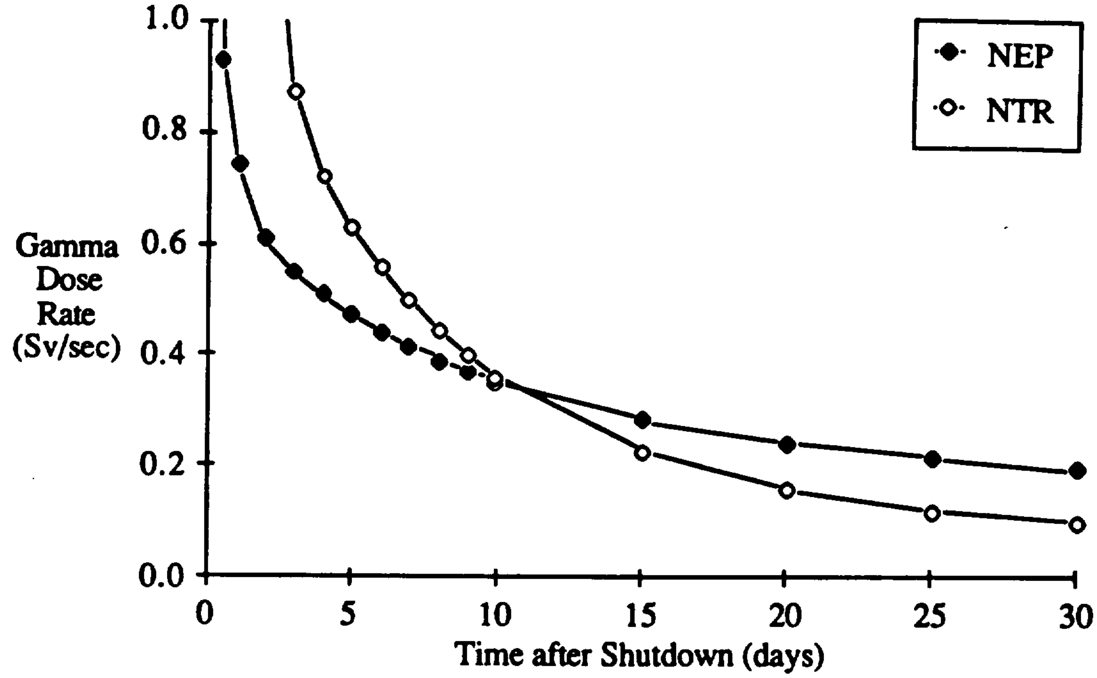
Mission phase	Duration (days)	Power (MW_{th})
Earth spiral out from 450 km	443	25
Heliocentric trip to Mars		
Thrust	253	25
Coast	162	0.2
Thrust	85	25
Mars spiral in	86	25
Mars operations	150	0.4
Mars spiral out	39	25
Heliocentric trip to Earth		
Thrust	74	25
Coast	211	0.2
Thrust	68	25
Earth spiral in to 450 km	239	25
Earth orbit arrival	variable	0.0

Servicing returned nuclear vehicles: When a vehicle arrives at some adequate distance from a human occupied facility, the reactor is completely shut down. The neutron radiation stops quickly, leaving the gammas as the only escaping radiation from the fission product buildup. After a variable cooldown time, the vehicle can be visited by astronauts for unloading or service. The four figures on the next two pages show the results of this trade-off.

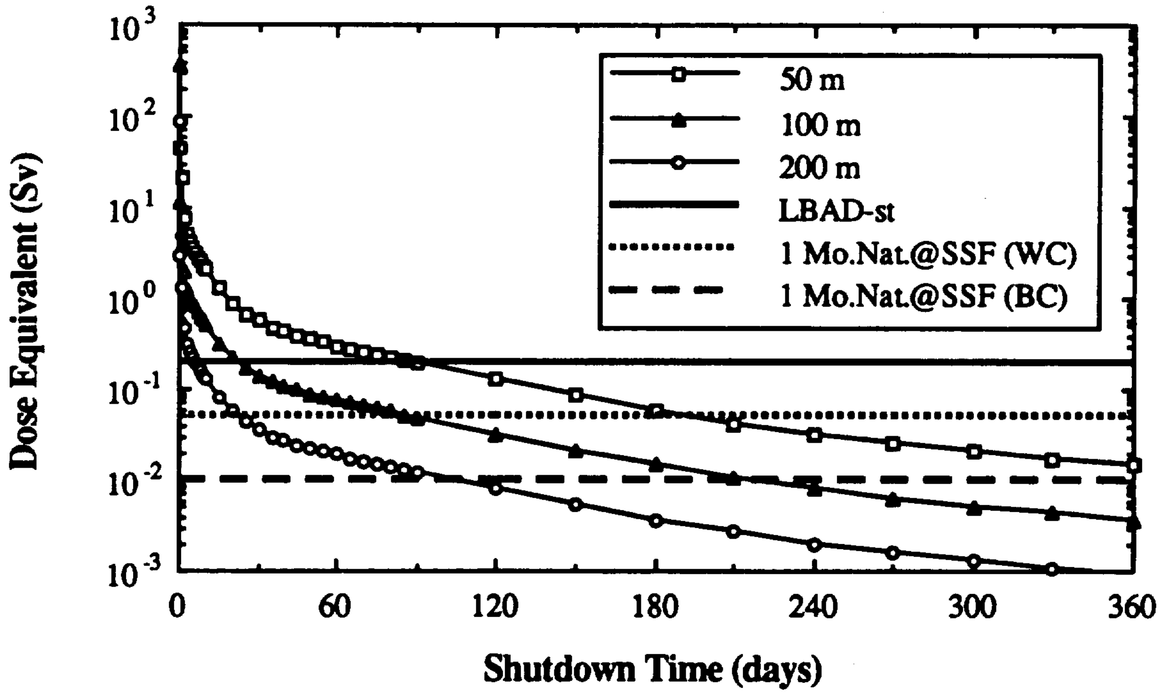
The NTR has a large buildup of fresh fission products from its recent capture burn. The NEP does not have this recent buildup of short lived radionuclides. They have mostly decayed away. But it does have a larger inventory of long lived fission products. The perspective of an EVA visitor is shown in the next figure. The NTR is a stronger source of radiation for the first ten days. Both sources weaken with time, and beyond ten days the NTR has dropped in strength to even less than the NEP.

There is no doubt that if the NTR is safe for the crew on board, it is also safe for extended visits by other astronauts so long as they stay within the shadows of the dome and disk shields

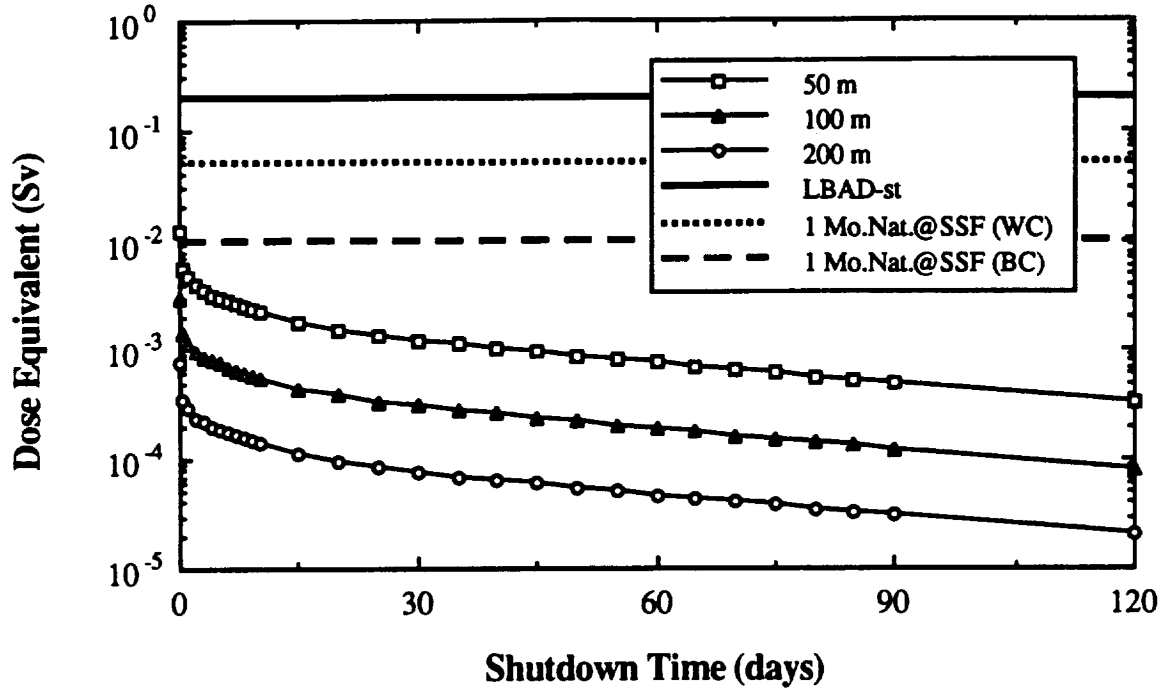
NEP & NTR Shutdown Gamma Dose Rates



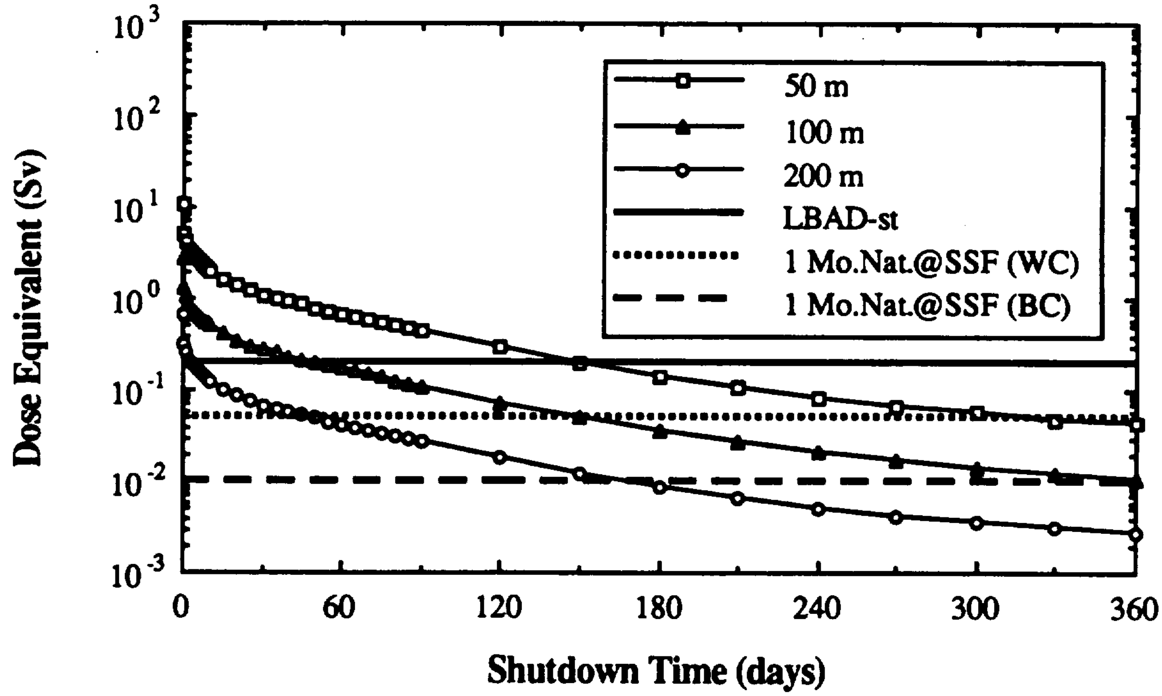
4 Hour EVA: Dose vs. Shutdown Time for Variable Separation Distance Shutdown NTR Reactor, Outside Shield, Reactor-Only Source Term



4 Hour EVA: Dose vs. Shutdown Time for Variable Separation Distance
Shutdown NEP Reactor, Inside Shadow Shield, Reactor-Only Source Term



4 Hour EVA: Dose vs. Shutdown Time for Variable Separation Distance
Shutdown NEP Reactor, Outside Shadow Shield, Reactor-Only Source Term



built into the engine. Their flyby approach should avoid the side of the reactor.

Should humans service the NTR vehicle outside of the shield shadows, they must observe cooldown times (as shown) of a few days to many months. Wait time depends on the proximity of approach and the dose they would be willing to receive.

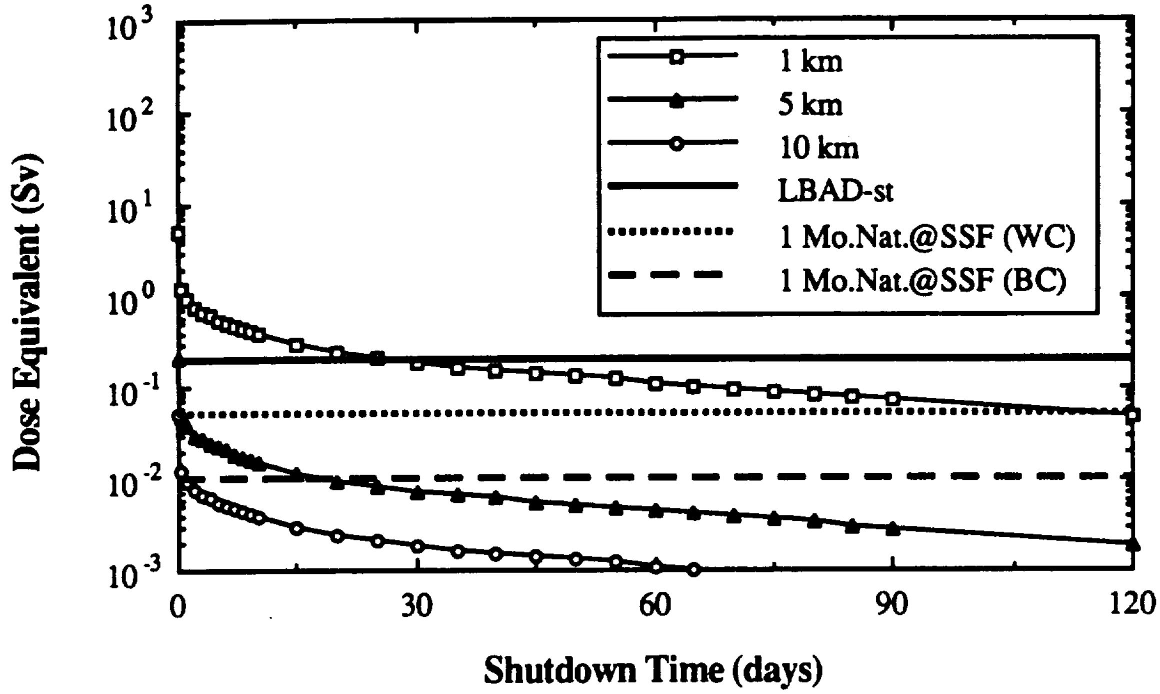
The NEP cargo vehicle gives similar results. It is not obvious that the NEP is safe to service. But, behind the shadow of the instrument shield, the results show that an EVA visit is safe at any distance over 50 m from the reactor by any dose criteria, even immediately after shutdown. Outside the shadow, however, the cooldown wait is from days to a year, depending again on the proximity and tolerable dose criteria.

If these waiting times and approach restrictions are not reasonable, a portable shield of some sort could be employed to protect service personnel in these areas. Some very preliminary estimates suggest that an aluminum shield 17.5 cm thick or a tungsten shield 2.4 cm thick would give adequate protection some ten days after shutdown.

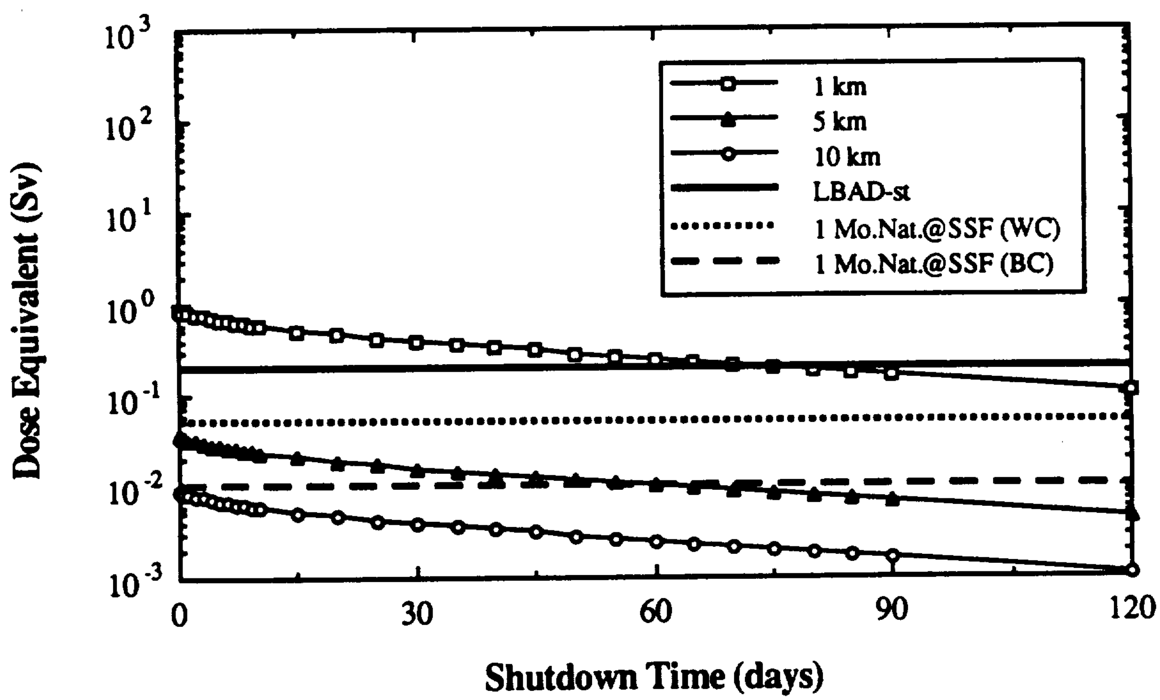
<u>Parked nuclear vehicles:</u> After an appropriate cooldown, the nuclear vehicle can be moved to a closer, more convenient parking distance relative to the inhabited site. The next six figures on the following pages give these parking trade-offs. Parking periods begin after the cooldown times indicated on the graphs. Unless otherwise stated, parking periods are taken as 30 days. All parked dose calculations are based on no incidental shielding from the vehicle structure. A brief dynamic analysis revealed that fortuitous shielding would be very unlikely.

The first two graphs show the 30 day doses. Note the hard to discern points on the y axis of the NTR graph. If the NTR is parked at 5 km immediately after shutdown it gives the LBAD-st

30 Day Parking: Dose vs. Shutdown Time for Variable Separation Distance
Shutdown NTR Reactor, Outside Shield, Reactor-Only Source Term



30 Day Parking Dose: Dose vs. Shutdown Time for Variable Separation Distance
Shutdown NEP Reactor, Outside Shadow Shield, Reactor-Only Source Term



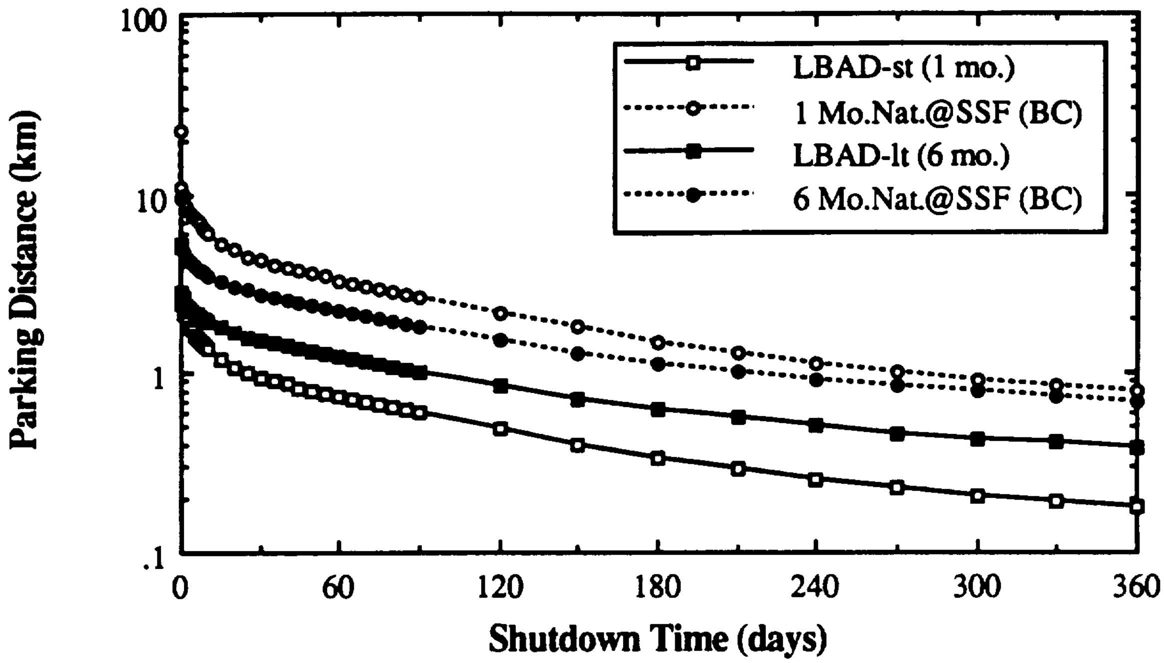
dose in 30 days. This is a result of the recent capture burn discussed earlier. In contrast, the NEP vehicle is an acceptable neighbor at this distance right away. But after only a day or so, the NTR is a more compatible neighbor. Because of the 30 day integration, the NTR and the NEP are equivalent for month long parking periods starting the second day.

The third and fourth figures show these same results in a different form. They define the parking restriction as a function of shutdown time and dose limiting criteria. Except for the first day concerns about the NTR, the radiation-related parking restrictions are quite modest. Perhaps they may be less restrictive than collision avoidance criteria.

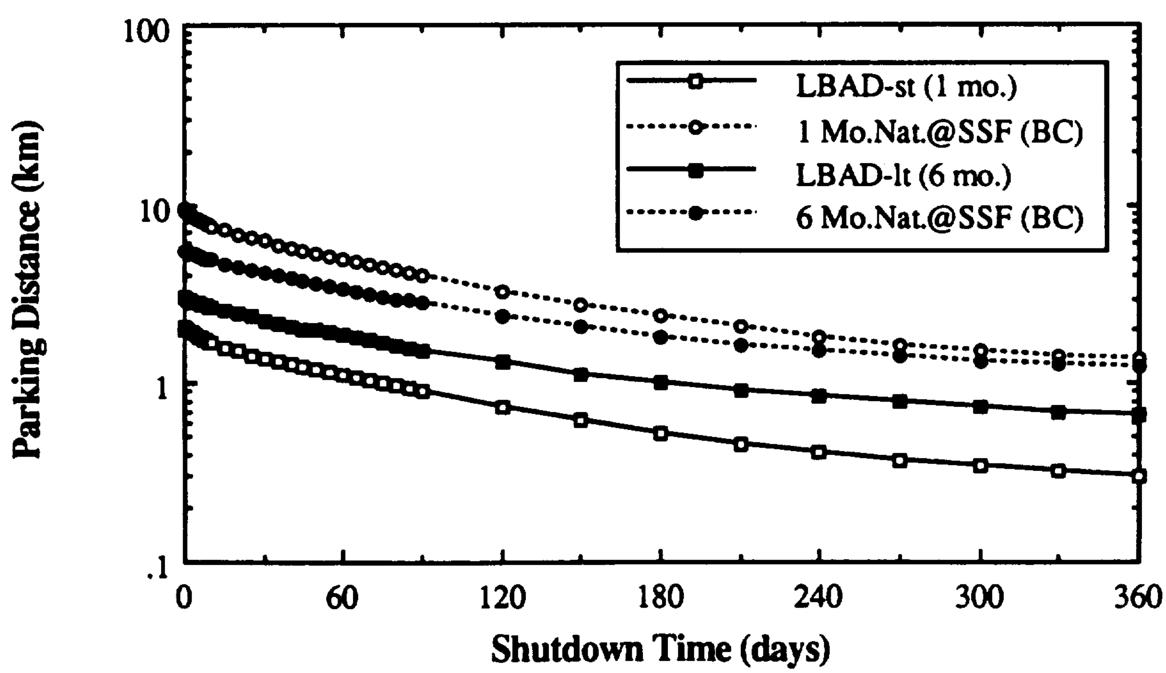
The fifth figure in the parking sequence compares distance requirements for the single and dual mode NTR's with the NEP. The fresh-burn NTR peaks on the y-axis are discernible, reminding us to keep these vehicles at a greater distance the first two days. After this the NEP needs the most space because it had accumulated 32,310 MW_{th}-days of use delivering its massive cargo. By comparison, the dual mode NTR had only 210 MW_{th}-days on its second stage engine and the single mode reactor had less. The dual mode vehicle needs a little more room than the single mode vehicle. But all three vehicles can be parked nearby quite safely. A similar trend would have been observed if a dose criterion more stringent than the LBAD had been chosen, but the actual parking distances would be larger.

The final figure extends parking duration for the NTR to six months, rather than 30 days. At 5 km or greater, radiation from the shut down reactors is never a long term issue. For the first couple days after shutdown, however, the short term recent burn issue remains. If, for some unexplained reason, we should wish to park a returned NTR vehicle for six months only 1 km away, then we must wait until it has cooled down for five months. Note that worst case long term background is never a plausible criterion, since it is worse than the LBAD-1t.

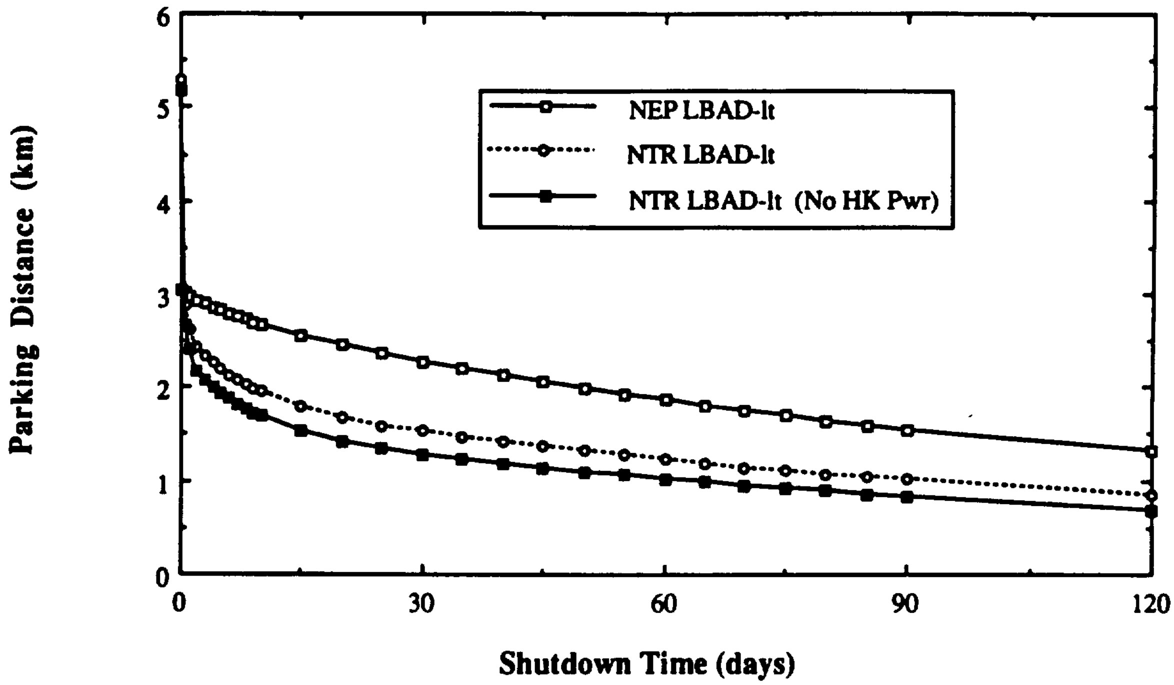
Parking Distance Required to Reduce Dose to Specified Levels
Shutdown NTR Reactor, Outside Shield, Reactor-Only Source Term



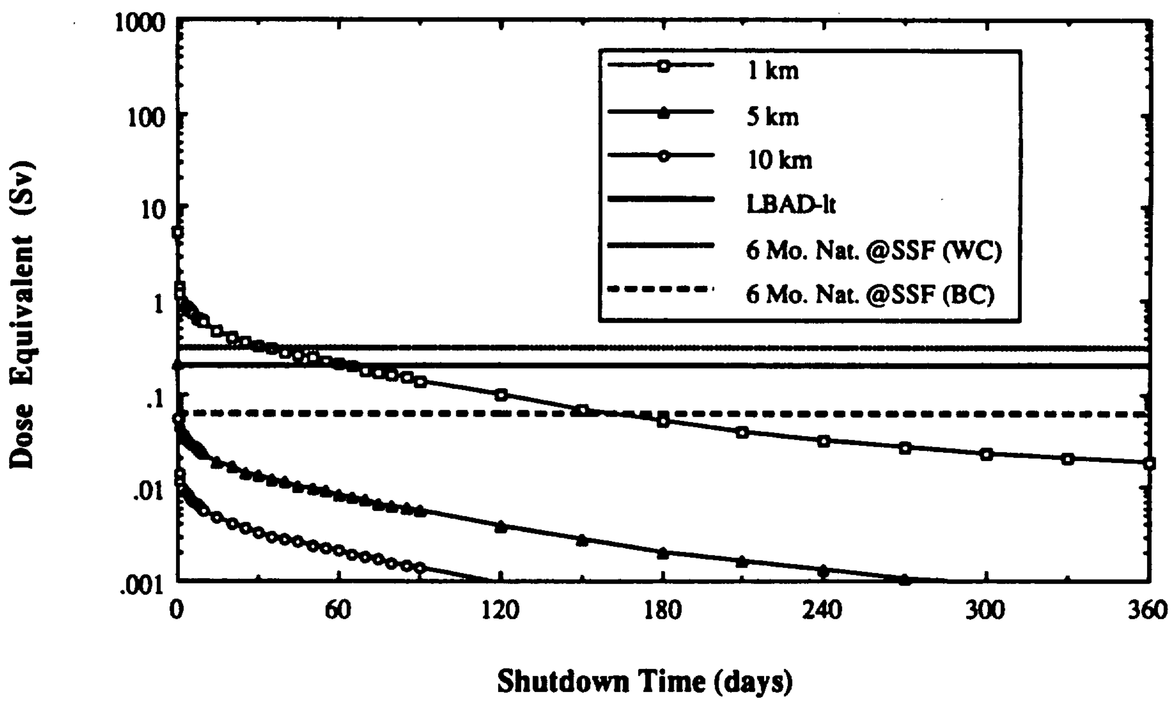
Parking Distance Required to Reduce Dose to Specified Levels
Shutdown NEP Reactor, Outside Shadow Shield, Reactor-Only Source Term



Parking Distance Required to Reduce Dose to Specified Levels
Shutdown Reactors, Outside Shadow Shields, Reactor-Only Source Terms



6 Month Parking Dose: Dose vs. Shutdown Time for Variable Separation Distance
Shutdown NTR Reactor, Outside Shadow Shield, Reactor-Only Source Term
(No Housekeeping Power during Coast Periods)



The relationship of six month to one month parking restrictions, for both vehicles, can be seen by looking back to the third and fourth figures in the set. LBAD-1t is the tighter LBAD. But in low earth orbit, background equivalent dose rates are more restrictive than LBAD's. And after long shutdown times, the one month and six month background criteria are virtually the same.

Let's summarize our results concerning returned nuclear vehicles:

They can be safely serviced by humans at any time within the shadow of their built-in shields. Outside these shadows, however, some supplemental protection is needed. Close in, even long cooldown times do not help enough. This shielding may be an integral part of the vehicle, but more likely it will include some portable shielding stored at the arrival node.

Parking restrictions are also readily livable. It is indeed reasonably achievable to use a very tight criterion to fulfill the ALARA principle for parked nuclear vehicles. Using the best case background dose at space station (1 rem/mo), we can define the following minimum parking distances:

Parking an NTR vehicle can be thought of in terms of three bench marks. Within two days of any propulsive maneuver, the vehicle should be parked at least 20 km away from human habitation (unless vehicle shielding is added). After two days it can be parked at 10 km. Or, after 15 days it can be parked at 5 km.

Parking an NEP vehicle can be thought of in terms of two bench marks. At any time after return it can be parked 10 km away. After 50 days cooldown it can be parked 5 km away. Closer distances are acceptable if shielding is provided.

If it is desired to hard dock previously operated nuclear vehicles at a facility housing people, then shielding of the reactor is essential. It should be quite reasonable to do.

Systems - radiation trades: By examining exploration systems in the context of natural radiation hazards we can identify major risks and possibilities for synergistic shielding benefits. On the next page Table 6, Systems - Radiation Trade Space Risk Analysis, correlates the most pertinent natural radiation risks with two cases each of platforms, non-nuclear vehicles and nuclear vehicles. The shading summarizes the degrees of risk. By understanding these interplays we can understand the man-made radiation sources in the total space exploration context.

Solar particle events (SPE) and GCR are hazards to any system outside the earth's magnetic shield, i.e. all exploration systems except low earth orbit (LEO) platforms at low or mid latitudes. This same magnetic field which protects LEO, also traps and energizes charged particles, creating the Van Allen belts. The lower belt in particular (the proton belt) is lethal to any human system which does not exit it quickly enough. Two very different types of vehicles are subject to such disaster.

How much shielding is advisable to protect against GCR and SPE? The graphs on the following page from NASA Langley (reference 1) address that issue. The GCR are highly penetrating and at least 5 g/cm^2 is needed to keep them under the NCRP annual limit. This is the specified, or "spec," spacecraft in the 1989 OEXP studies. The SPE are intense enough to be lethal if this is the only protection. So storm shelters of 20 g/cm^2 were specified to reduce the SPE below the NCRP monthly limit. Clearly, even more shielding would lead to healthier astronauts. Also, since the uncertainty in the GCR and its effects is a factor of two, the uncertainty in the necessary shielding is a factor of ten!

NEP vehicles in Earth orbit: What happens when people ride a low thrust NEP vehicle in an Earth spiral orbit? They die, not from reactor radiation, but from naturally trapped radiation. The Air Force Weapons Laboratory (AFWL) quantified some cases (reference 12) with expected results. Table 7, three pages ahead, shows the best case which still had a bad outcome.

SYSTEM - RADIATION TRADE SPACE
Risk Analysis

Manned System \ Natural Radiation Source	Solar Particle Events	Trapped Radiation	Galactic Cosmic Rays
LEO Space Station/Platforms	Shielded by magnetic field except for polar orbits	Significant protons below 10,000 km	Shielded by magnetic field except for polar orbits
Space Platforms Beyond HEO		Significant electrons below 30,000 km only	
Chemical Interplanetary Personnel Vehicle		Penetrates belts quickly	
Earth Aerobraked Interplanetary Personnel Vehicle. (Non-Nuclear)		High radiation risk for non-ideal Earth capture	
Nuclear Electric Personnel Vehicle		LETHAL FOR PRACTICAL CASES	
Nuclear Thermal Personnel Vehicle		Penetrates belts quickly	

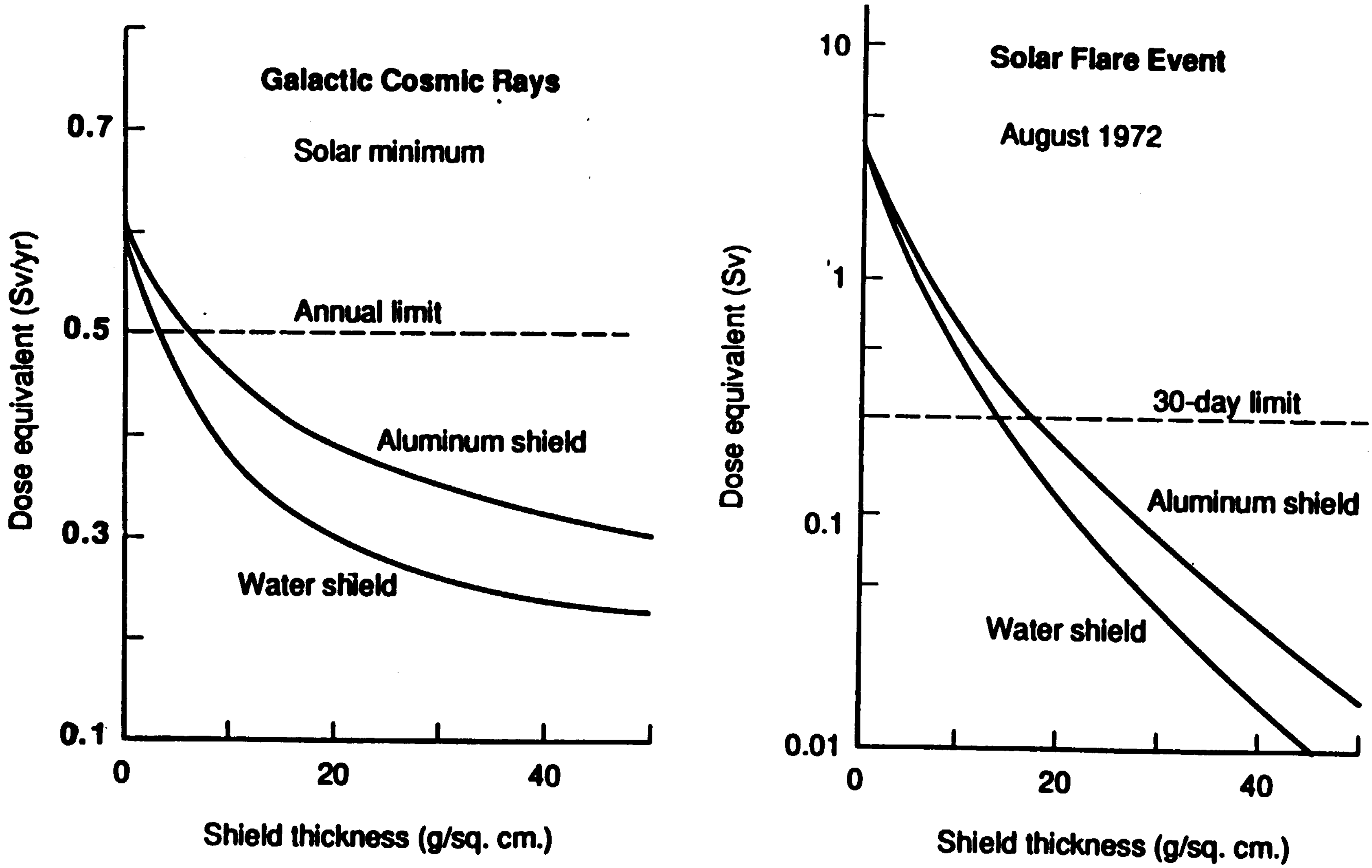
Significant Radiation Hazards

Potentially Lethal Radiation Hazards

2.36.071089.AJW.TMV.06

Desirable Shielding from Natural Radiation

Dose Equivalents to blood forming organs



Source: NASA Langley, Dr. Lawrence Townsend, July 1989

Doses Absorbed During "Fast" Spiral Escape from Earth

"Fast" Sprial conditions:	10 MWe, Crew capacity = 5
Total mass:	323 MT
Thrust	326 N
Start orbit	450 Km, i=28°, e=0
Finish orbit	70,000 Km, i=28°, e=0
Duration of spiral	58 days
(Time to escape)	(75 days)

	Peak Period	Thin S/C 2 gm cm^{-2}	"Spec" S/C 5 gm cm^{-2}	Shielded S/C 10 g cm^{-2}
Protons	Days 0-15	> 800 rads	> 400 rads	150 rads
Electrons	Days 15-60	~ 60 rads	~ 20 rads	few rads

Two NEP personnel vehicles were synthesized for this study. The "fast" one tabulated here was powered by a 10 MW_e reactor. Three shields were examined: a typical thin space module, the OEXP "spec" Mars module, and a water jacket shield at twice "spec" thickness. This was not the former NEP cargo vehicle, but was as fast a NEP as we could reasonably define. It still took 75 days to escape Earth. But the lethal dose was delivered in the first 15 days by the proton belt. The doses are shown in rads. A "Q-factor" is needed to convert them to rems. For protons, Q can vary from 1 to 14. Subsequent calculations have shown that Q = 3 is best for this situation. Our lowest dose converts to 450 rems, which is lethal.

There are two ways to improve the spiral NEP ride. An even thicker shield could be considered. However, our confidence in the dose calculations gets shakier for very thick shields. Quadruple the thick shield might be needed to barely survive. The other option is to take a high thrust hop over the proton belt and then spiral. This would work, since people can survive the trip through the electron belt behind the thick shield. But this scheme would still require the crew to fritter away 60 days extra in Earth orbit for no good reason. No practical case of riding the NEP in Earth spiral is envisioned, except possibly a desperation return from high Earth orbit down to the upper edge of the proton belt if no other transport were available.

<u>Radiation exposure in aerobraked vehicles:</u> The other vehicle which may have problems with the trapped belts is an aerobraked vehicle returning to Earth. This is an ironic contrast, since the aerobrake vehicle with its high g's and nonpropulsive mode is the antithesis of the NEP with its minute g forces and great propulsive capacity. With help from the AFWL again (reference 13) we investigated this complex situation too.

An aerobraking vehicle must come within about a 100 km of the Earth's surface to hit the right density atmosphere to brake or turn. If in one pass it can go directly to rendezvous with the

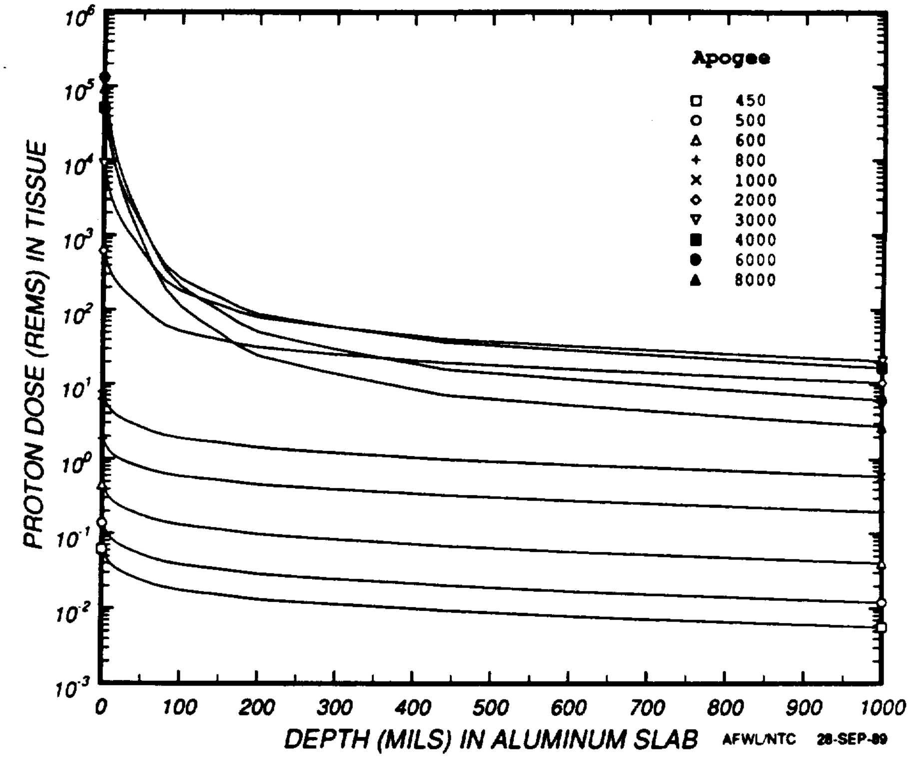
space station, it will have stayed below the proton belt and avoided radiation. But many other situations are likely. The three dimensional geometry of the trans Earth trajectory plane intersecting with the final capture plane can involve a variety of angles and inclinations. Rendezvous with the station itself can involve phase changes. Together, these factors can dictate multiple passes and an assortment of intermediate orbits and holding delays, which may place astronauts in the trapped belts for unacceptable times. Further, g-force limitations or braking errors might result in orbits penetrating the trapped belts.

Our calculations were not intended to sort out these complex possibilities, but to calculate the radiation doses across the range of possibilities. The results shown in the two graphs on the next page are for braking orbits which keep perigee at 100 km, but enter the proton belt. Q factors as a function of energy were incorporated directly into the calculations. Doses are given for one day as a function of shield thickness, with braking orbit apogees and inclinations as parameters.

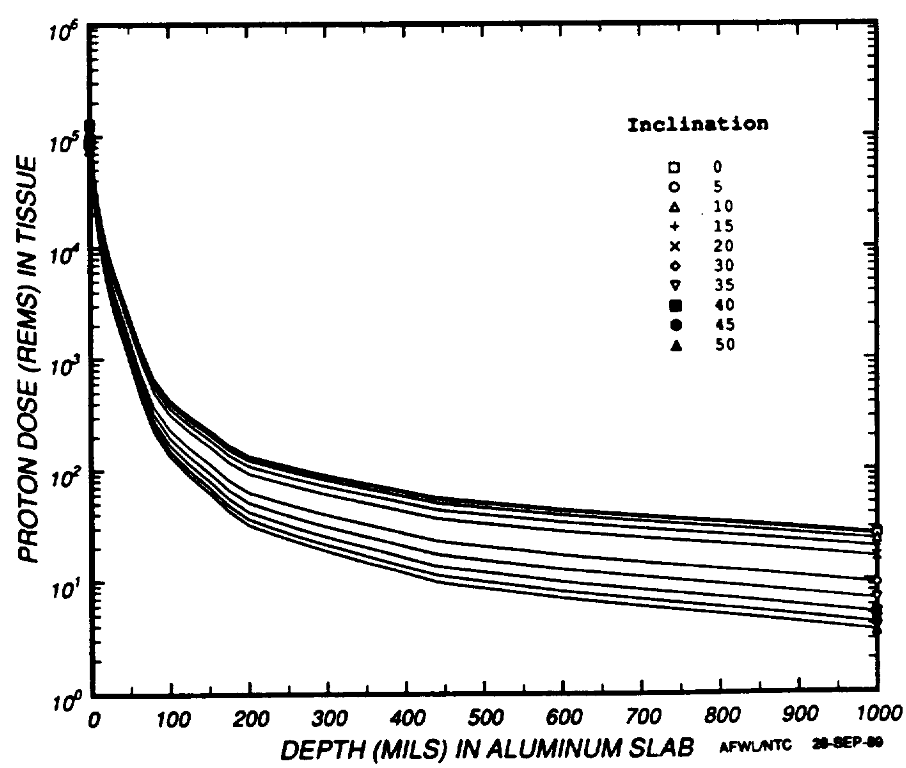
The worst doses occur at apogees of a few thousand kilometers and at low inclinations. The "spec" shield corresponds to 729 mils, and the thin skinned module to 291 mils. By design intent, aerobraked modules are as light as possible. At the SSF inclination of 28.5 degrees, the maximum daily dose is 30 rem behind the "spec" shield if apogee goes to 3000 km. It would be 60 rem behind the thin skin. Apogees must be kept below 1000 km to avoid significant dose. At zero degrees inclination and 5000 km apogee, 100 rem is received behind the thin skin or 40 rem behind the "spec" shield. The latter drops to 25 rem at 20 degrees and 15 rem at 30 degrees.

Such doses would be very serious for returning Mars astronauts who have had substantial radiation before this nasty "welcome home." If aerobraking, with its g stresses and vibration, is still to be retained, either capture below 1000 km or exiting the braking orbits far sooner than a day would be a necessity.

Doses (Rems) from solar min protons in tissue with quality factors
Braking orbit Perigee: 100 km Inclination: 28.5 deg
PROTON DOSES IN A SEMI-INFINITE ALUMINUM SLAB
TIME AFTER SCENARIO START: 1.000 DAYS



Doses (Rems) from solar min protons in tissue with quality factors
Braking orbit Perigee: 100 km Apogee: 5000km
PROTON DOSES IN A SEMI-INFINITE ALUMINUM SLAB
TIME AFTER SCENARIO START: 1.000 DAYS



Synergistic shielding opportunities: Table 8 on the next page, System - Radiation Trade Space Opportunity Analysis, summarizes the synergistic shielding possible when we consider reactors and natural radiation together. Applications of both multifunctional and expendable shields are identified.

In LEO, the portable shields we defined for protecting personnel while servicing nuclear vehicles or for shielding docked nuclear vehicles can be put to good long term use keeping the trapped protons off our people. This makes them "multifunctional" shields. On high platforms we shield the SPE and GCR, instead of the trapped particles.

On board vehicles, a multifunctional shield is again a permanent shield that protects people from both reactors and from natural radiation. Implied is a shift of our shield mass from reactors to crew compartments where it may be more productive. The shielding mass budget can also be more generous on nuclear vehicles than on chemical vehicles because the higher specific impulse of nuclear vehicles gives better payload fractions.

An expendable shield could be made of propellant or payload. Propellant is attractive in that it could be burned off as we enter the martian or lunar gravity wells, when it is most burdensome and least needed. This idea could be used for single vehicles, or for vehicle combinations which leave reusable interplanetary transports in high orbits. Examples are: shields of payload which we use at destination (particularly water), Phobos or Deimos regolith shields, or burnable shields of our destination propellant. NTR vehicles would work well with propellant shields since they are so propellant versatile.

Prudently used, nuclear systems can potentially yield a "net dose avoidance". This dose avoidance can deliver us from the dilemma we would be in if LBAD's in deep space would go to zero. Nuclear propulsion can indeed be part of the radiation solution, rather than part of the radiation problem.

SYSTEM - RADIATION TRADE SPACE
Opportunity Analysis

Manned System / Natural Radiation Source	Solar Particle Events	Trapped Radiation	Galactic Cosmic Rays
LEO Space Station/Platforms	Not relevant for non-polar orbits	Multifunctional shielding	Not relevant for non-polar orbits
Space Platforms Beyond HEO	Multifunctional shielding	Not relevant	Multifunctional shielding
Chemical Interplanetary Personnel Vehicle	Expendable shielding	Not relevant	Expendable shielding
Earth Aerobraked Interplanetary Personnel Vehicle. (Non-Nuclear)	No expendable mass on return flight	No expendable mass during aerobraking	No expendable mass on return flight
Nuclear Electric Personnel Vehicle	Expendable or multi-functional shielding	LETHAL, SHIELDS INADEQUATE	Expendable or multi-functional shielding
Nuclear Thermal Personnel Vehicle	Expendable or multi-functional shielding	Not relevant	Expendable or multi-functional shielding

"Expendable" shield material is propellant or payload, e.g., water.
"Multifunctional" shield addresses reactor and natural radiations.

Opportunity for Synergistic Shielding

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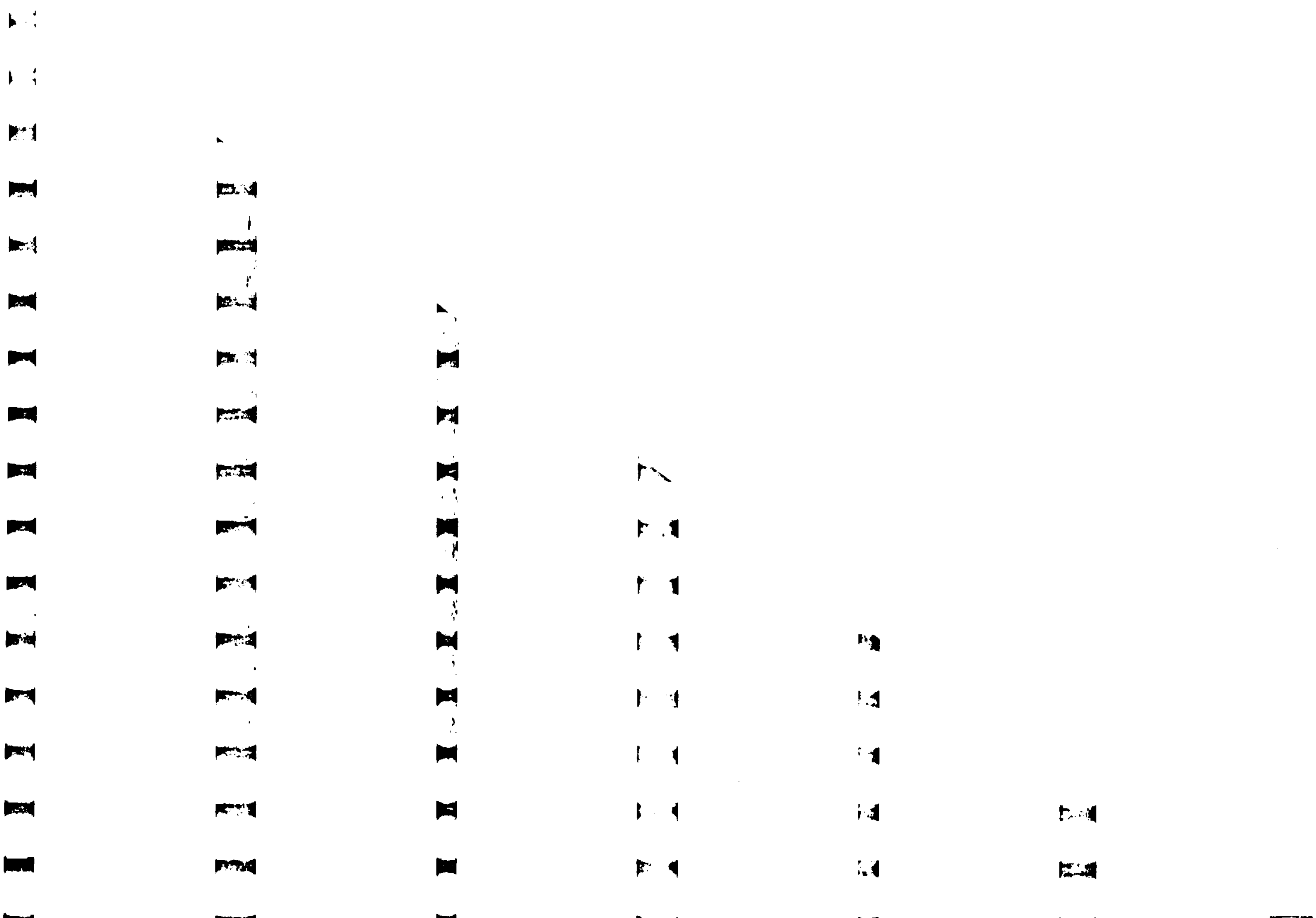

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Performance Comparisons of Nuclear Thermal Rocket and Chemical Propulsion Systems for Piloted Missions to Phobos/Mars

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PERFORMANCE COMPARISONS OF NUCLEAR THERMAL ROCKET AND CHEMICAL PROPULSION SYSTEMS FOR PILOTED MISSIONS TO PHOBOS/MARS

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Abstract

Performance capability of nuclear thermal rocket (NTR) and chemical propulsion systems, operating with and without aerobraking, are compared for a selected set of Mars mission opportunities in the 2000 to 2020 timeframe. Both high- and low-energy mission opportunities are investigated. Results are presented as the required initial mass in low Earth orbit (IMLEO) to perform the missions. Missions exclusively using chemical propulsion systems have the greatest initial masses. Significant mass reductions are realized by utilizing either aerobrake or NTR technology or both. As mission energy requirements increase, the benefit of implementing aerobrake or NTR technology increases, resulting in IMLEO mass reductions on the order of 60 to 75 percent when compared with all-propulsive chemical missions. By combining both advanced technologies, still greater mass reductions are possible. The effect on the propulsion system comparison and the IMLEO, of such factors as trajectory type, launch opportunity, Mars capture orbit, mission flight mode, ΔV optimization, aerobrake mass, vehicle recovery, and shortened trip time is also presented.

Introduction

The National Aeronautics and Space Administration (NASA) has considered chemical, nuclear thermal rocket (NTR), and aerobrake propulsion systems for piloted Mars missions as far back as the early 1960's.[1-3] Both chemical and NTR technologies underwent extensive testing during the 1960's. The Apollo lunar landing program provided a catalyst for the development of large liquid oxygen/liquid hydrogen (LOX/LH_2) fueled engines used to power the upper stages of the Saturn V rocket. Similarly, a joint NASA/Atomic Energy Commission program was initiated in 1960 to develop a Nuclear Engine for Rocket Vehicle Application (NERVA).[4] Building on the technology base provided by the Rover nuclear rocket program[5] begun in 1955, the NERVA program demonstrated reusable, high thrust, high specific impulse, NTR systems. The Rover/NERVA programs were terminated in 1973, short of flight demonstration, because of the decision to delay indefinitely a piloted Mars mission, for which the NTR was primarily being developed.

Today, NASA is again focusing on the human exploration of Mars[6] in response to the United States National Space Policy's charge to expand human presence into the solar system. The Office of Exploration (OEXP) at NASA Headquarters and the NASA field centers have been conducting studies aimed at determining the technology and infrastructure needed to support future Mars and lunar initiatives.

Both earlier studies[7] and more recent ones[8,9] by NASA Lewis Research Center's (Lewis) Advanced Space Analysis Office (ASAO), NASA Marshall Space Flight Center (MSFC), and others indicate that aerobraked chemical and all-propulsive NTR systems each have the potential to reduce initial mass in low earth orbit (IMLEO) for typical Mars missions by approximately 50 percent or more compared with an all-propulsive chemical mission. Because of the different ground rules and assumptions made by the various investigators, drawing overall conclusions regarding relative system performance is difficult. To help clarify the performance issue, OEXP asked ASAO to conduct a consistent performance comparison study of NTR and chemical propulsion systems. Since results were to be compared with other studies being performed for OEXP, ground rules for the Lewis study were coordinated with other NASA field centers and contractor organizations supporting OEXP.

The principle figure-of-merit used for comparing the propulsion systems is the required IMLEO. The IMLEO has been used frequently in past studies and continues to be of interest owing to its association with program costs and mission complexity.

Descriptions of propulsion systems characteristics and the vehicle configurations used are outlined in the following section. Then, the analytic approach of the study is discussed. Next, results are presented for the two mission scenarios that were investigated, expeditionary and evolutionary exploration of Mars. Finally, a summary of the technical

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results and conclusions which have been reached is presented.

Propulsion System Characteristics and Vehicle Configurations

Performance and mass data for propulsion and aerobrake technologies were obtained from numerous sources, including various NASA field centers, government laboratories, and industry. Of the concepts under consideration in this study, chemical propulsion is the most highly developed, with a variety of engine designs available in the thrust ranges of interest to mission planners. For the high-energy, large-mass propulsion phases, high-thrust engines are desirable in order to minimize the performance losses associated with a finite burn duration (discussed in the next section). An improved-performance, space-based derivative of the current high-thrust space shuttle main engine (SSME) was utilized mainly for Earth-departure primary propulsion although its use for both Mars capture and departure maneuvers was also examined.[8,10] Clusters of lower-thrust engines, derived from RL-10 technology,[8,10] were used for Mars departure with lower-mass vehicles such as the chemical/aerobrake vehicles systems discussed below.

The NTR is next in terms of technology maturity. Between 1955 and 1972, a total of twenty reactors were designed, built, and tested.[5,11] Thrust levels of 55, 75, and 250 klbf were achieved at corresponding reactor power levels of 1100, 1500, and 5000 thermal megawatts. Engine restart and sustained burn capability (greater than one hour) was also demonstrated. During the 1960's, NTR testing was conducted at the Nuclear Rocket Development Station[5] at Nevada's Nuclear Test Site. Here, the hydrogen propellant was exhausted directly into the atmosphere in an open cycle mode. Today, closed cycle tests would be necessary to meet current environmental and safety requirements.

Three sets of NTR performance levels were investigated. The first, to be referred to as "'72 NTR" in the text, represents 1972-vintage NERVA and Phoebus-2A nuclear rocket performance capability with graphite matrix/composite reactor fuel elements[5,12] providing an engine specific impulse of 900 seconds. The second performance level, referred to as "'89 NTR", represents the performance of similarly sized engines built with state-of-the-art materials and propulsion system components (pressure vessel, turbopump, nozzle, etc.) that increase the engine thrust-to-weight ratio beyond the 1972 technology levels. The third performance level, referred to as "advanced NTR", assumes that the '89 NTR materials and propulsion systems along with advanced high-temperature uranium-zirconium carbide reactor fuel elements[5,12] will cause both the engine thrust-to-weight ratio and the specific impulse to exceed the 1972 technology levels. A propulsion system using both the '72 NTR performance level engines and aerobraking was also investigated and is referred to as NTR/AB. Radiation shield weights were obtained from NASA contractor studies[13] of NTR stages (conducted during the 1960's and early 1970's) for lunar and interplanetary applications. Estimates of performance penalties due to postburn reactor cooldown are also included.

In contrast to the chemical and NTR systems, well-defined mass and sizing assumptions for the aerobrake systems were more difficult to obtain. While the Apollo and space shuttle programs have provided data on Earth descent aerobraking, no data base exists that validates orbital capture aerobraking.[14] Establishing this capability will require development efforts in the areas of lightweight, high-temperature thermal protection systems, autonomous guidance, navigation and control systems, and aerobrake configurational design. The capability for in-space assembly of large, spacecraft-compatible, aerobrake structures must also be developed. In regards to testing, it is prudent to assume that a significant scale flight test would be required to validate performance under conditions expected during Mars and/or Earth entry.

Due to uncertainties which exist in the aerobrake technology and design areas, mass estimation is difficult. In the Lewis study, the aerobrake mass is calculated to be a given percentage of the payload to be braked (i.e., mass of the entire vehicle less aerobrake). This percentage is based on the aerobrake lift-to-drag ratio (L/D) and typically increases in going from low L/D (0.2) conical shields, to medium L/D (1.0) biconic shields, to high L/D (2.5) winged vehicles.[14] Because of OEXP's current interest in the blunt conical and biconic concepts, these two configurations were the primary focus of the Lewis analysis. The Lewis study also assumed that the aerobrake L/D ratio and corresponding mass fraction decreased for the later missions due to advancements in aerobrake technology.

The principle propulsion system and aerobrake mass assumptions made in this study are summarized in Table 1. Propellant tank mass is calculated as a percentage of the total propellant load required for a particular maneuver. Chemical propulsion system tank masses range from 7 to 15

Table 1 - Principle Propulsion System and Aerobrake Sizing Assumptions

a) Engine systems

Engine Systems:	Propellant	Isp, s	Thrust, klbf	Engine Mass, t	Shield Mass, t
SSME Derivative	LOX/LH2	480	532	3.6	N/A
RL-10 Derivative	LOX/LH2	471	20	0.2	N/A
'72/'89 NTR	LH2	900	75	11.3/5.5	4.5
'72/'89 NTR	LH2	900	250	19.0/15.8	9.0
Advanced NTR	LH2	1000	75	5.5	4.5
Advanced NTR	LH2	1000	250	15.0	9.0
Auxiliary Chem	Storable Bipropellant	310-316	Low	Low	N/A

b) Aerobrake mass for various vehicles
[Calculated as a percentage of braked payload mass]

Vehicle	Relative L/D Ratio	Aerobrake Mass, %
Expeditionary Cargo	Medium	17.5
Expeditionary Piloted	Medium	20.0
Evolutionary (Piloted)	Low	13.2

percent of the required propellant load. The NTR systems have tank masses ranging from 14 to 20 percent of the propellant load. The NTR tankage fractions are assumed to be greater than the LOX/ LH_2 fueled chemical systems because of the lower density of the NTR LH_2 propellant.

Expendable and recoverable vehicles were considered in the Lewis study. Figure 1 provides a representative sampling of expendable, direct entry configurations showing the relative placements of crew modules and the main and auxiliary propulsion systems. To obtain a near optimal thrust-to-weight ratio for the spacecraft, separate stages are traditionally used for the main propulsion phases: a trans-Mars injection stage (TMIS) for Earth departure, a Mars orbital capture stage (MOCS), a trans-Earth injection stage (TEIS) for Mars departure, and an Earth orbital capture stage (EOCS). An alternative approach is to use a common engine system with staged propellant tank sets to achieve comparable results. For the all-propulsive vehicles (shown in Figs. 1a and b), both approaches are utilized. The all-propulsive chemical vehicle uses three SSME derivative engines and associated tankage for the TMIS. A single SSME is then used for the remaining MOC and TEI burns with the spent MOC propellant tanks being jettisoned after capture into Mars orbit. Separate MOC, TEI, and EOC stages and additional engines are utilized for the recoverable configuration to optimize performance. The NTR system uses a single 250-klbf-thrust (Phoebus class) engine plus tankage for the TMIS. To minimize vehicle mass, the remainder of the mission employs a single 75-klbf-thrust (NERVA class) engine with staged MOC tanks. Cooldown propellant is provided to the NTR following MOC to prevent structural damage to the reactor. The NERVA-powered core spacecraft shown in Fig. 1b can also be recovered if additional propellant and tankage is provided for EOC and TEI/EOC cooldown.

The aerobraked chemical and NTR vehicle configurations are depicted in Figs. 1c and d. The chemical TMI and TEI stages use a single SSME and six RL-10 derivative engines, respectively, while the NTR system uses the same Phoebus and NERVA-class engines. For the recoverable configurations, a common MOC/EOC aerobrake is assumed. All vehicles include additional small auxiliary bipropellant stages for midcourse correction

(MCC) and reaction control system (RCS) orbit maneuvering. The crew live in interplanetary mission modules (IMM) for the majority of the flight phases. An Earth crew capture vehicle (ECCV) provides for capsule reentry at Earth.

Analytic Approach

To provide an accurate assessment of the IMLEO requirements, many ground rules and modeling assumptions were required for the analysis. Much of this information was obtained from OEXP documents and concurrent studies,[10,15] including details on mission profiles, orbit altitudes, injection energies, trip times, tankage fractions, boiloff rates, propellant margins, aerobrake masses and cargo and payload masses (e.g., IMM, ECCV, Mars ascent-descent vehicles, and science payloads).

To complement the various trade studies on aerobraked chemical propulsion systems undertaken for OEXP, the ASAO examined a representative sampling of expeditionary and evolutionary mission opportunities. The expeditionary missions assumed a split cargo-piloted sprint scheme[7,16] (termed "split-sprint") in which an unmanned cargo vehicle (carrying all hardware to be delivered to Mars and, in some cases, the fuel for TEI) is launched on a slow, one-way, relatively low-energy conjunction-class trajectory to Mars. To limit the time the crew is exposed to the space environment, the piloted vehicle is launched on a higher energy, opposition-class sprint trajectory. In the evolutionary missions, opposition- and conjunction-class trajectories will be used to establish and support a permanently inhabited base. Short duration, high energy trajectories were also studied as a way to reduce the interplanetary travel time and, thus, increase mission time at Mars.

Table 2 details the velocity change requirements, referred to as the delta-V's (ΔV's), for typical missions. For the main propulsion phases, ΔV's were calculated by first determining the interplanetary trajectory energy requirements and then finding the velocity change required to depart from or arrive into a particular orbit about Earth or Mars. In addition to the main propulsion phases, MCC and RCS ΔV's were included to simulate inflight maneuvers. Also, the effect of finite burn propulsion losses was considered to account for performance penalties associated with low thrust-to-

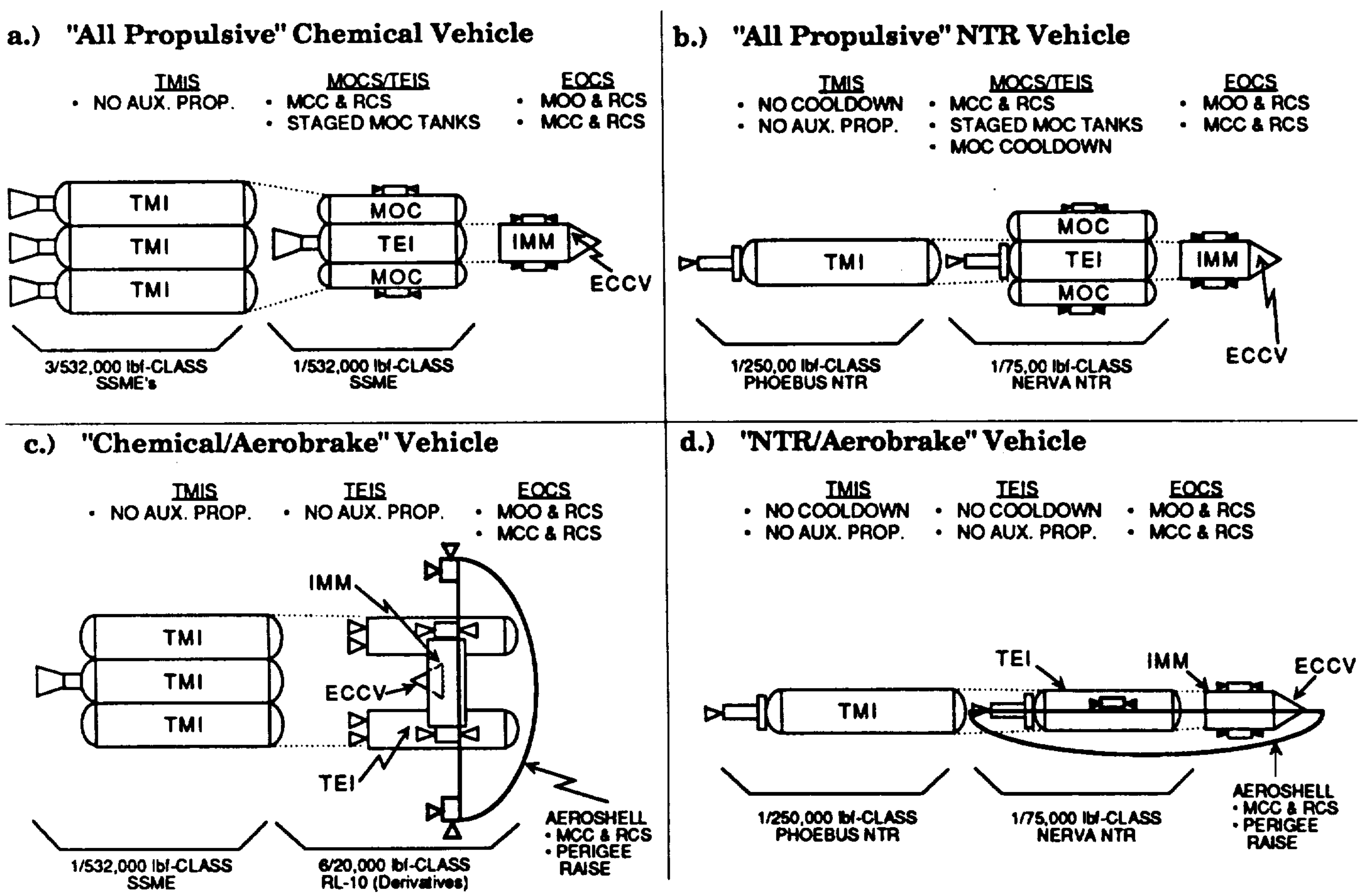

Figure 1 - Baseline Expendable Piloted Vehicle Configurations

Table 2 - Nominal Velocity Requirements

- All missions depart Earth from 500-km-altitude circular orbits
- Arrival/departure orbit altitudes from Mars:
  Expeditionary: Piloted and Cargo - 500 km circular
  Evolutionary: Mission 1 - 250 by 18000 km
  Mission 5 - 6000 by 6000 km
- Injection delta-V's (TMI, MOC, TEI, and EOC) optimized for aerobraked missions (large velocity changes at Mars/Earth capture).

Manuever	Propulsion System	Velocity Change (m/s) 2002 Expeditionary Piloted	Velocity Change (m/s) 2002 Expeditionary Cargo	Velocity Change (m/s) Evolutionary Mission 1	Velocity Change (m/s) Evolutionary Mission 5
Trans-Mars Injection	Main Impulse	4188	3512	4057	3670
Spin-Up/Spin-Down	Auxiliary	--	--	31	31
Mid-Course Correction	Auxiliary	50	50	50	50
Reaction Control	Auxiliary	50	50	50	50
Mars Capture:					
All-Propulsive	Main Impulse	4566	3410	4528	3819
Post-AB Perigee Raise	Auxiliary	86	86	12	559
Mars Operations - 1	Auxiliary	20	100	100	100
Mars Operations - 2	Auxiliary	--	--	50	--
Reaction Control	Auxiliary	50	50	50	50
Trans-Earth Injection	Main Impulse	3854	End Mission	1773	1726
Spin-Up/Spin-Down	Auxiliary	--		31	31
Mid-Course Correction	Auxiliary	50		50	50
Reaction Control	Auxiliary	50		50	50
Earth Capture:					
All-Propulsive	Main Impulse	N/A: Direct		3680	4240
Post-AB Perigee Raise	Auxiliary	Entry		108	108

weight ratio burns.[17] Finite burn losses were included for the TMI, MOC, and TEI burns. The EOC finite burn losses were not included as it was assumed that all systems would have relatively high thrust-to-weight ratios upon Earth arrival.

To determine the IMLEO requirements of the various propulsion systems for the particular missions under consideration, a vehicle sizing computer code was developed. Using the specified missions, payloads, and stage and propulsion system scaling assumptions, the computer code determined propellant loadings and vehicle masses for various piloted and unpiloted spacecraft.

Expeditionary Mars Exploration Study

As mentioned earlier, the expeditionary analysis was based on a split-sprint mission profile. The Mars cargo vehicle (MCV) will carry the Mars ascent-descent vehicle (MADV) on a one-way, conjunction-class trajectory, taking 200 to 400 days. The Mars piloted vehicle (MPV) will leave 12 to 24 months later on an opposition-class trajectory (travel time, 150 to 300 days) to transport the crew, TEIS and science payloads to Mars. Either aerobraking or propulsive braking will provide Mars orbital capture of both the MCV and MPV. The crew will transfer to the MADV in which they will descend to and, following a 20-day stay, ascend from the Martian surface. After completing a 30-day stay in Mars orbit, the TEIS will provide the Mars-departure propulsion for the return to Earth. The mission ends 160 to 300 days later with an Apollo-type capsule reentry at Earth.

For the reference expeditionary mission, IMLEO requirements were determined for chemical and NTR propulsion systems operating all propulsively (AP) and with aerobrakes (AB). Several trade studies were also performed. The effect of the Mars parking orbit apogee altitude on the propulsion system comparison was addressed. The effect of the launch opportunity was assessed by examining IMLEO requirements in the 2000 to 2010 timeframe. Finally, the effect of flight mode (split-sprint and single-vehicle missions) on the propulsion system comparison was also investigated.

Reference Mission Results

The 2002 mission opportunity is the reference mission used for the propulsion system comparison (Table 3). The baseline Chem/AB system studied by Lewis has a combined total mass of 686 t (t = 1000 kg) for the piloted and cargo vehicles. By comparison, the all-propulsive chemical system (Chem/AP) is 2.5 times more massive, having an IMLEO of 1782 t. The all-propulsive '72 NTR has an IMLEO of 675 t, which is approximately 98 percent of the baseline Chem/AB system. The '89 NTR and advanced NTR systems have IMLEO's of 87 and 72 percent of the baseline Chem/AB system, respectively. The NTR/AB system is the lightest of all the propulsion options examined, having an IMLEO that is approximately 60 percent of the Chem/AB baseline. For the trade studies discussed herein, only results for the '72 NTR and advanced NTR are presented for the all-propulsive NTR systems, because they are expected to bracket performance characteristics for the NTR.

Table 3 - Expeditionary Mars Exploration, Propulsion System Comparison

Propulsion System	IMLEO, t		Total IMLEO, t	% of Chem/AB IMLEO
	Piloted	Cargo		
Chem/AB	547	139	686	100
Chem/AP	1494	288	1782	260
'72 NTR	499	176	675	98
'89 NTR	439	158	597	87
Advanced NTR	356	140	496	72
NTR/AB	308	113	421	61

Trade Study Results

The sensitivity of the piloted vehicle IMLEO to variations in the Mars parking orbit apogee altitude is shown in Fig. 2. The IMLEO decreases with increasing Mars capture orbit apogee because as the parking orbit apogee increases a lower velocity change is required to capture and to leave. Although significant decreases in IMLEO occur, the results indicate that orbit selection does not strongly affect the comparisons between aerobraked and all-propulsive systems.

To determine if mission opportunity either affects the comparison between propulsion systems or results in significant mass penalties, the IMLEO requirements for available mission opportunities in the 2000 to 2010 timeframe were compared (Fig.3). Overall, the choice of mission opportunity neither has a major effect on the propulsion system comparison nor leads to major mass penalties for the advanced propulsion systems. The '72 NTR system has a slightly lower IMLEO than the Chem/AB system for the 2002 and 2010 opportunities, but this trend reverses for the 2004 and 2007 opportunities. This reversal is due to the 2004 and 2007 mission options each having substantially greater MOC ΔV requirements (e.g., approximately 5.6 km/s for 2004 versus approximately 4.6 km/s for 2002), which leads to substantial mass increases for the all-propulsive systems. As for propulsion system performance, the advanced NTR has a lower IMLEO than either the Chem/AB or '72 NTR options, with the NTR/AB option always showing the lowest IMLEO of all the systems considered.

The final trade study in the expeditionary mission analysis involved comparing IMLEO for three possible mission modes: the baseline split-sprint, a traditional split-sprint, and a single-vehicle (all-up) mission mode. The traditional split-sprint assumes that both the TEIS and MADV are carried out on the MCV. The TEIS would then be mated to the MPV for the return to Earth. The baseline split-sprint mission transfers the TEIS to the MPV because of a concern over the potential inability to mate the MPV and TEIS when in Mars orbit. The all-up mode moves all hardware (TEIS and MADV) onto the MPV; this provides for mission simplification (one versus two vehicles) at the potential cost of higher total IMLEO. The comparison of all three mission mode options for the 2002 opportunity is shown in Fig. 4. Since the ranking of the propulsion systems from greatest to least IMLEO remains roughly the same over the three modes, the mission mode does not strongly affect the relative

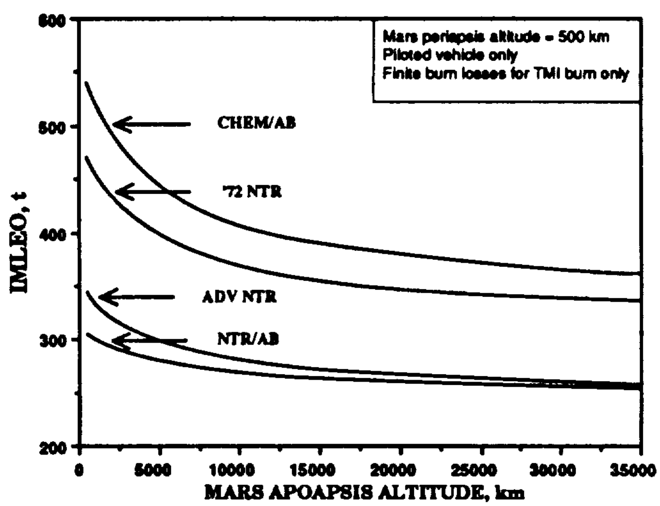

Figure 2 - Expeditionary Mars Exploration, IMLEO Sensitivity to Mars Parking Orbit

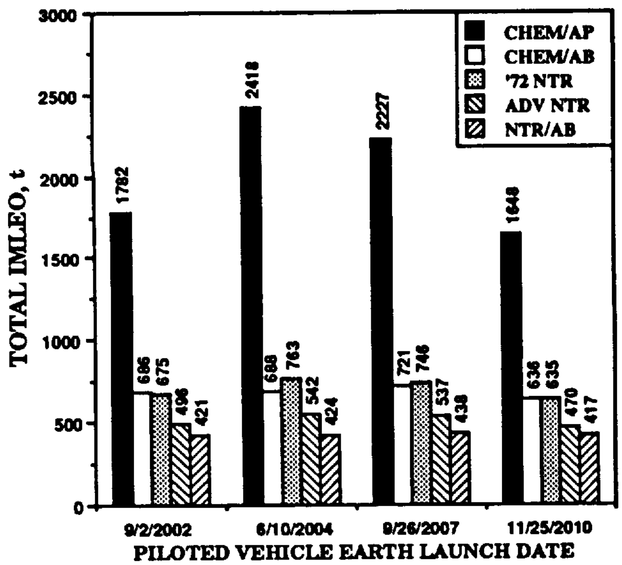

Figure 3 - Expeditionary Mars Exploration, IMLEO Sensitivity to Launch Opportunity

comparisons among the various systems. The mission mode does strongly affect total IMLEO, however. The baseline split-sprint and all-up modes show comparable IMLEO values for all but the Chem/AP system. For the mission modes considered, the traditional split-sprint shows the lowest IMLEO values as a result of transferring the TEIS onto the lower energy, conjunction-class outbound trajectory. In terms of propulsion system performance, Chem/AB and '72 NTR systems have comparable IMLEOs, whereas the advanced NTR and NTR/AB show continuing reductions in IMLEO.

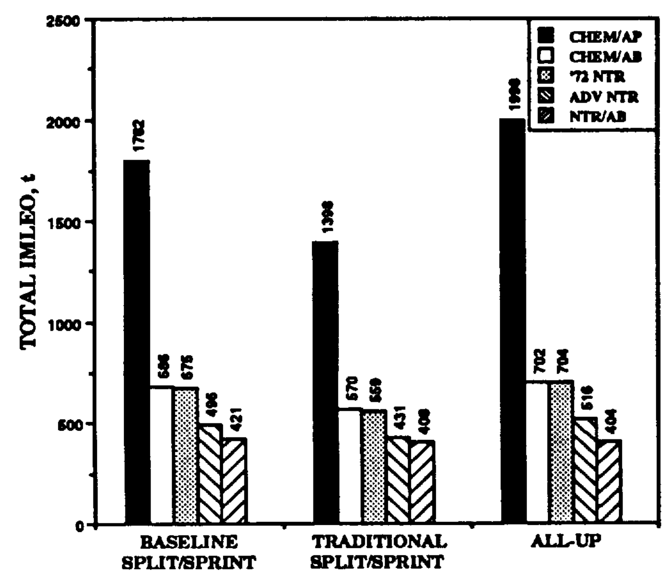

Figure 4 - Expeditionary Mars Exploration, IMLEO Sensitivity to Mission Flight Mode

Evolutionary Mars Exploration Study

The Mars evolutionary case study outlined in references 10 and 15 encompasses a total of seven, all-up flights. The first flight is to be an exploration mission to the Martian moons, Phobos and Deimos, in which a high-energy, opposition-class trajectory is utilized. The next five flights, for the first Mars landing mission and for subsequent missions aimed at establishing a permanently inhabited Mars base, will use lower energy, conjunction-class trajectories. The seventh flight, launched in 2016, will support the fully operational base. At that time, the emphasis in Mars transfer vehicles will switch from slow, artificial-gravity vehicles to fast, zero-g vehicles capable of quick shuttle flights between Earth and Mars. To provide a representative sampling of the evolutionary missions, the first (opposition-class trajectory mission in 2004), fifth (conjunction-class trajectory mission in 2011), and seventh (quick shuttle mission in 2016) flights were selected for detailed study.

For the evolutionary mission analysis, IMLEO requirements for chemical and NTR propulsion systems operating both all propulsively and with aerobrakes were determined for the 2004 and 2011 missions. As with the expeditionary analysis, several trade studies were also performed. The effect of ΔV optimization on the IMLEO for the all-propulsive configurations was investigated for the 2004 mission. The aerobrake mass fractions required to provide Chem/AB IMLEO's comparable to those of all-propulsive NTR vehicles was determined for both the 2004 and 2011 missions. The mass penalty associated with recovery of the Earth-departure stage was also addressed. Finally, the IMLEO's required to support quick (one-way trips of less than 6 months duration) piloted missions to and from an operational Mars base were determined.

Reference Mission Results

The results of the propulsion system comparison for the 2004 and 2011 missions (Table 4) show IMLEO's for the baseline Chem/AB system to be 573 and 662 t, respectively. The increased mass for the 2011 mission is attributed primarily to the additional propellant required to reach the more difficult 6000-km-altitude, circular Phobos parking orbit (versus the less energy-demanding, 250- by 18000-km elliptical orbit assumed for the 2004 mission; see Table 2).

The evolutionary mission's all-propulsive systems incur much greater mass penalties, as compared to the aerobraked vehicles, than in the

Table 4 - Evolutionary Mars Exploration, Propulsion System Comparison

Propulsion System	IMLEO, t		Percent of Chem/AB IMLEO	
	[a]2004	[b]2011	[a]2004	[b]2011
Chem/AB	573	662	100	100
Chem/AP	3800	3141	663	475
'72 NTR	1133	933	198	141
'89 NTR	1031	857	180	129
Advanced NTR	787	680	137	103
NTR/AB	380	443	66	67

[a]2004: First Flight, Opposition-Class Mission
[b]2011: Fifth Flight, Conjunction-Class Mission

expeditionary mission. This is due primarily to the additional propulsive maneuver required to recover the entire core spacecraft into a 500-km circular orbit about Earth. In the expeditionary mission, the core spacecraft is assumed to be expendable and a relatively small crew capsule is used for a direct entry at Earth arrival. Compared to the baseline Chem/AB system, Chem/AP is between 6.6 and 4.8 times heavier for the 2004 and 2011 missions, respectively. This comparison shows quite dramatically the mission leverage that may be realized by chemical systems if a common aerobrake can be developed for use at both Mars and Earth. Similarly, whereas the '72 NTR system was comparable in mass to the Chem/AB system for the expeditionary missions, the baseline evolutionary scenario requires IMLEO values for the '72 NTR that are between 2.0 (for the 2004 mission) and 1.4 (for the 2011 mission) times heavier than the Chem/AB system. For the '89 NTR system, these numbers decrease to 1.8 (2004 mission) and 1.3 (2011 mission). Greater mass savings are realized with the advanced NTR system; the IMLEOs are approximately 1.4 times heavier than the baseline Chem/AB system for the 2004 mission and essentially the same for the 2011 mission. Finally, the NTR/AB has the lowest IMLEO of all the systems studied, with masses ranging from 380 to 443 t. As with the expeditionary mission, the evolutionary mission trade study results for the all-propulsive NTR systems will focus on the '72 NTR and advanced NTR systems.

Trade Study Results

The impact of a given ΔV budget on propulsion system performance was the subject of the first trade study. For aerobraked systems, the ΔV budgets (Table 2) were optimized by accepting larger ΔV increments on the MOC and EOC maneuvers in order to reduce the planetary departure ΔV's. Entry velocities and g-loadings at Mars and Earth may be constrained in some instances

Table 5 - Effect of Minimizing Total Mission ΔV for 2004 Evolutionary Mars Exploration Mission

a) Nominal (no finite burn losses) ΔV summary (km/s)

	Earth Departure	Mars Capture	Mars Departure	Earth Capture	Total
Baseline ΔV's	4.057	4.528	1.773	3.680	14.038
Minimum ΔV Mission	4.076	3.653	2.212	3.549	13.490

b) IMLEO summary for various propulsion systems

Propulsion System	Baseline IMLEO, t	Minimum IMLEO, t	Percent of Baseline Chem/AB [a]B/L ΔV → [b]Min ΔV
Chem/AP	3800	3110	663% → 543%
'72 NTR	1133	988	198% → 172%
Advanced NTR	787	705	137% → 123%

[a]B/L ΔV: Baseline ΔV's
[b]Min ΔV: Minimum ΔV Mission

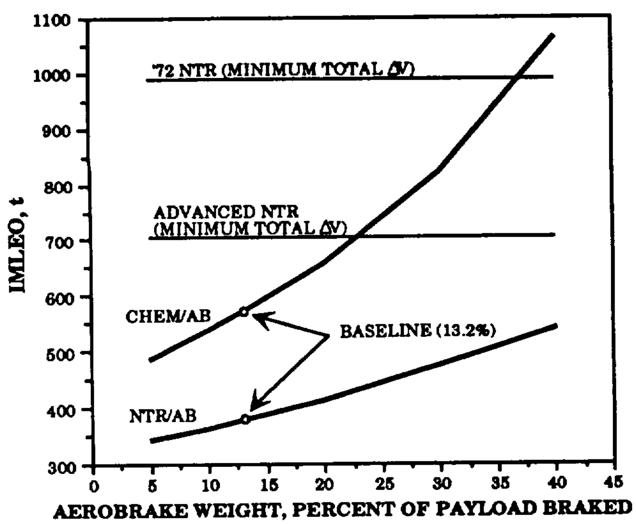

Figure 5 - Evolutionary Mars Exploration, 2004 Mission, IMLEO Sensitivity to Aerobrake Mass

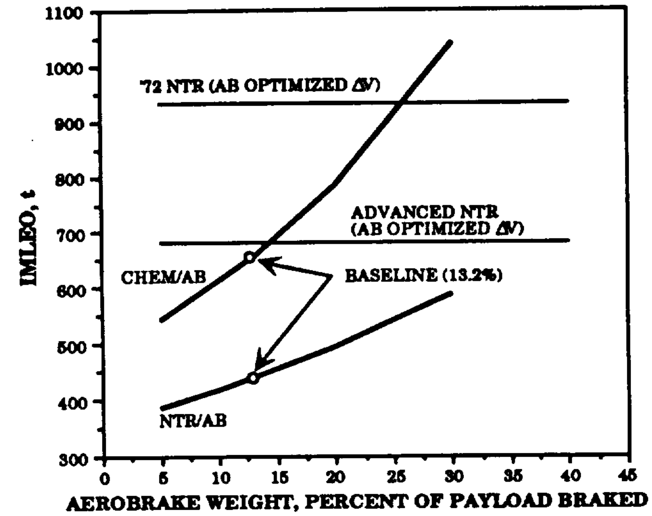

Figure 6 - Evolutionary Mars Exploration, 2011 Mission, IMLEO Sensitivity to Aerobrake Mass

because of safety concerns for the crew and spacecraft. For all-propulsive vehicles, a more reasonable approach is to minimize the total mission ΔV. Such a ΔV budget was calculated for the 2004 mission by varying the Earth and Mars departure and arrival dates. Table 5 shows both the ΔV summaries and the resulting IMLEO for the all-propulsive chemical and NTR systems. By minimizing the total mission ΔV, substantial mass reductions are realized for the all-propulsive systems, ranging from 690 t for the Chem/AP system to 145 t for the '72 NTR to 82 t for the advanced NTR.

The sensitivity of IMLEO to aerobrake mass was examined for both the 2004 and 2011 missions (Figs. 5 and 6, respectively). By varying the aerobrake mass fraction, the cross-over points of comparable IMLEO between the aerobraked chemical and all-propulsive NTR systems were determined. These data show that aerobrake mass, as a percent of payload, can range up to approximately 22 to 37 percent for the 2004 mission and approximately 13 to 26 percent for the 2011 mission and still have smaller IMLEO's than the all-propulsive NTR systems. These data also show the range of mass reductions possible in going from the '72 NTR to the advanced NTR performance levels.

The next trade study investigated the mass penalty associated with recovering the TMIS, in addition to the basic core spacecraft, for the first and fifth flights. The basic recovery profile assumes a retro-burn by the TMIS after insertion of the payload on its interplanetary trajectory. This retro-burn places the TMIS on a 24-hour elliptical orbit, and a final circularization burn returns the stage to its original 500-km altitude parking orbit about Earth. A comparison of the expendable and recoverable IMLEO requirements for both the 2004 and 2011 missions is shown in Table 6.

For the chemical systems, recovery of the TMIS results in a mass penalty of 22 percent (2011 mission) to 37 percent (2004 missions) whereas the range for the NTR systems vary from 24 to 32 percent for the 2004 mission and from 17 to 23 percent for the 2011 mission. A second recovery option examined for the all-propulsive NTR systems assumed a single Phoebus-class engine with staged tanks performs the entire round-trip mission. The corresponding mass penalty associated with this recovery option is significantly less -- ranging from 7 to 10 percent for the 2004 mission, and from 9 to 11 percent for the 2011 mission.

The seventh flight in the evolutionary scenario will depart Earth in 2016 and is intended to initiate the operational phase of the Mars base by extending the number of days available to the crew for Mars surface operations. For a given mission duration, this will be accomplished by reducing interplanetary transit times. A comparison of advanced propulsion systems and their IMLEO requirements is shown in Fig. 7 as a function of one-way trip time to Mars. The vehicles will leave from a 500-km circular orbit about Earth and brake into a 6000-km circular orbit about Mars. Propellant for the Earth return trip is expected to be provided at Phobos either by in-situ propellant production or by fuel transfer from cargo vehicles. Both the outbound Earth-to-Mars and inbound Mars-to-Earth missions were analyzed for equal transit times. One-way trip times as short at 80 days and as long as 6 months were examined. For all cases, the out-

Table 6 - Effect of Earth-Departure Stage Recovery on Evolutionary Mars Exploration Missions

Propulsion System	2004 Mission			2011 Mission		
	Baseline IMLEO	Recovery IMLEO	Increase, %	Baseline IMLEO	Recovery IMLEO	Increase, %
Chem/AB	573	780	36	662	810	22
Chem/AP	3800	5194	37	3141	3828	23
'72 NTR	1133	[a]1456	29	933	[a]1147	23
Advanced NTR	787	[a]972	24	680	793	17
NTR/AB	380	502	32	443	543	23
'72 NTR[b]	--	1209	7	--	1014	9
Adv. NTR[b]	--	864	10	--	758	11

[a]Two 250-klbf engines on TMI stage to reduce finite burn gravity losses.

[b]Single 250-klbf engine for entire mission, percent increase relative to baseline configuration.

bound TMI/MOC tankage for the MPV can accommodate the return propellant requirements for missions of the same duration. Thus, shorter inbound trips are possible by filling the tanks to capacity, and employing higher energy return trajectories. Piloted missions to Mars on the order of 120 days appear to be possible with all-propulsive NTR systems for IMLEO values ranging from approximately 490 t (advanced NTR) to about 750 t ('72 NTR). Shorter trip times are also indicated, but with the steep rise in mass, propellant loadings rapidly become prohibitive.

Aerobraked chemical and NTR systems appear to be capable of very short trip times (80 days for an NTR/AB system with IMLEO less than 500 t) if it were not for the large g-loads and entry velocities encountered at Mars. Reference 15 defined an

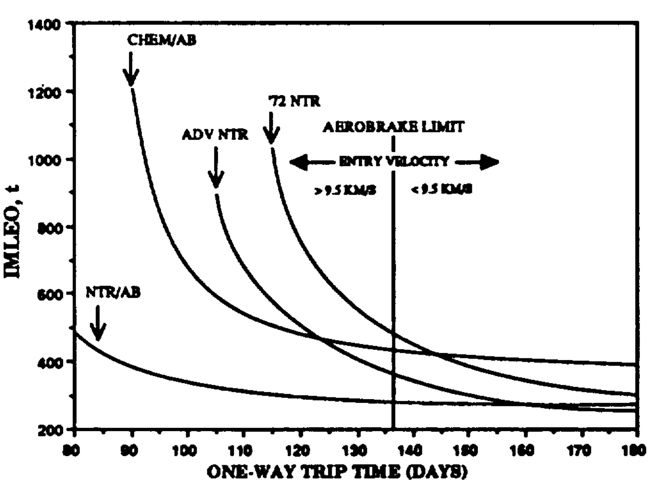

Figure 7 - Evolutionary Exploration of Mars, 2016 Mission, "Quick Trip" Propulsion Comparison

entry velocity limit of 9.5 km/s, thereby restricting one-way trip times for aerobraked systems to approximately 140 days. A combination of propulsive and aerodynamic braking may allow for shorter trip times. As trip time increases, the all-propulsive NTR systems become more mass competitive with the aerobraked systems. This is due to the aerobraked systems needing to provide a fixed ΔV to raise orbit perigee after the Mars arrival aeropass; the Mars capture ΔV requirements decrease for the all-propulsive missions as trip time increases. Depending upon the NTR system, the IMLEO requirements become less for the all-propulsive NTR systems than the Chem/AB system studied at trip times of 125 to 150 days.

Summary

For the expeditionary Mars missions, IMLEO requirements for the baseline chemical/aerobrake and 1972-vintage NTR systems are comparable and can provide approximately a 60 percent reduction in IMLEO over the all-chemical systems. Further mass reductions are possible with '89 NTR and advanced NTR technology levels and by combining NTR and aerobrake technologies. While the choice of Mars parking orbit apogee affects IMLEO, it does not have a strong effect on the comparison between systems. Mission opportunities in the 2000 to 2010 timeframe do not strongly affect the propulsion system comparison, although specific opportunities do provide advantages for all-propulsive sytems over aerobraked systems (and vice versa). Total IMLEO is comparable between single-vehicle and split-sprint mission modes, except for the all-propulsive chemical system in which significant mass savings are realized for split-sprint missions.

For the evolutionary Mars missions, IMLEO requirements for the all-propulsive systems are significantly higher than for the aerobraked systems. This mass increase is attributed primarily to the additional propellant and tankage requirements for spacecraft recovery in Earth orbit. Significant mass reductions are possible, for the all-propulsive vehicles, through mission ΔV optimization. Aerobrake mass for chemical/aerobraked systems can be relatively high and still provide lower IMLEO's than 1972-vintage NTR systems, however by using the '89 NTR and advanced NTR technology levels, the IMLEO benefit of aerobrakes is reduced. Recovery of a separate Earth-departure stage requires IMLEO increases of 17 to 37 percent, depending on the system. By using a single 250-klbf-thrust NTR engine for an entire mission, mass penalties can be reduced to 7 percent of the nominal mission IMLEO. Piloted missions to Mars of 120 to 140 days (and longer) are possible with reasonable IMLEO's for the advanced propulsion systems, however entry velocity limits at Mars may constrain trip times to at least 140 days for aerobraked systems.

Three general conclusions can be drawn from this study:

1. An all-propulsive chemical option for piloted missions to Mars will require IMLEO's approximately two to six times larger than either the chem/AB or NTR systems. While conjunction-class missions can reduce the mass advantage that NTR and chem/AB systems have over the all-chemical system, limiting mission opportunities to only low-energy trajectories is premature at this time.

2. Based on the ground rules and assumptions used in this study, the chem/AB system generally requires somewhat less IMLEO than the NTR for evolutionary-type missions with this trend reversing for the expeditionary mission scenario. Overall, the differences are such that both systems can be considered comparable. Combining the NTR with an aerobrake results in large mass savings; however, this option requires the development of both technologies.

3. As with any study, the results of the Lewis effort depend on the ground rules and assumptions. In comparing chem/AB and NTR systems, the critical assumptions involve the mass of the aerobrake, and the specific impulse and mass of the NTR. The assumed performance levels for the NTR are derived primarily from the established technology base of the 1960's and 1970's. The aerobrake assumptions are more preliminary and are based on a simple linear relationship between the aerobrake weight and the mass to be braked. The technology development and validation work for aerobrakes required to derive a more sophisticated and informed set of ground rules remains to be done. Consequently, it is not clear at this time which technology has the lower IMLEO. The NTR, having an established technology base and substantial mass advantage over the all-chemical system, provides a credible all-propulsive option for piloted missions to Mars. The chem/AB has comparable IMLEO requirements, but lacks the technology maturity to make this option as credible as the NTR. Both options have advantages and problems. Choosing between them will be based on considerations other than IMLEO.

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Nomenclature

AB	aerobrake
ASAO	Advanced Space Analysis Office (NASA Lewis)
AP	all-propulsive
Chem/AB	system with chemical engines and aerobraking
Chem/AP	system with chemical engines
EOCS	Earth orbital capture stage
IMLEO	initial mass in low Earth orbit
LOX/LH_2	liquid oxygen-liquid hydrogen
L/D	aerobrake lift-to-drag ratio
MADV	Mars ascent-descent vehicle
MCC	midcourse correction
MCV	Mars cargo vehicle
MOCS	Mars orbital capture stage
MOO	Mars orbital operations
MPV	Mars piloted vehicle
NERVA	Nuclear Engine for Rocket Vehicle Application program
NTR	nuclear thermal rocket
NTR/AB	system with '72 NTR performance level engines and aerobraking
OEXP	Office of Exploration (NASA headquarters)
RCS	reaction control system
SSME	space shuttle main engine
t	metric tonne = 1000 kg
TEIS	trans-Earth injection stage
TMIS	trans-Mars injection stage

2.2.2

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Advanced Propulsion Options for the Mars Cargo Mission

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ABSTRACT

This report summarizes the results of an evaluation of a variety of advanced low-thrust propulsion options for the cargo-delivery portion of a split-mission piloted Mars exploration scenario. The propulsion options considered were solar sails, 100-MW_e class nuclear electric propulsion (NEP), 100-MW_e class solar electric propulsion (SEP), magnetic sails (magsails), mass drivers, rail guns, solar thermal rockets, beamed-energy (laser and microwave) propulsion systems, and tethers. The requirement was to transport 400 metric tons (MT) of cargo from a 500-km altitude low Earth orbit (LEO) to a 6000-km altitude Mars orbit (e.g., Phobos' orbit) for the 2014 opportunity. The primary figures of merit used in this study were total initial mass in low Earth orbit (IMLEO) and the Earth-to-Mars trip time.

The baseline propulsion system, against which the advanced propulsion concepts were compared, was an aerobraked chemical (O_2/H_2) propulsion system with a specific impulse (I_{sp}) of 470 lb_f-s/lb_m. This system had an initial total mass in LEO of 1646 MT (including payload) and had an Earth-to-Mars trip time of 294 days. It was found that solar sails can provide the greatest mass savings over the baseline chemical system. However, solar sails suffer from having very long trip times. A good performance compromise between a low IMLEO and a short trip time can be obtained by using 100-MW_e class NEP systems; they can even be lighter and faster overall than the baseline chemical system. Such systems may be particularly suited to the piloted portion of the mission, where a premium is placed on trip time. A 100-MW_e SEP system is a close competitor to the NEP system, providing almost as good a performance, but without the technological, operational, or "political" constraints of space nuclear power.

Magsail, mass driver, beamed-energy, and tether concepts were found to have moderate benefits in mass or trip time, but their performance is contingent on several factors which could reduce their effectiveness. For example, the magsail concept, like the solar sail, has infinite specific impulse. However, magsails can only operate far from a planet; this imposes a large infrastructure overhead since a fleet of orbit transfer vehicles (OTV) are required to transport the magsails and their payloads from LEO to the magsail operational orbit. Mass drivers have a low I_{sp} for the Mars cargo mission but they do have a high efficiency (electric-to-jet power). They also can make use of any material as propellant. Thus, if copious amounts of "free" lunar O_2 propellant were available, a mass driver operating at modest power levels (10 MW_e or less) could show a mass savings over the baseline system, and do so for trip times on the order of 500 days. However, this is contingent on the availability of "free" lunar propellant; without this "free" propellant, the mass driver is not competitive. Beamed-energy concepts were found to provide some benefits in mass when used as OTVs to deploy the payload (with a chemical O_2/H_2 stage for Earth escape and aerocapture at Mars) at GEO altitudes. A laser-augmented SEP vehicle used for the round trip to Mars also provides significant trip time savings over an un-augmented SEP system, since the laser provides a rapid Earth escape/capture. However, all the beamed-energy concepts suffer from the limited range over which power can be beamed (e.g., microwaves to GEO or near-visible light to the Moon). Even the laser-augmented SEP system, which reverts to a normal solar powered SEP far from the Earth, requires very high-powered lasers (10-MW beam or more) to provide any significant trip time savings. Also, the space-based infrastructure (laser/microwave power stations, orbital relay mirrors) required to support beamed-energy transmission would need to be "amortized" over many users. Lastly, tether systems show only a small advantage in IMLEO over the baseline system. This is due primarily to the need to break up the 400 MT payload into twenty 20-MT segments, each with its own chemical O_2/H_2 stage for tether-assisted Earth escape and Mars capture. Also, there is a significant LEO, Deimos, and Phobos tether station set-up mass investment which must be "amortized"

over many missions. However, tethers may have greater benefits for the piloted portion of the mission. For example, tethers can be used to lower (de-orbit) landers and raise ascent vehicles. Also, a tether station on Deimos can provide a vehicle returning to Earth with Mars' escape velocity, thereby greatly reducing the trans-Earth injection propulsion requirements.

Two concepts were found to have very poor performance for the Mars cargo mission scenario assumed in this study. These were solar thermal propulsion and rail guns. Solar thermal propulsion suffers from having too low an I_{sp} (1200 lb_f-s/lb_m) for this mission. Rail guns suffer from both a low I_{sp} and a low efficiency (electric-to-jet power); they require high powers (50 MW_e) for optimum performance and can only show a mass savings over the baseline chemical system if copious amounts of "free" lunar oxygen are available as propellant in LEO.

Based on the results of this study, solar sails, 100-MW_e class NEP systems, and 100-MW_e class SEP systems should be considered in detail for application to the Mars cargo mission. Further, 100-MW_e class NEP and SEP systems should be evaluated in detail for the piloted portion of future Mars missions since they have the potential for significant savings in both IMLEO and trip time as compared to the baseline chemical systems. Similarly, tethers should be evaluated for the piloted portion of the Mars mission since they may provide major savings in mass for the Mars-to-Earth portion of the trip. Magsails, mass drivers, and beamed-energy concepts should also be considered for the Mars cargo mission, although their performance will depend on a number of factors (e.g., "amortization" of a space-based laser for laser propulsion vehicles).

Finally, it should be noted that the conclusions reached in this study are highly mission-scenario dependent. Thus, a concept that has no benefit for the Mars cargo mission scenario assumed in this study may show significant benefits for the piloted mission. Similarly, concepts that are not attractive for Mars missions may provide major benefits when used for cis-lunar missions (e.g., LEO-to-GEO OTVs or lunar base missions). Also, different thrusting or trajectory strategies (e.g., low-thrust spiral planetary escape or capture, as used in this study, versus multiple-impulse medium-thrust trajectories) may have a significant impact on performance. Furthermore, in this study, the concepts were used in a "pure" Mars cargo mission mode with a minimum of mixing of modes. For example, only the beamed-energy concepts were used in a LEO-to-GEO OTV mode due to the limitations in transmission distances. Future studies should consider the option of "mixed" mission modes of operation; such as, for example, the use of an advanced concept for a LEO-to-GEO OTV-type transfer followed by trans-Mars injection by a second system. This may be a particularly attractive approach, since a number of previous studies have shown that systems with I_{sp}s of 1000 to 1500 lb_f-s/lb_m (e.g., mass drivers, rail guns, solar thermal propulsion, laser/microwave thermal propulsion) can provide major savings in IMLEO as compared to chemical systems, and savings in trip time as compared to high-I_{sp} electric propulsion systems at comparable power levels. Finally, the same advanced propulsion concepts considered in this study for the Mars cargo mission should also be evaluated for the lunar base cargo mission, again with IMLEO and trip time as the primary figures of merit.

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CONTENTS

SECTION 1

INTRODUCTION

There are a wide variety of advanced propulsion concepts which hold the potential for significantly reducing the initial mass in low Earth orbit (IMLEO) or reducing the trip time required for missions to support future NASA piloted missions to Mars. The overall objective of this study was to evaluate the benefits (in terms of reduced IMLEO and trip time) of the use of several advanced low-thrust propulsion concepts for the cargo mission portion of a split piloted Mars mission in the year 2014. The concepts evaluated in this study include those that derive their power from sunlight or laser light, as well as those that use electric power from a nuclear reactor or solar photovoltaic cells.

1.1 CONCEPTS EVALUATED

Concepts and mission scenarios evaluated in this study are summarized in Figures 1-1 and 1-2. Those concepts which use sunlight directly include the solar sail, which uses momentum exchange from solar photons to "push" a gossamer sail, and the solar thermal rocket, which focuses sunlight into a thrust chamber to heat a propellant working fluid like hydrogen, which is then expelled through a conventional nozzle. A concept related to the solar sail is the magnetic sail (mag sail), which uses a magnetic interaction with the charged particles in the solar wind to "push" the "sail" (actually a superconducting solenoid magnet ring).

Two concepts which directly use beamed energy (e.g., laser light) from a remote beam source are the laser thermal rocket and the microwave thermal rocket. The laser thermal rocket is similar to the solar thermal rocket except that near-visible laser light from a remote laser transmitter (ground or space-based) is used instead of sunlight. Two types of microwave thermal rocket concepts are possible. The first is the analog of the laser thermal rocket in that microwave radiation is absorbed by the propellant and used to heat the propellant. By contrast, the electron-cyclotron resonance (ECR) microwave thruster concept uses a microwave beam to directly excite a propellant and expel it; the propellant is in fact not just heated thermally but rather is excited electromagnetically by coupling to the energy in the microwave beam. The ECR thruster concept is the one selected in this study for use with the microwave 'thermal" propulsion system

The laser or microwave radiation can also be used indirectly to power an electric thruster (e.g., ion thruster) by first converting the incoming photons to electricity by either "solar" photovoltaic cells (near-visible wavelength) or by a rectenna (microwave wavelength); these concepts represent electric propulsion vehicles with a potentially light-weight power supply (receiver) on the vehicle because the actual power supply (transmitter) is remotely located on the ground or in low Earth orbit (LEO).

A second general category of concepts are those which use a nuclear or solar electric power supply to operate electric propulsion thrusters. These include 100-MW class Nuclear Electric Propulsion (NEP) and Solar Electric Propulsion (SEP), as well as megawatt-class rail guns and mass drivers. In the rail gun and mass driver, the propellant is in the form of "pellets" which are accelerated electromagnetically in a "bucket" and shot out from the vehicle to provide thrust. Rail guns and mass drivers can use any material as the "pellet" mass and thus could use extraterrestrial materials as a propellant source, thus reducing the required IMLEO.

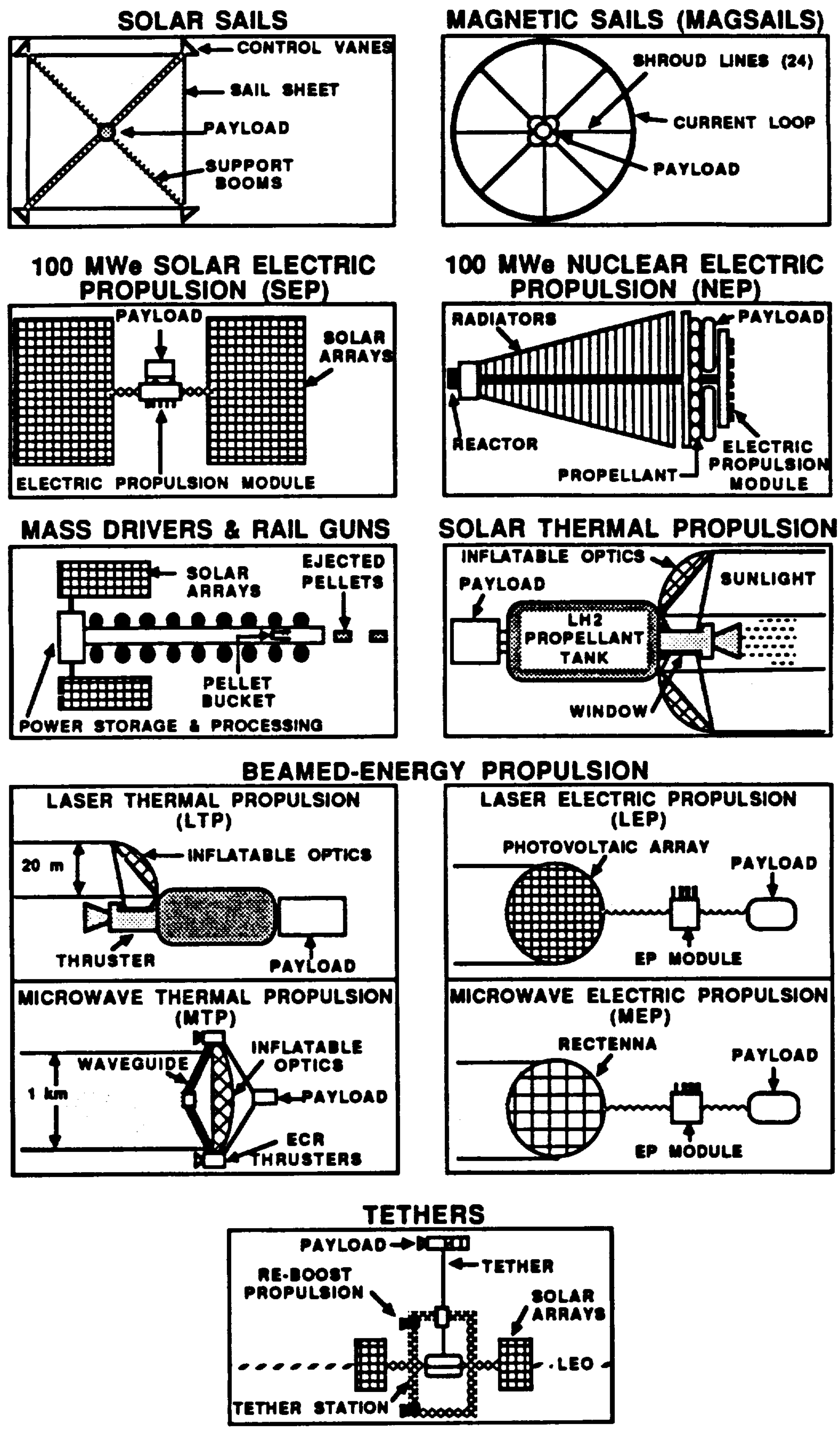

Figure 1-1. Advanced Propulsion Concepts Evaluated in This Study

CONCEPT	MIN/MAX ALTITUDE	TRANSFER TO MIN/MAX ALT FROM LEO	TRANSFER TO MARS & MARS ORBIT INSERTION	RETURN TO EARTH
• SOLAR SAIL • ADV. SAIL	2000 km (DRAG-FREE)	• CHEM • SEP	• SOLAR SAIL • ADV. SAIL	• SOLAR SAIL • ADV. SAIL
• MAGSAIL	1 AU (SOLAR WIND)	• SEP	• MAGSAIL	• MAGSAIL
• 100 MW SEP • SOLAR THERM • MASS DRIVER & RAIL GUN	(LEO) (LEO) (LEO)	(N/A)	• 100 MW SEP • STP • MD & RG	• 100 MW SEP • NONE • MD & RG
• 100 MW NEP	1000 km (NSO)	• SEP • 100 MW NEP (1st TIME)	• 100 MW NEP	• 100 MW NEP
• LASER THERM • μ-WAVE THERM • μ-WAVE ELECT • LASER ELECT	LEO>GEO LEO>GEO>LEO LEO>GEO>LEO LEO>GEO>LEO	• LTP • μTP • μEP • LEP	CHEM (O2/H2) AEROCAPTURE	NONE
• LASER ELECT	(LEO)	• LEP	SEP AFTER EARTH ESCAPE (LEP USED AS SEP)	SEP TO EARTH CAPTURE, THEN LEP TO LEO

Figure 1-2. Mission Scenarios

A final, non-propulsive concept is the use of tethers for orbit raising and lowering in Earth and Mars orbit, respectively. The use of tethers can significantly reduce the requirement of the spacecraft by using long cables to reel the spacecraft in or out of the deep gravity well of a planet and thus raise or lower orbits.

1.2 TRADE STUDIES

As mentioned above, the primary figures-of-merit used in evaluating concepts for this study were the initial mass in LEO and trip time required for the Mars cargo mission. The primary focus is on total system mass, including the empty or "dry" vehicle weights, propellant, and payload (400 MT total to Mars/Phobos orbit). Also included in the total mass is the weight of any supporting infrastructure. This infrastructure can take many forms, depending on the concept and mission scenario. For example, several of the concepts cannot operate directly from LEO, but instead have some minimum altitude at which they must operate. Thus, an added fleet of orbit transfer vehicles (OTVs) is required to boost the system from LEO to the minimum operating altitude; the dry weight and propellant required for the OTV fleet is included in the infrastructure mass requirement.

For trip times, the primary figure-of-merit is the Earth-to-Mars trip time, since the cargo mission is a one-way delivery. In most cases, however, the vehicles are re-usable, so a Mars-to-Earth trip time is also found. The round-trip time is important if the vehicles are to be phased properly with subsequent launch opportunities. For example, a system with a round-trip time of less than the Earth-Mars synodic period (2.2 years) could be used for the next launch opportunity; longer round-trip times would require skipping one or

more opportunities, thus requiring a larger overall cargo vehicle fleet for continuous operations.

However, in this study, it is assumed that the full system must be deployed the "first" time, so all associated masses are included and only the Earth-to-Mars delivery time is considered in detail. Re-use and "amortization" of vehicles for multiple cargo delivery cycles should be considered in detail in future studies to identify benefits and penalties associated with re-use of vehicles for a continuous Mars base operation and growth.

1.3 APPROACH

Results for each of the concepts listed above is described in detail below; the general approach used in evaluating each concept is discussed next. For each concept, the first step involves definition of a mission scenario. This includes the identification of a minimum operational altitude, as, for example, for solar sails which cannot operate below a 2000-km altitude due to air drag. The requirement of a high-altitude operation node impacts the infrastructure requirements, since an OTV is then required to transport propellant, cargo, and empty ("dry") vehicles from an assumed 500-km LEO base (e.g., space station), since the IMLEO is figured at a 500-km altitude base node. Thus, the total IMLEO would also have to include the OTVs (and their propellant) that are required to support operation of a solar sail.

Another example of the way in which the mission scenario impacts the overall operation of the concept is in the area of the laser propulsion concepts. Since the size of the transmitter and receiver optics increase with increasing transmission distance, these concepts are limited to operation only near the Earth. This makes it possible to achieve most or all of the Earth-escape velocity requirement (typically the largest Delta-V requirement in the mission) with the high-performance laser propulsion concept, but then a second stage (e.g., an aerobraked oxygen/hydrogen chemical propulsion system) is required for the Earth-to-Mars and Mars-orbit-insertion steps in the mission.

A final element in the mission scenarios involves the issue of re-use of the vehicle and thus the need to carry propellant for the return to Earth orbit. A qualitative assessment was made such that "expensive," complex electric propulsion vehicles are returned to Earth orbit; these include SEP, NEP, rail guns, mass drivers, and laser (visible and microwave) electric vehicles. By contrast, "inexpensive," simple thermal propulsion vehicles are expendable; these include aerobraked O_2/H_2, solar thermal and laser (visible) thermal systems, although the microwave thermal vehicle, because of its size and complexity, is returned. Finally, solar and mag sails are returned to Earth orbit because they use no propellant.

The second step in the analysis processes involves development of a series of scaling equations for each of the concepts so as to determine the empty or "dry" mass (M_{Dry}) of the vehicle. In general, the dry weight will be a function of the total propellant load (propellant tankage, refrigerators for cryogenic propellants), power (power supply, thrusters, power processors), specific impulse (I_{sp}), and efficiency of the various subsystems. Scaling equations are derived from a combination of literature sources and in-house analyses and evaluation.

The final step in the analysis involves the use of low-thrust trajectory computer codes to determine the propellant requirements and trip time as a function of exhaust (jet) power, specific impulse, and initial mass (dry, propellant, and payload mass). Combined with infrastructure requirements, output from the trajectory code permits evaluation of the IMLEO and Earth-to-Mars trip times as the primary figures-of-merit for each concept.

1.4 ASSUMPTIONS COMMON TO ALL CONCEPTS

Several ground rules and assumptions were established which were common for all of the concepts. The first was that the time frame of the mission be the year 2014. The primary requirement is to transport 400 metric tons (MT) of cargo from a 500-km, 28.5° low Earth orbit (LEO). This initial starting node was chosen as typical of a space station orbit. All calculations of IMLEO use this LEO node as a reference point. The payload is delivered via a slow minimum-energy conjunction-class trajectory to a 6000-km Mars orbit. This orbit is taken as the delivery node; it is at the same altitude as Phobos, although the need to actually rendezvous and land on Phobos was not considered in detail.

Several of the concepts described below are large in size; it was assumed that it would be neither practical nor desirable to have these vehicles dock directly with a space station or base in LEO or Mars orbit. Instead, a separate chemical stage was added to the payload to provide a small Delta-V capability (50 m/s) for any required rendezvous and docking of payloads in Earth or Mars orbit. For this purpose, the Orbital Maneuvering Vehicle (OMV) was used. This vehicle has a "dry" weight (M_{Dry}) of 4035 kg and a useable propellant (M_p) capacity of 4286 kg with an I_{sp} of 300 lb_f-s/lbm.[1] The OMV can provide a 50-m/s Delta-V for payloads weighing up to 100 MT; for payloads in excess of 100 MT, a "stretched" OMV was used with the following scaling equation:

$$M_{Dry}\ \mathrm{OMV} = 3136.1 + 0.20972 \cdot M_p \quad \text{[all masses in kg]}$$

Also, the OMV has a 463 W electric power system composed of solar cells and batteries (for shadow periods). Even though sunlight intensity at Mars is less than half that at Earth, the amount of time spent in sunlight and shadow in a 6000-km altitude Mars orbit is such that the OMV power system can provide about 66 % of its rated power at Mars.

In addition, structural or docking adapters were added to the payloads, thus increasing the "effective" payload weight. This is illustrated in Fig. 1-3 for the case of the OMV. Note that some of the concepts and mission scenarios require aerobraking of the payload into Mars orbit; this is performed by an O_2/H_2 stage with an I_{sp} of 470 lb_f-s/lbm and an aerobrake mass corresponding to 15 % of the vehicle (stage, propellant, and payload) mass at the start of the aeromaneuver.

Another study ground-rule was that the total 400 MT payload could be split into smaller units, such that the smallest unit was 20 MT. Thus, it is possible to see the effect on IMLEO and trip time by increasing the number of vehicles, but decreasing the payload per vehicle (and thus mass per vehicle), e.g. one vehicle (with a 400 MT payload), two vehicles flying in parallel (each with 200 MT payload), and so on to 20 vehicles (each with a 20 MT payload).

A final study assumption was that only one "new" or advanced concept be used at a time. For example, an aerobraked O_2/H_2 vehicle was used with the tether concept; a 100-kW class solar electric propulsion vehicle was used as the OTV for those concepts that cannot leave directly from LEO (e.g., solar sails). In the context assumed in this study, aerobraked chemical or 100-kW class SEP vehicles are considered to represent the baseline (non-advanced) propulsion technology available in the year 2014 time frame assumed for this study. Similarly, in the laser propulsion concepts, the beam power is limited to ranges of 1 to 10 MW since this would require electric power supplies for the lasers of 10 to 100 MW (electric) assuming a 10% electric-to-laser efficiency; in this case, beam powers in excess of 10 MW would require 100-MW class electric power supplies which would be considered a second "new" technology in addition to the laser. One area

that should be considered in future studies are synergistic combinations of advanced propulsion concepts (e.g., tethers and high powered SEP vehicles).

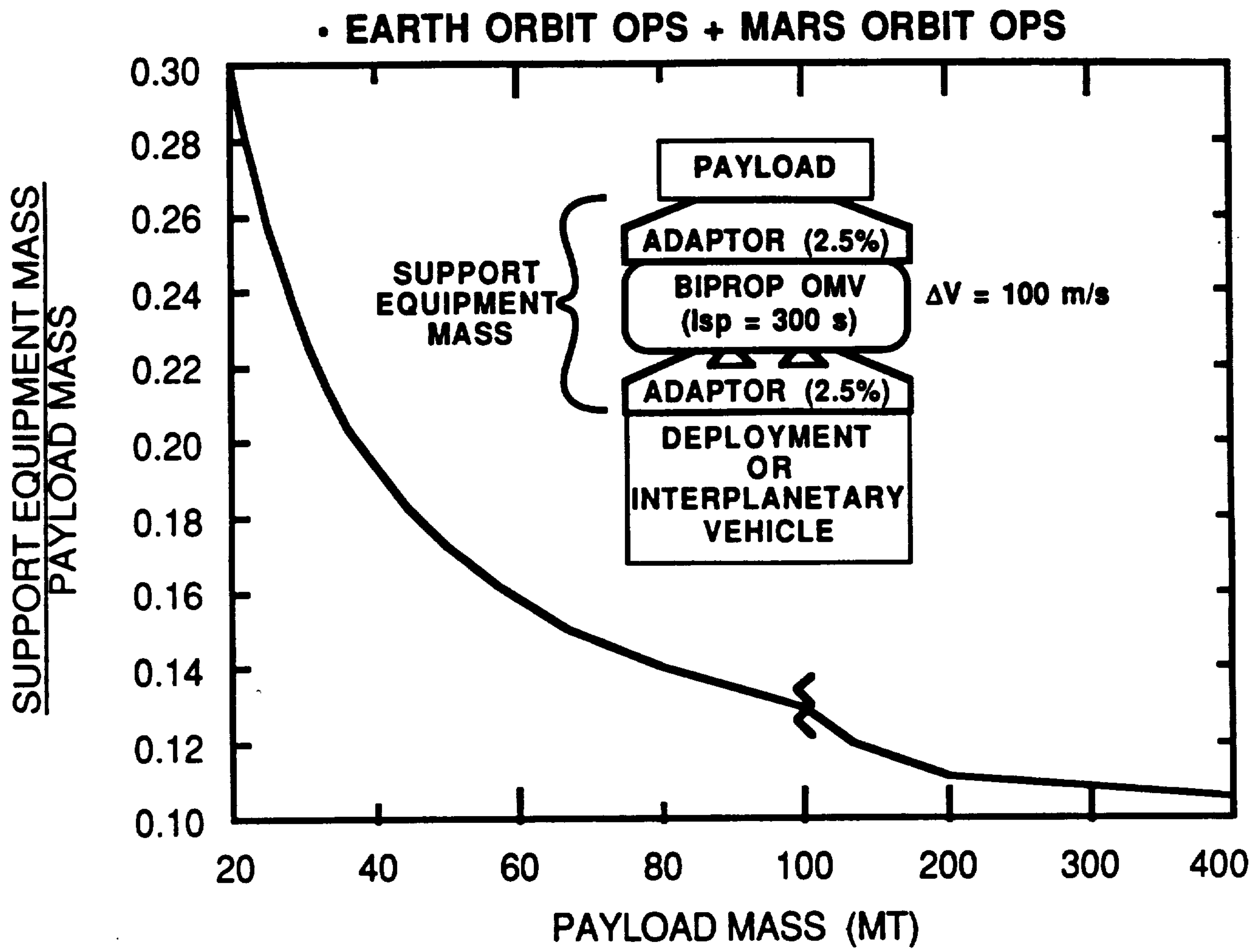

Figure 1-3. Effective Payload Weight Due to Adding an Orbital Maneuvering Vehicle (OMV) for Earth and Mars Orbital Operations

1.5 RESULTS FOR THE BASELINE AEROBRAKED CHEMICAL (O_2/H_2) SYSTEM

1.5.1 Introduction

For comparison purposes, the IMLEO and trip time for an aerobraked chemical (O_2/H_2) propulsion vehicle was found. A two-stage vehicle is used; both are expendable stages. The first stage is used for Earth escape and injection towards Mars, after which it is jettisoned. The second stage performs an Earth-to-Mars trajectory correction maneuver (TCM) and an aerobraking maneuver into an elliptical orbit with an apoapsis at Phobos' orbital altitude and a periapsis nominally at zero altitude above the surface of Mars. After the aeromaneuver, the aeroshell is jettisoned and the orbit circularized to match Phobos' orbit.

1.5.2 Vehicle Sizing

For this analysis, the full 400 MT payload (with a 2.5% adaptor) is sent to Mars. Both stages have a specific impulse (I_{sp}) of 470 lb_f-s/lb_m; the aeroshell on the second stage has a weight of 15 % of the vehicle aerobraked. Active refrigeration is added to both stages to store the cryogenic O_2/H_2 propellants. Finally, an adaptor is added between the first and second stage with a mass of 2.5 % of the loaded second stage (stage, aeroshell, payload).

The basic O_2/H_2 stage has a mass scaling equation[2] of:

$$M_{Dry}\ O_2/H_2\ \text{Stage} = 1000.0 + 0.115 \cdot M_p \quad \text{[all masses in kg]}$$

where M_{Dry} is the stage "dry" mass and M_p is the useable propellant mass. In this equation, the factor 1000.0 is the "fixed" mass of the vehicle (engines, avionics, etc.) and the factor 0.115 is the "tankage factor" for those components that scale with propellant mass (tanks, insulation, structure, etc.).

As mentioned above, an active refrigeration system is added to each stage to prevent boiloff of the cryogenic propellants during long storage or coast periods. For this purpose, a sorption compressor refrigerator is assumed. Sorption compressors operate by first adsorbing a gas on a suitable adsorbent material at low temperature (e.g., methane adsorbed on activated charcoal), and then desorbing the gas at high temperature to produce a high pressure gas. The gas is then expanded through a Joule-Thomson valve to produce cooling. Sorption refrigerators are currently under development for sensor cooling applications;[3] they have the advantage of having none of the rotating turbomachinery of conventional compressors. They also do not require large amounts of electric power, since the sorption/desorption cycle can be accomplished with waste thermal heat such as that available from an RTG. The scaling equation for a 3-stage (thermoelectric cooler, O_2 sorption cooler, H_2 sorption cooler) sorption refrigerator for liquid hydrogen temperatures (20 K) is:[4]

$$M_{Frig}\ [20\ K] = 45.9 + 21.1 \cdot W_{cool} \quad \text{[all masses in kg]}$$

where M_{Frig} is the mass of the sorption refrigerator (sorbent canisters, valves, radiators, RTG power supply) and W_{cool} is the total cooling load in Watts. The total cooling load for a 4-tank O_2/H_2 stage (2-O_2 + 2-H_2 tanks) is:[5]

$$W_{cool} = 0.04252 \cdot (M_p)^{2/3} \quad \text{[4-tank } O_2/H_2 \text{ stage, all masses in kg]}$$

Thus, the sorption refrigerator mass scaling equation for the 4-tank O_2/H_2 stage is:

$$M_{Frig}\ [20\ K] = 45.9 + 0.8972 \cdot (M_p)^{2/3} \quad \text{[all masses in kg]}$$

This refrigerator represents about 8 to 15 % of the overall stage dry mass for the first and second stage, respectively, of the vehicle.

The final scaling equation for the 4-tank O_2/H_2 stage (less adaptors and aeroshell) is thus:

$$M_{Dry}\ O_2/H_2\ \text{Stage w/ Frig} = 1045.9 + 0.115 \cdot M_p + 0.8972 \cdot (M_p)^{2/3} \quad \text{[all masses in kg]}$$

$$= 1.0459 + 0.115 \cdot M_p + 0.08972 \cdot (M_p)^{2/3} \quad \text{[all masses in MT]}$$

1.5.3 Delta-V Requirements

Trans-Mars insertion requires a hyperbolic excess velocity of 3.162 km/s; or a C_3 of 9.541 km^2/s^2; this corresponds to a Delta-V of 3.588 km/s from a 500-km LEO. This launch opportunity occurs on December 12, 2013 and has a trip time of 294 days. This increases to a C_3 of 15.041 km^2/s^2, but with a trip time of 224 days, by January 18, 2014.[6] For these analyses, the lower C_3 (9.541 km^2/s^2) is used. A 100-m/s Delta-V for Earth-orbit operations, performance losses, etc., is added to the required injection Delta-V to give a total of 3.688 km/s for the first stage. The second stage Delta-V budget includes a 100-m/s TCM, followed by aerobraking. The aeroshell is then jettisoned, and the orbit circularized. This consists of a 580-m/s orbit circularization (0x6000-km to 6000x6000-km altitudes); this is rounded to 600 m/s to include a small performance margin. A final 50 m/s for rendezvous and docking at the Mars-orbit base is added to give a total of 100 m/s pre-aerobrake and 650 m/s post-aerobrake for the second stage.

1.5.4 Results

With these assumptions, the baseline chemical O_2/H_2 system has an IMLEO of 1646 MT, as shown in Table 1-1, and an Earth-to-Mars trip time of 294 days. This system, illustrated in Fig. 1-4, serves as a baseline against which the IMLEO and trip time for the advanced propulsion options will be compared. The total IMLEO includes the 400-MT payload, a 10-MT adaptor between the payload and stage 2, a 160-MT second stage, a 14-MT adaptor between stage 1 and stage 2, and a 955-MT first stage. In addition, it is assumed that the stages will be launched dry from the Earth to LEO, so a dedicated propellant resupply tanker is required. This O_2/H_2 propellant tanker has a dry weight of 106 MT, assuming a tankage factor the same as the O_2/H_2 stages (0.115).

Table 1-1. Baseline O_2/H_2 Vehicle Mass Summary

Element	Mass (MT)	Delta-V (km/s)
Payload	400.0	
Stage 2		
Adaptor (2.5 %)	10.0	
M_{Dry}	11.4	
M_p (Post-Aerocapture)	63.9	0.650
Aeroshell (15 %)	72.8	
M_p (Pre-Aerocapture)	12.2	0.100
Total	170.3	
Stage 1		
Adaptor (2.5 %)	14.3	
M_{Dry}	106.6	
M_p (Earth Escape)	848.2	3.688
Total	969.1	
Total Stage 1 + Stage 2	1539.4	
Resupply Tanker ($0.115 \cdot M_p$ Total)	106.3	
Total IMLEO	1645.7	

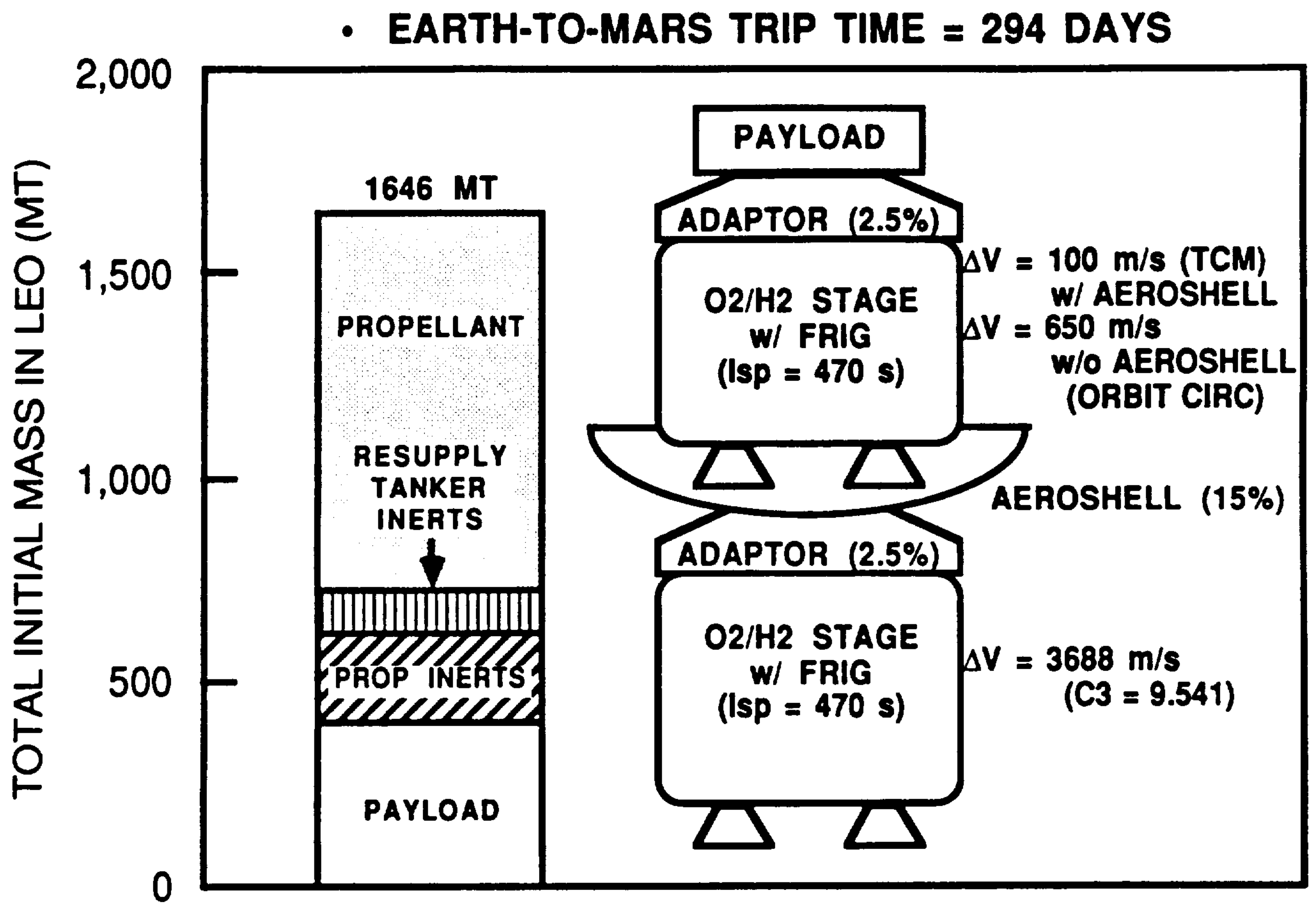

Figure 1-4. Initial Mass in Low Earth Orbit for the Baseline Chemical O_2/H_2 System

1.6 REFERENCES

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2. Palaszewski, B.A., et al., "Nuclear-Electric Propulsion: Manned Mars Propulsion Options," Presented at the Case for Mars III Conference, Boulder Colorado, July 22,1987.

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4. Palaszewski, B.A., and Frisbee, R.H., "Advanced Propulsion for the Mars Rover Sample Return Mission," AIAA Paper AIAA-88-2900, Presented at the AIAA/ASME/SAE/ASEE 24th Joint Propulsion Conference, Boston Mass. , July 11, 1988; and Jones, J.A., Personal Communication, December 1987.

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SECTION 2

SOLAR SAILS

2.1 INTRODUCTION

Solar sails operate by using momentum exchange with solar photons; this amounts to a force of 9 Newtons/km^2 at 1 AU. As such, a solar sail has "infinite" specific impulse, because it requires no propellant, but it has a low acceleration resulting in long trip times. Also, solar sails are typically large, gossamer structures with dimensions of kilometers; for example, a typical solar sail has an area of 4 km^2.

Solar sails were first extensively studied in the late 1970's for the Halley Comet rendezvous mission.[1] At that time, there was an extensive analyses made of solar sail fabrication techniques (thin silvered sheets and light-weight booms), control and dynamics, and trajectory analysis. The study found that solar sails were eminently feasible from a technology and mission performance point of view, but the development risk was considered too high for the short time available before launch. Instead, Solar Electric Propulsion (SEP) was considered less risky given the mission's schedule constraints.

Although the Halley Comet mission was not pursued by the United States, interest in solar sails for a variety of lunar and Mars cargo missions, as well as planetary mission, has continued because solar sails represent the most fuel efficient possible inter-orbital "supertanker" in space. Solar sails have been extensively studied in the past for Mars cargo missions; much of the discussions below are derived from these studies.[2,3]

Figure 2-1 illustrates two solar sail concepts. The first is the classic square sail consisting of a thin (few mills) sheet of silvered or aluminized plastic stretched over a supporting light-weight boom. Small "fly swatter" vanes are located at the corners of the sail; they have a combined area of 0.5% of the total sail area and are rotated to produce differential light pressure for use in maneuvering the sail.[3] The sail can also be maneuvered by shifting the payload so that the center of mass is offset from the center of (light) pressure. The second type of sail illustrated in Fig. 2-1 is the heliogyro solar sail. In this concept, the sail is spun like a helicopter blade; the sail material is unrolled and stabilized by centrifugal force. Maneuvering is accomplished by changing the "pitch" of the blades. The heliogyro sail is easier to deploy than the square sail; has a greater stability from random disturbances (due to its rotational inertia), but has a slower maneuvering rate due to the rotational inertia.[1] Thus, the two types of sails have different strengths and weaknesses, although the square sail, with its faster maneuvering (turning) response, might be favored for missions involving extensive planetary escape and capture spiral orbits (because the sail has to re-orient itself relative to the sun on each orbit).

Currently, there is no NASA-funded work on solar sails, although a private organization, the World Space Foundation, has built a prototype sail (880 m^2 area) as a demonstration of the required on-orbit deployment and maneuvering capability. The group is awaiting a launch vehicle to place the sail in a high-altitude orbit, because a sail cannot operate below an altitude of about 2000 km due to air drag would exceeding photon pressure at a lower altitude.[4]

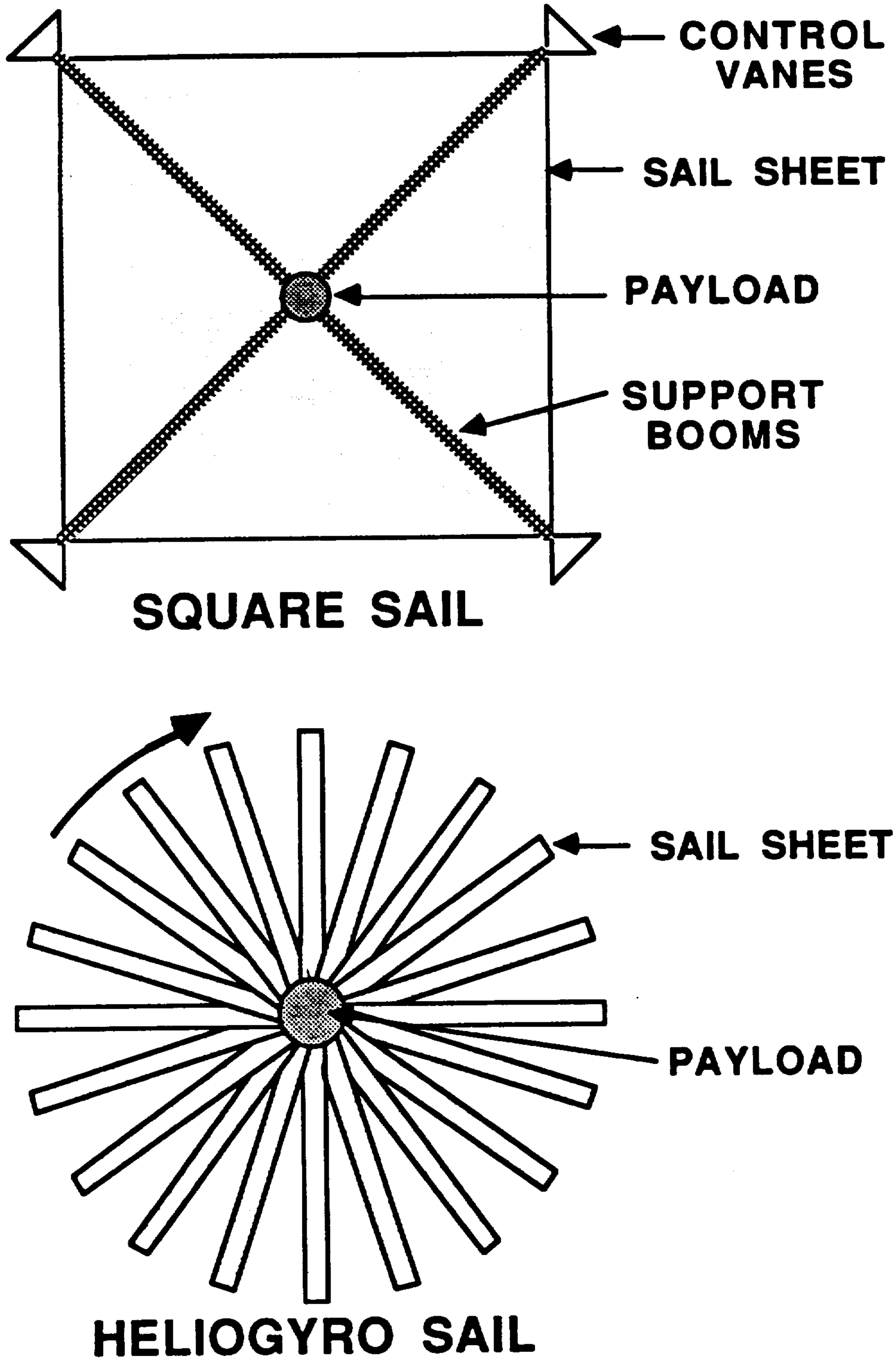

Figure 2-1. Solar Sail Concepts

In this study, no distinction is made between square and heliogyro sails. Instead, the primary performance parameter is the areal density (grams/m^2) of the sail. This parameter is an important measure of sail performance, because it determines the acceleration of the sail (i.e., solar pressure [N/km^2] divided by areal density [g/m^2] gives

acceleration). Areal density, in turn, is determined by both the thickness of the sail sheeting and the supporting structure. For example, in the Halley Comet mission and more recently in studies by Staehle on sails for Mars cargo missions,[2] deployable sails were assumed with a total areal density (sail sheet plus structure) of 5 g/m^2. A deployable sail requires relatively thick sail sheet (e.g., 2.5-micron thick Kapton) to survive folding (on the ground) and packing into a launch vehicle, followed by unfolding (deployment) on orbit. By contrast, Garvey[3] and Drexler[5] have considered sails erected or constructed (fabricated) on orbit; because these sails do not need to be folded/unfolded, the sheet can be much thinner (e.g., 0.015 to 0.2-microns thick). This results in sails which are erected or fabricated on-orbit with areal densities ranging from 1.0 g/m^2 (Garvey) to less than 0.3 g/m^2 (Drexler). Thus, a Garvey- or Drexler-type sail could have significantly higher acceleration, and thus shorter trip time, than a deployable Staehle-type sail. For a given area, the Staehle sail would also be significantly heavier (greater IMLEO). However, this must be balanced against the infrastructure requirement of a sail erection/fabrication facility in orbit. This facility would basically be a separate space station,[3] whose mass would have to be included in the IMLEO for the advanced sails.

2.2 ASSUMPTIONS

2.2.1 Solar Sail Size and Areal Density

Four types of solar sails are considered in this study. The first is a 4- km^2 area deployable sail with an areal density of 5 g/m^2 (i.e., a Staehle- type sail[2]). This sail does not require any on-orbit assembly facility, but it does require a fleet of orbit transfer vehicles (OTV) to transport the sails and their payloads to a 2000-km minimum operational altitude. Thus, although the sails are light (20 MT each), there is an added infrastructure (OTVs and their propellant) required to support operation of the sails. The second type of sail is also 4-km^2 in area, but it is erected and fabricated on-orbit in LEO. Its areal density is a factor of five lower than that of a Staehle sail, or 1 g/m^2 (i.e., a Garvey-type-sail[3]). This sail also requires a fleet of OTVs as above plus a LEO fabrication facility, discussed in detail below.

The last two sails are assumed to have areal densities of 0.2 g/m^2 (i.e., Drexler-type sails[5]), one with an area of 4 km^2, the other with an area of 10 km^2. As with the Garvey sails, the Drexler sails require both a LEO fabrication facility and an OTV infrastructure. Differences in the mission scenarios between the deployable Staehle-type sail and the advanced Garvey and Drexler sails which are fabricated on-orbit, are illustrated in Figs. 2-2 and 2-3. Table 2-1 summarizes the performance parameters of the four sails based on their type (deployable vs on-orbit fabrication), areal density, area, mass, and characteristic acceleration (A_c) at 1 AU assuming a 95 % reflectivity of the sail material.[3] (Note that areal density in units of g/m^2 is numerically the same in units of MT/km^2, and thrust in Newtons divided by mass in MT gives A_c in units of mm/s^2.)

2.2.2 Trajectory Analysis

2.2.2.1 <u>Solar Sail Trajectories.</u> The low-thrust solar sail heliocentric trajectories were analyzed by Carl G. Sauer Jr. of JPL.[6] The planetary escape and capture spirals were modeled after the method of Sands.[7] The usual free parameter is the characteristic acceleration (A_c) of the loaded sail for the Earth-to-Mars trip and the unloaded (empty) sail for the Mars-to-Earth trip. The characteristic acceleration of the sail is found by dividing the total thrust at 1 AU by the total mass of the sail with any payload (M_{PL}). The total thrust at 1 AU is:

$$\text{Thrust [N]} = (9\ \text{N/km}^2) \cdot (\text{Sail Area [km}^2]) \cdot (0.95\ [\text{Reflectivity}])$$

Table 2-1. Solar Sail Performance Parameters

Type	Staehle	Garvey	Drexler	Drexler
Construction	Deploy	Fabricate	Fabricate	Fabricate
Area (km^2)	4	4	4	10
Density (g/m^2)	5	1	0.2	0.2
Mass (MT)	20	4	0.8	2.0
Thrust (N)[a]	34.2	34.2	34.2	85.5
A_c (mm/s^2) [a]	1.71	8.55	42.75	42.75
Thrust/Weight [a]	0.17×10^{-3}	0.87×10^{-3}	4.36×10^{-3}	4.36×10^{-3}

Note: (a) Values at 1 AU assuming 95 % reflectivity

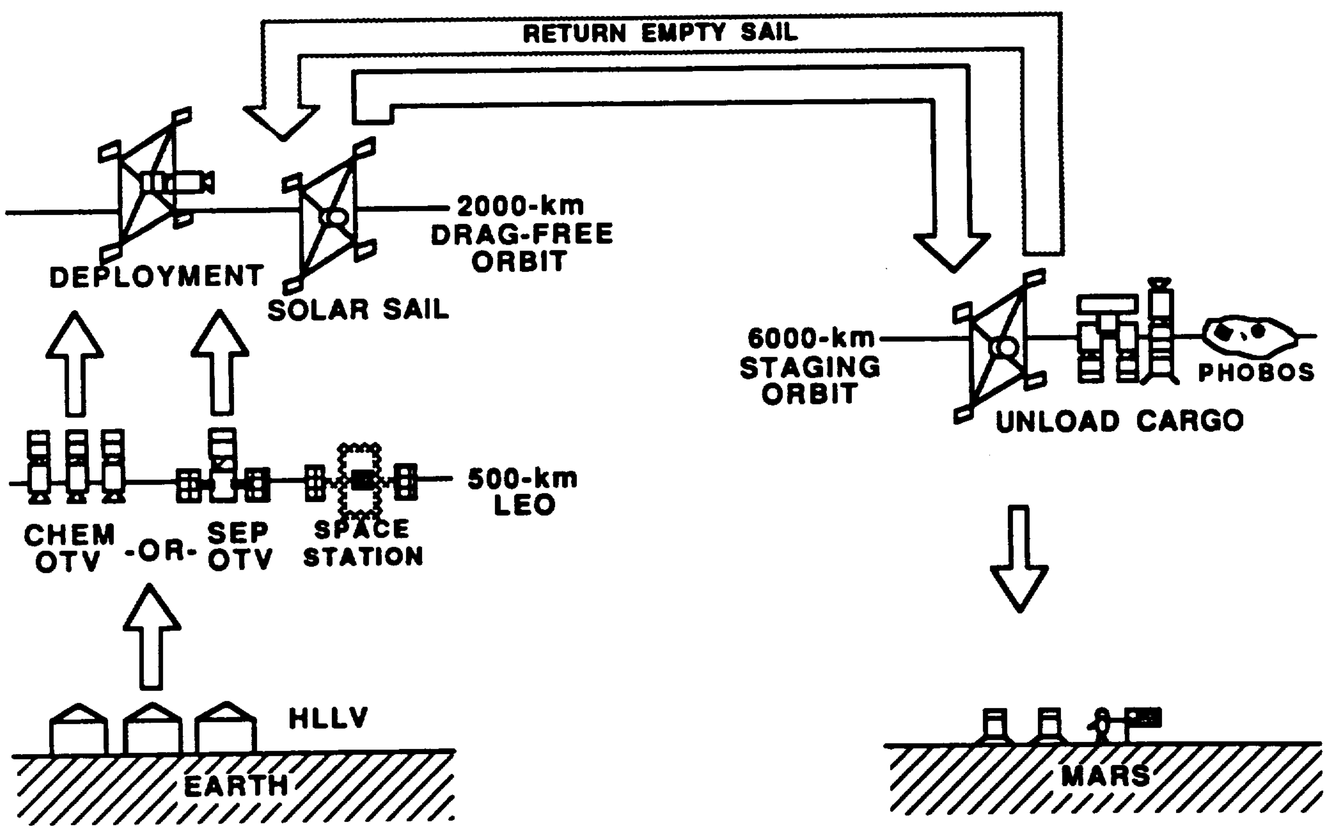

Figure 2-2. Solar Sail Mission Scenario

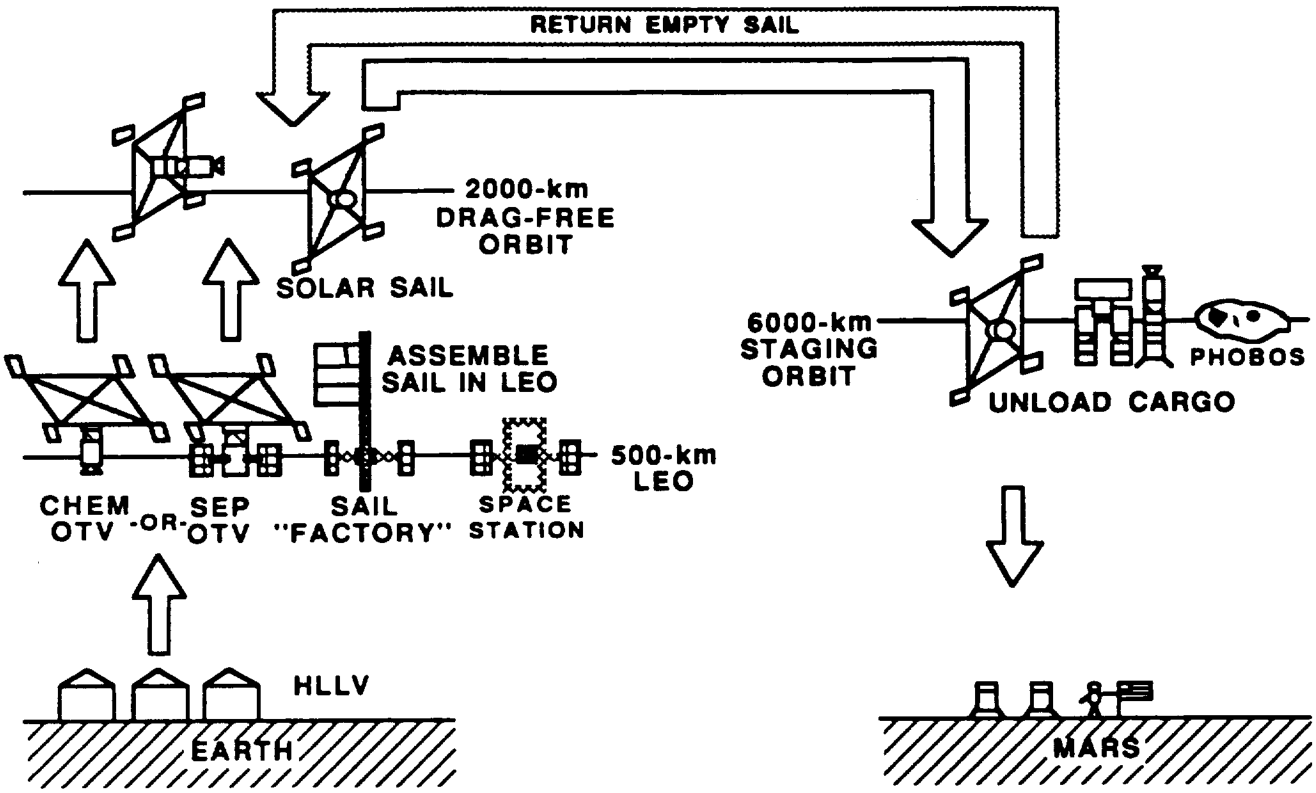

Figure 2-3. Advanced Solar Sail Mission Scenario

The sail mass (including payload, M_{PL}) is:

$$\text{Sail Mass [MT]} = (\text{ Areal Density [MT/km}^2]\ )\bullet(\text{ Sail Area [km}^2]\ ) + M_{PL}\text{[MT]}$$

Finally, the characteristic acceleration is:

$$A_c\ [\text{mm/s}^2] = (\text{ Thrust [N] }) / (\text{ Sail Mass [MT] })$$

Given A_c, the trajectory analysis computer code calculates departure and arrival dates as well as the corresponding trip times. These results are illustrated in Figs. 2-4 through 2-6. Note that values of A_c less than 0.6 mm/s^2 are not considered because this represents a lower limit for both the analysis codes as well as for maneuvering near a planet (e.g., not enough acceleration to turn the sail and re-orient it as it passes from shadow to light). This severely limits the total payload that can be placed on a sail. For example, the payload must be broken into 14 units (and OMVs and interstage adaptors attached) and placed on 14 sails for a 4-km^2 Staehle-type sail, 9 for a 4-km^2 Garvey- or Drexler-type sail, and 4 for a 10-km^2 Drexler-type sail.

Figure 2-4 illustrates the trip time for the Earth-to-Mars transfer as a function of A_c. For sails with low acceleration, the total time can exceed 5 years, although the high performance large Drexler-type sail can have a trip time approaching one year. Interestingly, the trip time for the heliocentric portion of the trip remains fairly constant with changes in A_c; the dominant impact is due to the Earth escape spiral time, which becomes very large for the slow, heavily-laden sails. The Mars-to-Earth trip time is shown in Fig. 2-5. In this case, the un-loaded sails have a significantly higher acceleration than the loaded sails so most of the trip time is due to the heliocentric transfer. However, the

sail's acceleration also impacts the arrival and departure dates at each planet, as shown in Fig. 2-6. Because of the interplanetary trajectory requirements, the fastest (i.e., least-loaded) Staehle-type sail actually arrives at Mars 90 days after it should have left Mars for the return to Earth. This of course does not mean that a Staehle-type sail cannot be used for the cargo mission; rather, it simply means that the return to Earth is along a non-optimum (i.e., non-minimum trip time) trajectory. Thus, for the Staehle-type sail, there is a "negative" layover time at Mars, whereas the other sails have a positive layover time ranging from about 100 to 260 days, as shown in Fig. 2-7.

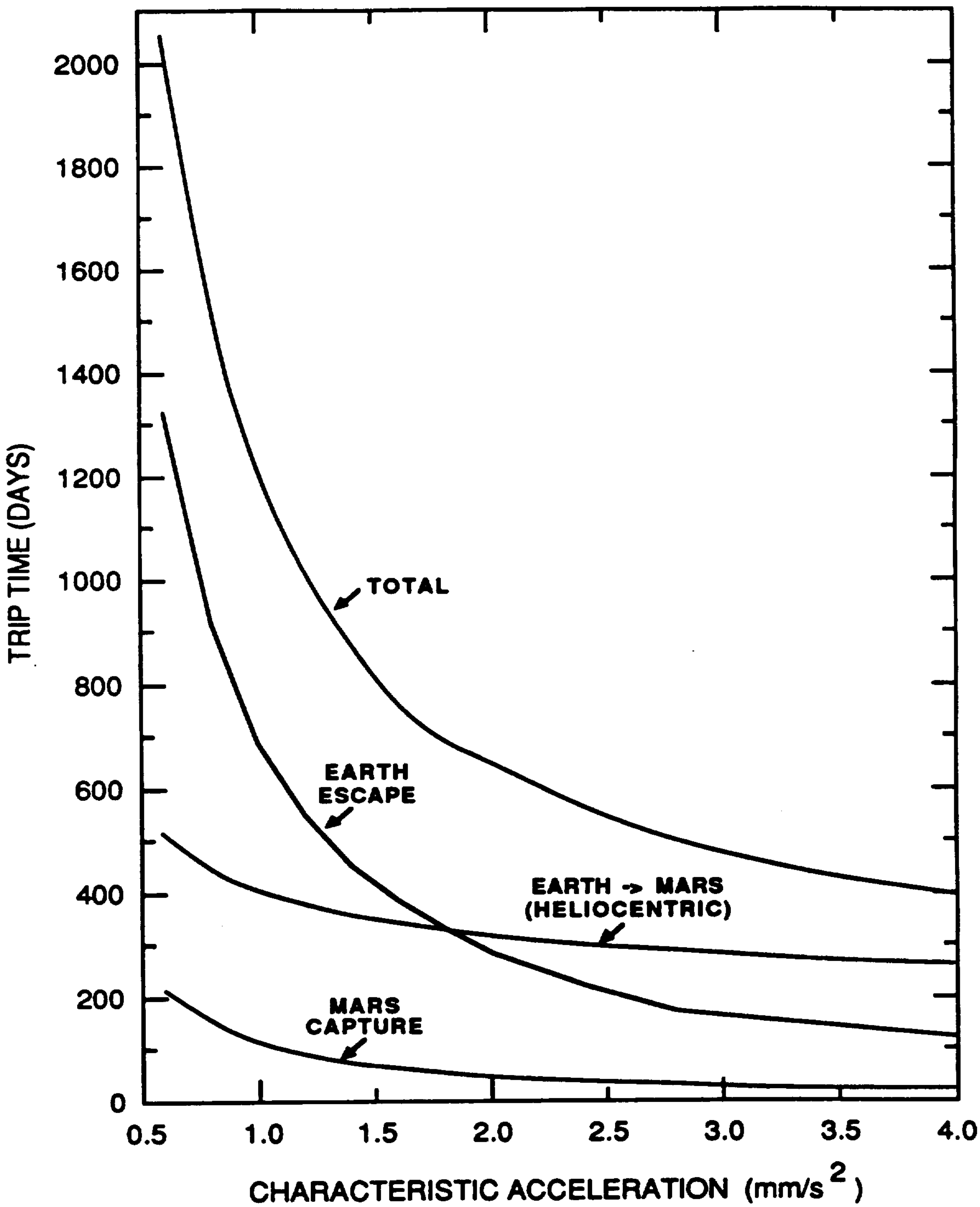

Figure 2-4. Solar Sail Earth-to-Mars Trip Time vs. Characteristic Acceleration

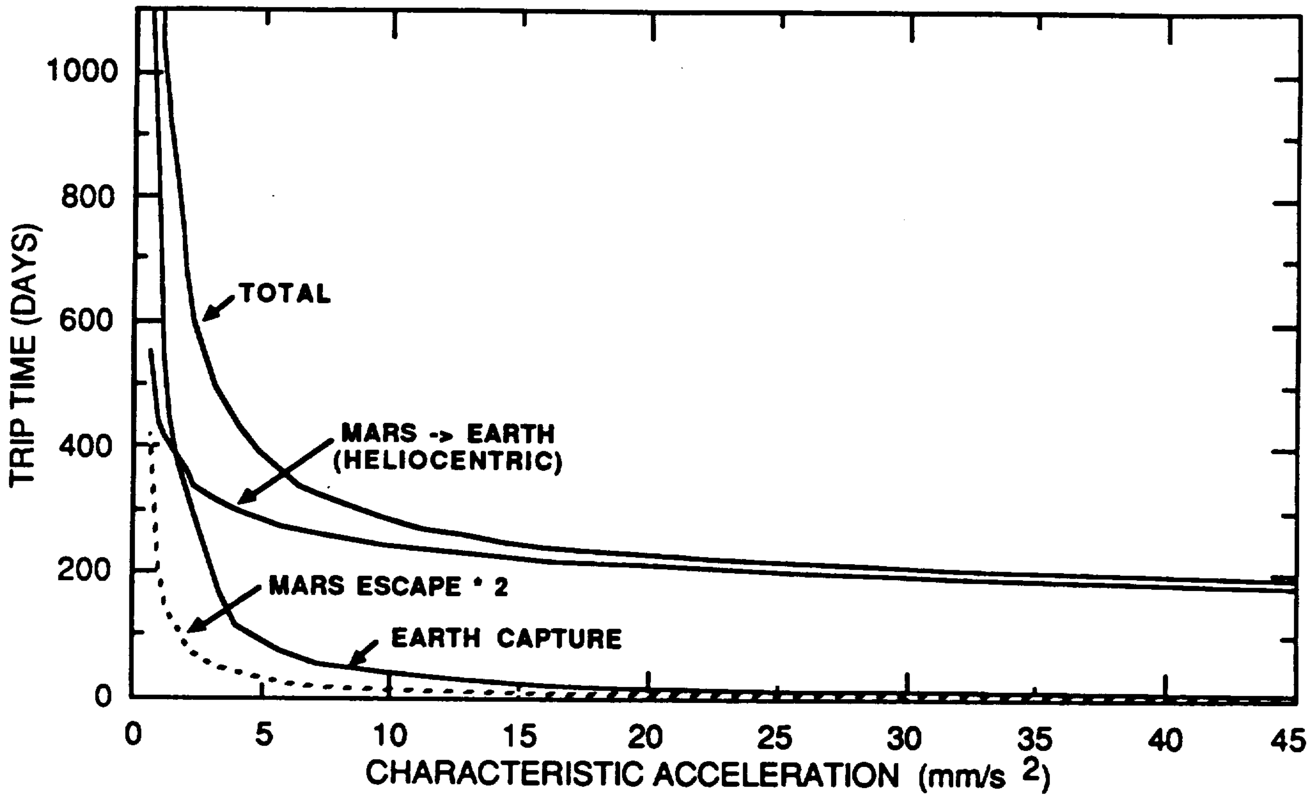

Figure 2-5. Solar Sail Mars-to-Earth Trip Time vs. Characteristic Acceleration

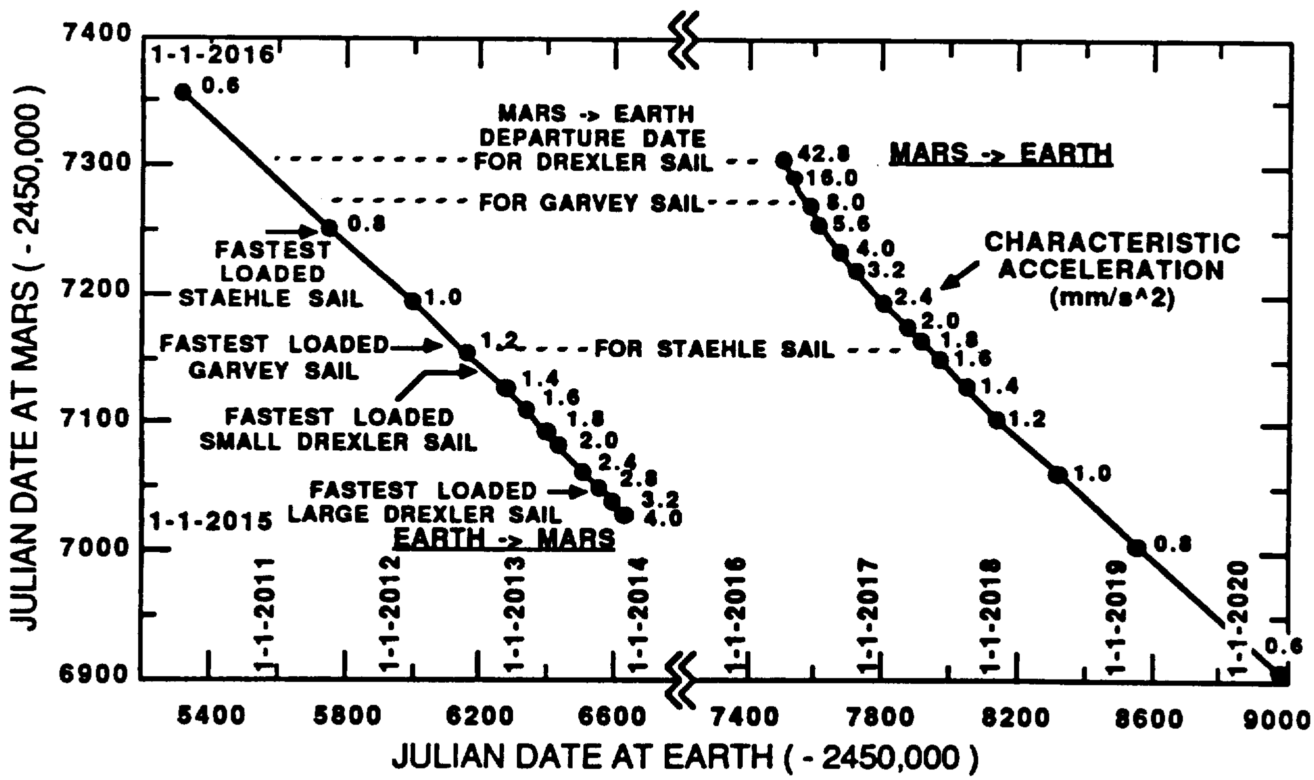

Figure 2-6. Solar Sail Departure and Arrival Dates vs. Characteristic Acceleration

2.2.2.2 Low-Thrust OTV Delta-V and Trip Times. A low thrust-to-weight OTV is required for orbit transfer of the advanced Garvey- and Drexler-type sails to prevent damage to the fragile sail support structure because they are already assembled in LEO. It is assumed that a similar low-thrust OTV requirement exists for the Staehle-type sail, although it could be transferred by a high-thrust OTV and unfurled in the 2000-km sail orbit. The low-thrust OTV Deta-V requirement was found by using the Edelbaum equation:[8]

$$\text{Delta-V} = \left(V_1^2 + V_2^2 - 2V_1V_2 \cos(\text{pi}/2 \cdot \text{Theta}) \right)^{1/2}$$

where V_1 and V_2 are the orbital velocities of the initial and final orbits and Theta is the difference in inclination between the initial and final orbits. Transfer from a 500-km altitude LEO to a 2000-km altitude sail deployment orbit with no plane change results in a Delta-V requirement of 715 m/s for a low-thrust OTV.

For low-thrust spiral trajectories, the total trip time can be found by assuming that the engines are operating continucusly, such that the total engine operating time is equal to the trip time. The engine operating time is found by dividing the mass of the total propellant (M_p) required for the transfer by the rate of propellant consumption (M-Dot):

$$\text{Run Time} = M_p \,/\, \text{M-Dot}$$

The propellant flow rate is found from the specific impulse (I_{sp}) and thrust:

$$\text{M-Dot [kg/s]} = \text{Thrust [N]} \,/\, (g_c \cdot I_{sp}\ [\text{lb}_f\text{-s/lb}_m])$$

where the factor $g_c = 9.8$ is used to convert I_{sp} in units of lb_f-s/lb_m to units of N-s/kg. Finally, thrust is found from the total jet power (P_{Jet}) and I_{sp}:

$$\text{Thrust [N]} = 2 \cdot P_{Jet}\ [\text{W}] \,/\, (g_c \cdot I_{sp}\ [\text{lb}_f\text{-s/lb}_m])$$

For the Solar Electric Propulsion (SEP) OTVs, it is also necessary to take into account the effect of shadowing on trip time because the SEP OTV coasts when in shadow. The fraction of time in shadow for each orbit is found by simple geometry; an average is found by summing over the orbital altitudes of the spiral transfer. For a transfer from a 500-km to a 2000-km altitude circular orbit, the OTVs spend an average of 68.1 % of the time in sunlight. Thus, the SEP OTVs have a total trip time that is 1/0.681 times that of the total engine operating time.

2.2.3 Support Infrastructure

2.2.3.1 OTV Infrastructure. For the OTV infrastructure, it was assumed that chemical oxygen/hydrogen (O_2/H_2) OTVs and 100-kW Solar Electric Propulsion (SEP) OTVs would be available as part of the infrastructure supporting operations between LEO and Geosynchronous Earth Orbit (GEO). Three types of OTVs were taken from recent LEO-GEO mission studies.[9] The first was a low-thrust O_2/H_2 OTV (not aerobraked) with an I_{sp} of 480 lb_f-s/lb_m; the second was a 100-kW (electric) hydrogen-arcjet SEP OTV with an I_{sp} of 1500 lb_f-s/lb_m; the third, and preferred OTV was again a 100-kW SEP OTV, but with a Xe-ion thruster with an I_{sp} of 4746 lb_f-s/lb_m. The dry mass and maximum propellant mass of each of these OTVs is listed in Table 2-2.

These three OTVs were selected to investigate the interplay between I_{sp} (and thus IMLEO) and tip time for the OTVs. Thus, the chemical OTV has the fastest trip time but uses the most propellant, while the Xe-ion SEP OTV uses the least propellant but has the longest trip time (656 days, including shadow coast periods, when fully loaded). It was assumed that an initial set-up time of 800 days (roughly the 2.2-year Earth-Mars synodic

period) would be available for sail fabrication (if required) and OTV placement of the sails and their payloads in the 2000-km operational orbit. As shown below, there will be a trade-off between the time available for sail fabrication and OTV operations which impacts the relative size of the fleet of fabrication facilities and OTVs. Also, multiple sails (typically 10 to 20 sails) are required to ensure that the fraction of the total payload carried by each sail is small enough so that acceleration of the loaded sail (with a 95% reflection efficiency[3]) is acceptable (>0.6 mm/s^2). For example, if ten 4-km^2 sails are to be fabricated on-orbit, two orbital factories are required because each can produce only 5 sails (one 4 km^2 sail every 140 days) in the allotted time, leaving 100 days (out of 800 total) for transfer of the sails to the 2000-km orbit. This is easily done by a single fully-loaded chemical OTV, but a fleet of partially loaded SEP OTVs is required to meet the required delivery schedule. (The payloads are transferred during the 700-day fabrication period). For example, for a system of ten 4-km^2 Garvey-type sails, a fleet of either 2 H_2-Arcjet OTVs or 3 Xe-Ion OTVs are required.

Finally, as with the baseline chemical O_2/H_2 system, a propellant resupply tanker was added to transport propellants to LEO. The resupply tanker tankage factor for the three OTVs was 0.032, 0.221, and 0.333 for the O_2/H_2, H_2-Arcjet, and Xe-Ion OTVs, respectively.[9]

Table 2-2. OTV Performance Parameters

Type	I_{sp}[a] (lb_f-s/lb_m)	M_{Dry}[a] (MT)	M_p(Max)[a] (MT)	P_{Jet}[a] (kW)	Trip Time[b] (Days)	M_{PL} Max[c] (MT)
O_2/H_2	480	7.160	37.000	10357	0.5	207.353
H_2-Arcjet	1500	7.485	15.032	49	563.3	286.251
Xe-Ion	4746	4.042	2.676	75	655.9	164.593

Notes: (a) Ref. 9
(b) Average fraction of time in sunlight per orbit = 68.1%
(c) Low-thrust Delta-V = 715 m/s (500-km LEO to 2000-km altitude circular orbit transfer)

2.2.3.2 LEO Fabrication Facility. The LEO fabrication facility was sized[3] to produce one 4-km^2 area sail in 140 days; it has a 25-kW photovoltaic power system, weighs 140 MT, and requires 1 MT of drag-makeup propellant resupply. Most of the drag is due to the solar panels becaust the sail is held edge-on to the orbital velocity vector to minimize its drag in LEO. Once assembled, the sail and its payload are boosted to a 2000-km altitude orbit for operation.

2.2.3.3 Payload Support. It was assumed that the large gossamer sails could not dock directly with a base in Earth or Mars orbit, so an Orbital Maneuvering Vehicle (OMV) and associated structural adaptors were added to each payload so as to provide a 50-m/s Delta-V capability in Earth and Mars orbit for any necessary rendezvous and docking of the payloads, as was shown in Fig. 1-4.

2.2.4 Atmospheric Drag

Interestingly, the year 2014 is a maximum in the solar cycle and atmospheric drag

will be at a maximum. A 2000-km altitude Earth orbit was found to give acceptably low drag (e.g., drag = 0.2 % of the sunlight force) for the sails even under the extreme worst-case solar maximum atmosphere.

2.3 RESULTS

Figures 2-7 and 2-8 illustrate the IMLEO and trip time for the four solar sail concepts. Figure 2-7 shows a mass breakdown of the various elements in the system (payload, payload OMVs, sails, sail fabrication facilities, and OTVs) for a system consisting of 20 solar sails, each with 20 MT of payload per sail (effectively 26 MT with the addition of the OMV). A solar sail system is very light, being roughly equal in mass to the mass of the (net) payload transported. For the advanced sails, the IMLEO of the total system is one-half that of the baseline O_2/H_2 system. The major disadvantage of a solar sail system is its long trip time, ranging from 500 to 2300 days, depending on sail concept.

An interesting trade-off occurs between deployable sails (Staehle) and sails fabricated on-orbit (Garvey and Drexler). In this case, the weight of a fleet of "thick" deployable sails is significantly greater than the weight of a fleet of sails fabricated on-orbit, even when the fabrication facility infrastructure is included. Trip times are also significantly shorter for the lighter sails, as shown in Fig. 2-8. Finally, the OTV deployment system is only a small fraction of the total; any of the chemical or SEP OTVs could be used to deploy the sails to 2000-km altitude.

2.4 CONCLUSIONS

Although technologically more demanding, it appears that there are significant benefits resulting from using sails erected or fabricated on-orbit. A reasonable goal, in terms of technological challenge and performance benefit, is the Garvey-type sail with an areal density of 1 g/m^2. This sail has good performance with its thin sail sheet, yet inherits much of the structure and dynamics control technology developed for the Halley Comet mission. For example, the IMLEO of a 20-sail fleet of Garvey-type sails is only 727 MT with an Earth-to-Mars trip time of 1060 days; this represents only 44% of the IMLEO and 3.6 times the trip time of the baseline O_2/H_2 system.

The major disadvantage of the solar sail system is its long trip time. For the 20 Garvey-type sail fleet described above, 584 days are spent spiraling out from Earth, 384 days in heliocentric transfer, and 92 days for spiral in to Mars orbit. Future studies should consider methods to reduce the trip time at the expense of a higher IMLEO. For example, the payload could be aerobraked into Mars orbit with a chemical propulsion system to avoid the 92-day Mars-capture spiral. Similarly, a chemical or electric stage could inject the sail and payload out of Earth orbit and avoid (or at least partially reduce) the long Earth-escape spiral. This may have significant impact on trip time, but also on mass, because the Earth-escape portion comprises the largest fraction of the total trip time (and Delta-V).

In summary, few other concepts have the potential for providing such significant reduction in IMLEO as the solar sail; detailed studies will be required to properly assess the technology requirements and mission benefits of solar sails as compared to other options for the Mars cargo mission.

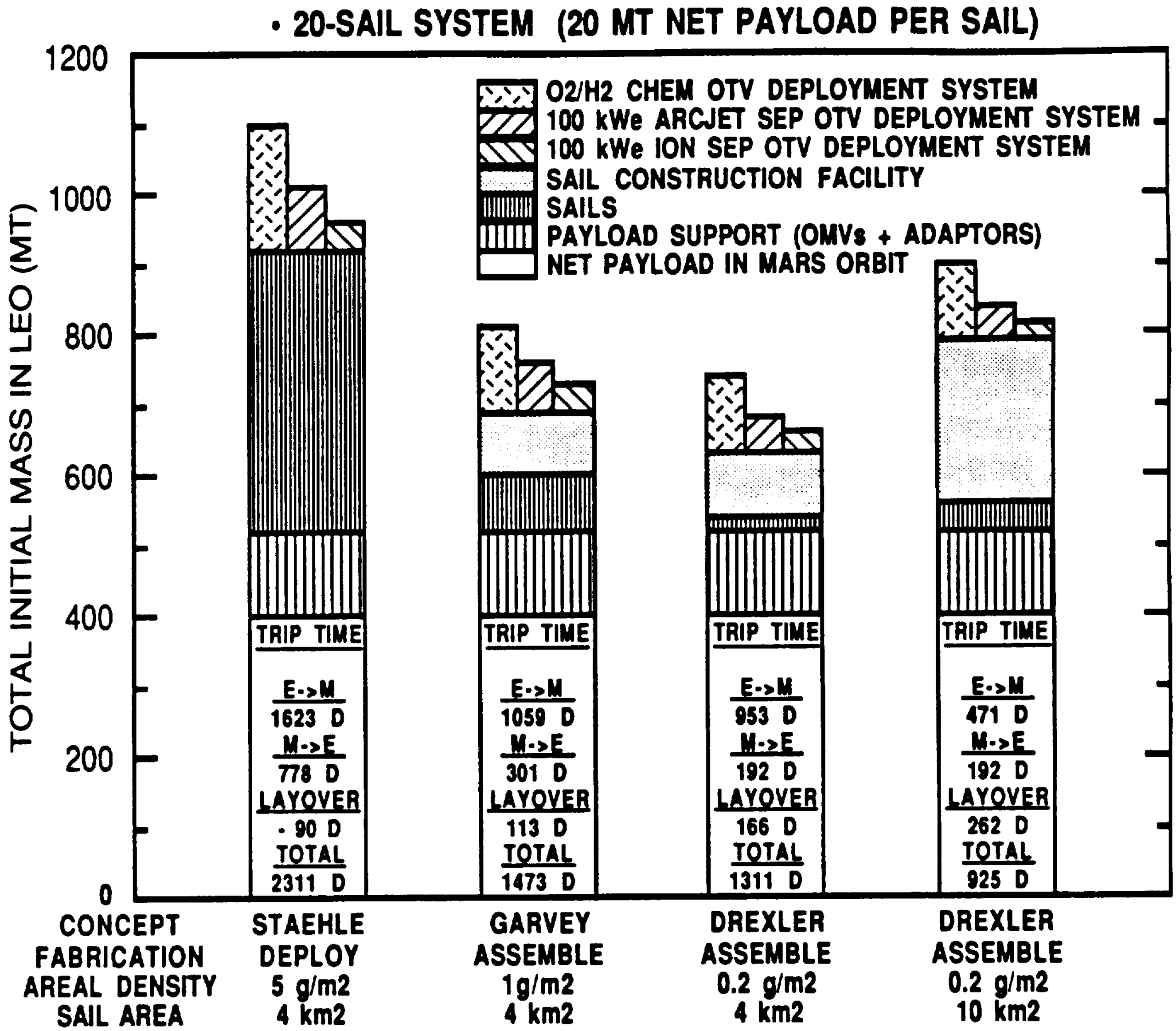

Figure 2-7. Mass Breakdown for Elements in a 20-Sail System

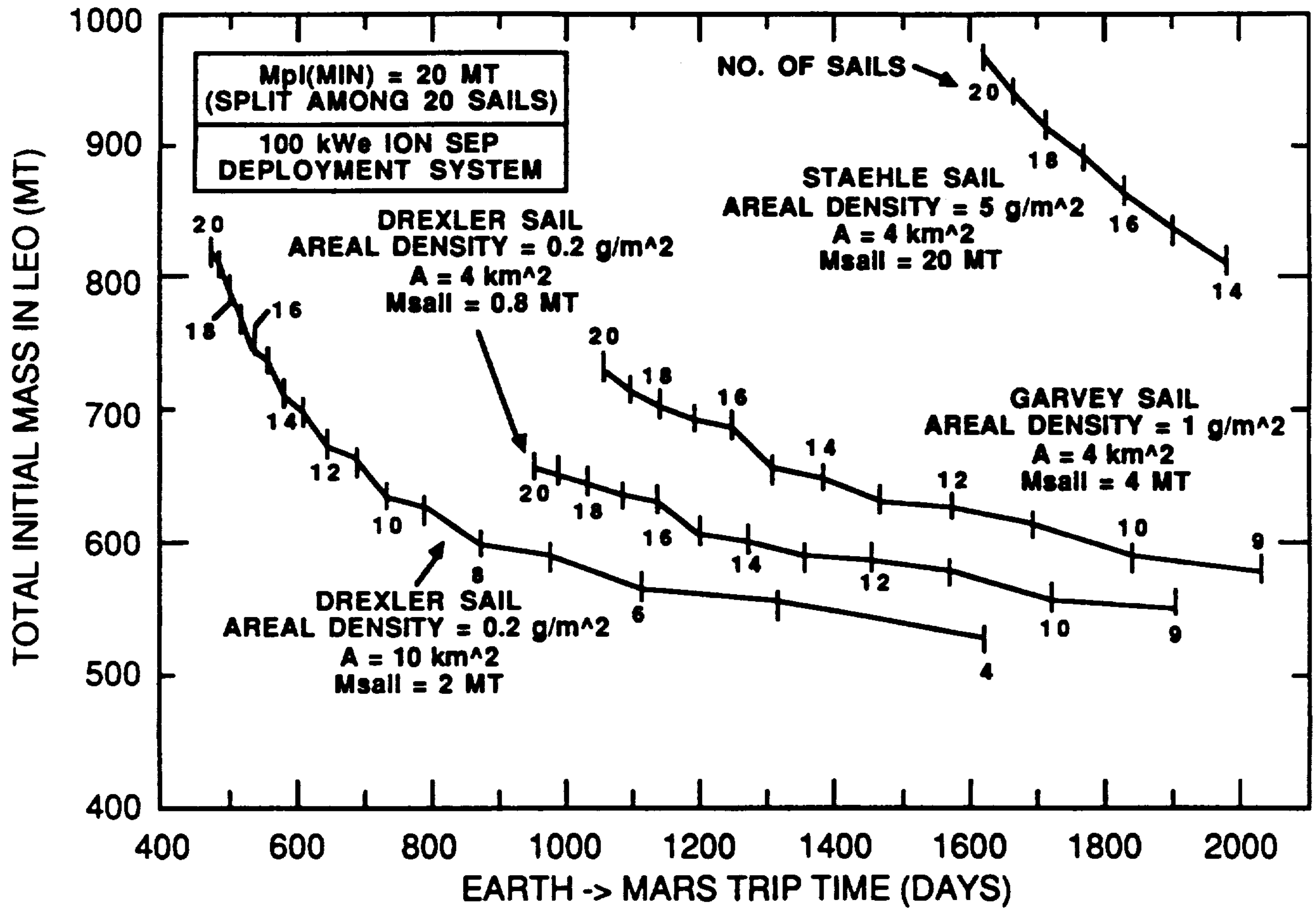

Figure 2-8. Initial Mass in Low Earth Orbit vs. Earth-to-Mars Trip Time for Solar Sail Propulsion

2.5 REFERENCES

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SECTION 3

100-MW CLASS SOLAR ELECTRIC PROPULSION

3.1 INTRODUCTION

A Solar Electric Propulsion (SEP) system, as shown in Fig. 3-1, consists of a solar photovoltaic power supply, a power processor unit (PPU) which converts the solar array power output to the form required by the thrusters, and the electric thrusters. In this study, a 100-MW class SEP system was analyzed. A similar-sized Nuclear Electric Propulsion (NEP) system is described in the next Section.

Previous studies[1] have shown significant benefits for the Mars cargo mission utilizing NEP systems with a total (power and propulsion) specific mass of 10 kg/kW, an I_{sp} of 5000 lb_f-s/lb_m, and a power level of 1 to 10 MW electric (4 MW typical). This SEP study (and the NEP study described in the next Section) was aimed at investigating ultra-high power SEP (and NEP).

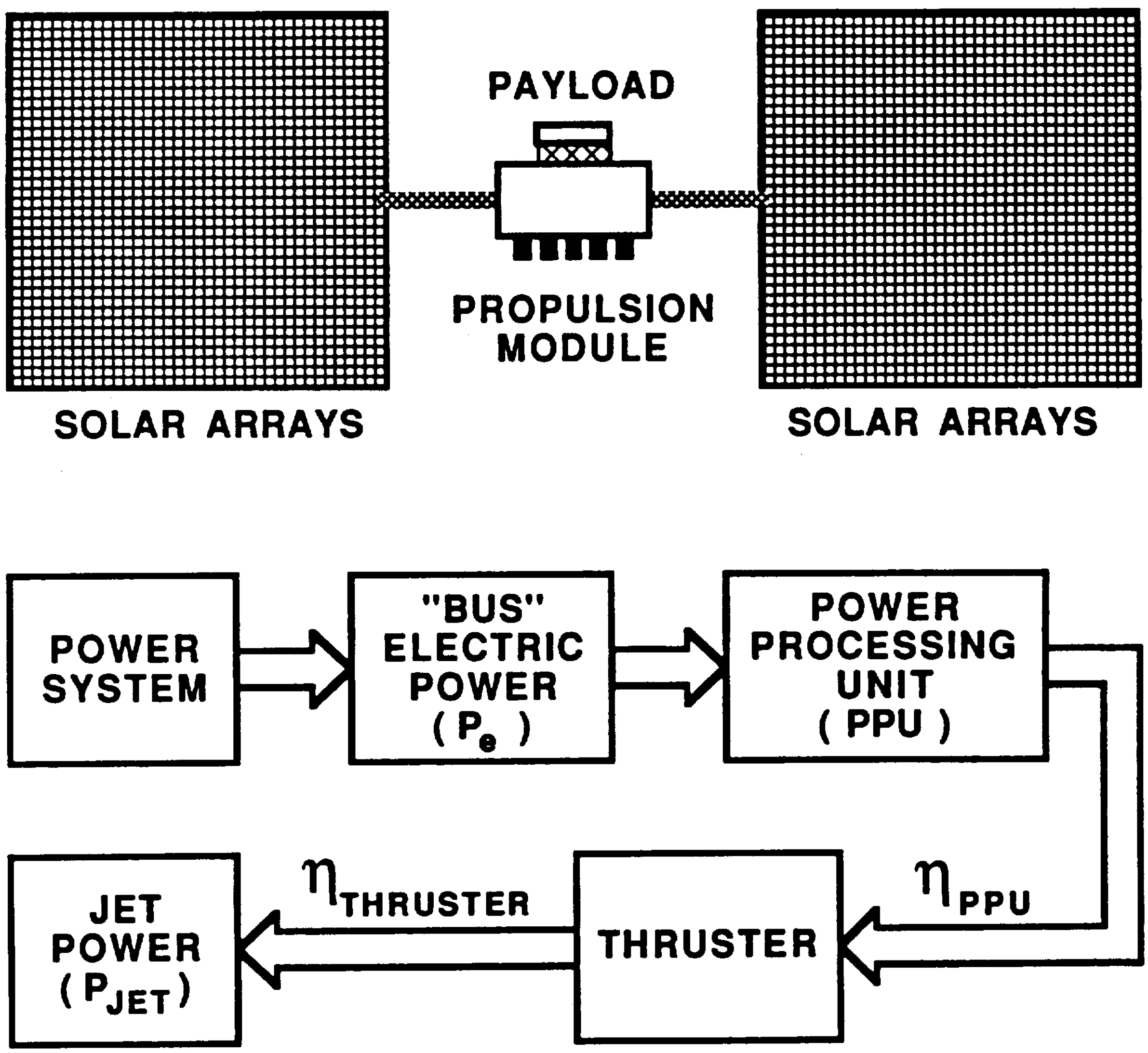

Figure 3-1. Solar Electric Propulsion (SEP) Concept

Note that in discussing SEP (and NEP) concepts, it is the "bus" electric power (P_e) that is quoted; this is the (average) power output from the solar arrays (or nuclear reactor). As shown in Fig. 3-1, the "bus" electric power is then fed to the power processor unit (PPU) and from there to the thruster. There are losses and inefficiencies in the PPU and thrusters, such that the propulsion or jet power (P_{Jet}) is typically 50 to 90 % of the input "bus" electric power.

From an operational point of view, a SEP vehicle has an advantage over a NEP vehicle in that the SEP vehicle can operate from LEO; by contrast, a NEP has a minimum operational altitude of about 1000 km to ensure that no radioactive components enter the Earth's biosphere in case of catastrophic failure of the NEP vehicle. However, the SEP vehicle suffers from shadowing in Earth or Mars orbit, resulting in a longer trip time than a NEP vehicle which has a continuous power source. Similarly, power output from the solar array drops off as the vehicle moves away from the sun. However, the efficiency (sunlight-to-electricity) of the solar array increases with decreasing temperature. Thus, the power output from a solar array drops off more slowly than a $1/R^2$ distance from the sun, as shown in Fig. 3-2.

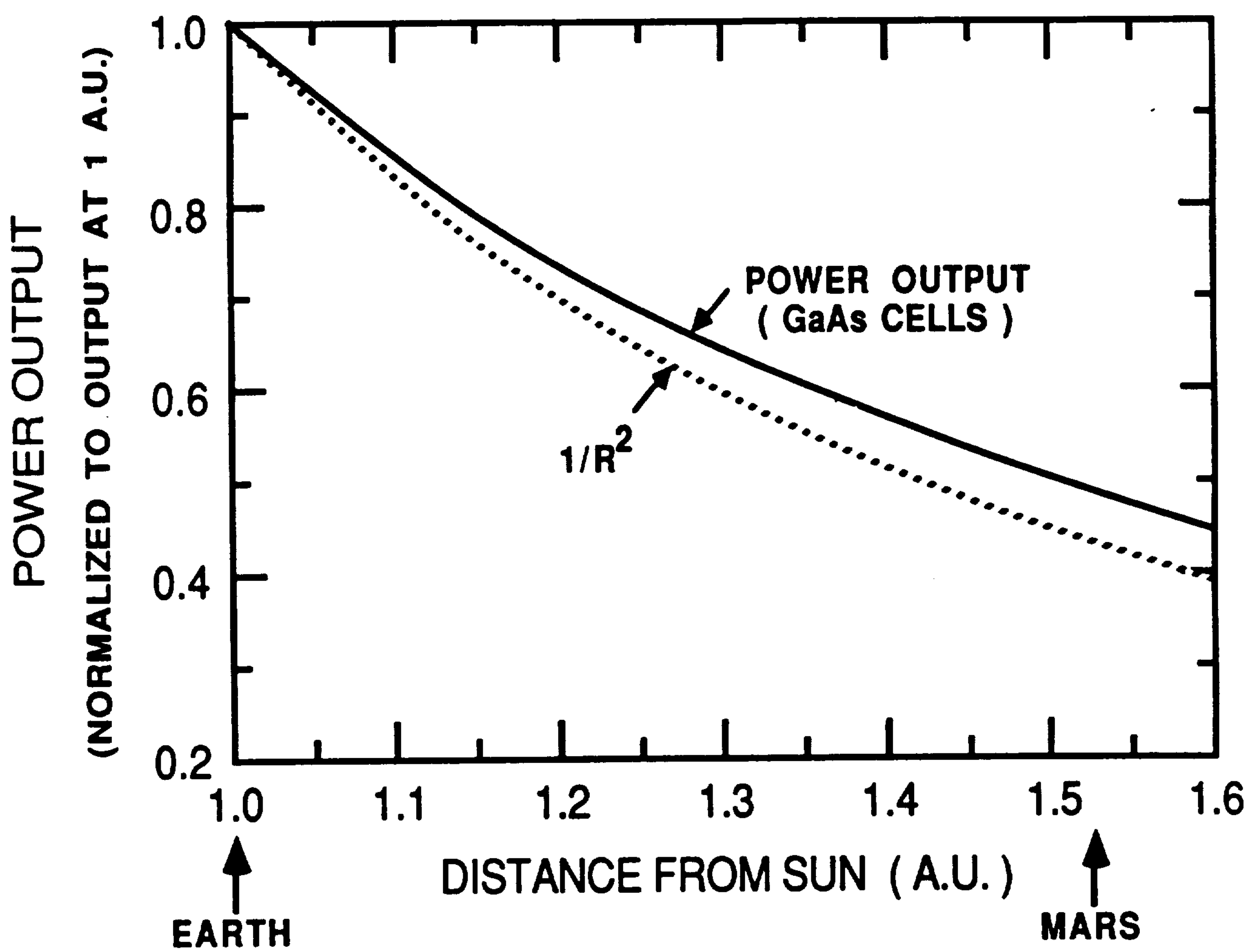

Figure 3-2. Solar Photovoltaic Array Power Output vs. Distance from the Sun

3.2 ASSUMPTIONS

3.2.1 Mission Scenario

The SEP vehicle leaves from LEO, so no supporting OTV infrastructure is needed. Power levels from 100 to 500 MW_e are considered, as is the option of dividing the payload among several vehicles flying in parallel. The mission scenario is illustrated in Fig. 3-3.

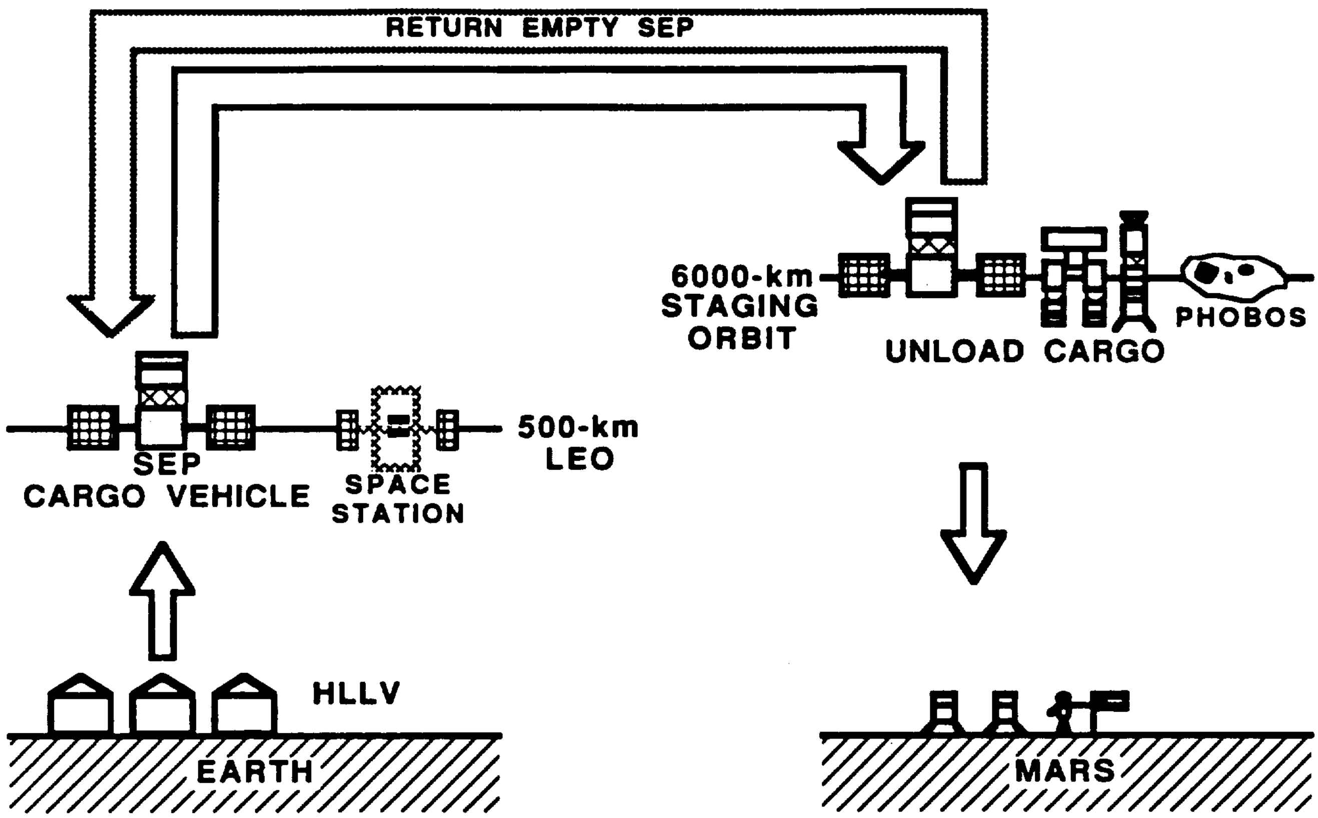

Figure 3-3. 100-MW Class Solar Electric Propulsion Mission Scenario

3.2.2 100-MW_e SEP Vehicle Sizing

The two primary figures of merit for the various elements in an electric propulsion systems are their specific mass, expressed in units of kilograms per kilowatt of electric power (also, a $kg/kW_e = MT/MW_e$), and their efficiency, expressed as the ratio of power input divided by power output. Specific mass and efficiency are often abbreviated as alpha (α) and eta (η), respectively.

3.2.2.1 <u>Solar Array Specific Mass.</u> Multi-megawatt solar arrays were evaluated by Paul M. Stella of JPL.[2] It was assumed that by 2014, the efficiency of the arrays would be 25 to 27 %, or about twice the state-of-the-art, as shown in Fig. 3-4. At an assumed efficiency of 25 %, 3.6 m^2 are required per kilowatt of electric power (with 17.7 % distribution losses). The blanket (cells plus wiring) areal density was estimated[2] to be 0.45 kg/m^2. Addition of a supporting structure would raise this figure to 1.0 kg/m^2 with an uncertainty of ± 20 %.[2] Thus, the overall specific mass of the solar array is 3.6 kg/kW_e. The mass of the array is

then:

$$\text{Solar Array Mass [kg]} = (3.6\ [\text{kg/kW}_e]) \cdot P_e\ [\text{kW}_e]$$

$$\text{Solar Array Mass [MT]} = (3.6\ [\text{MT/MW}_e]) \cdot P_e\ [\text{MW}_e]$$

where P_e is the "bus" electric power and the value of the specific mass in kg/kW$_e$ is numerically equal to MT/MW$_e$.

This solar array specific mass value is baselined for systems between 1 MW$_e$ and 500 MW$_e$. Finally, it was assumed that there would be no loss in performance due to radiation (Van Allen belts or solar flare) degradation of the cells. This is based on two assumptions; first, trip time through the Earth's Van Allen radiation belt would be greatly reduced for a high-powered SEP vehicle, and second, technology improvements by 2014 would provide more radiation-resistant cells than are available today.[3]

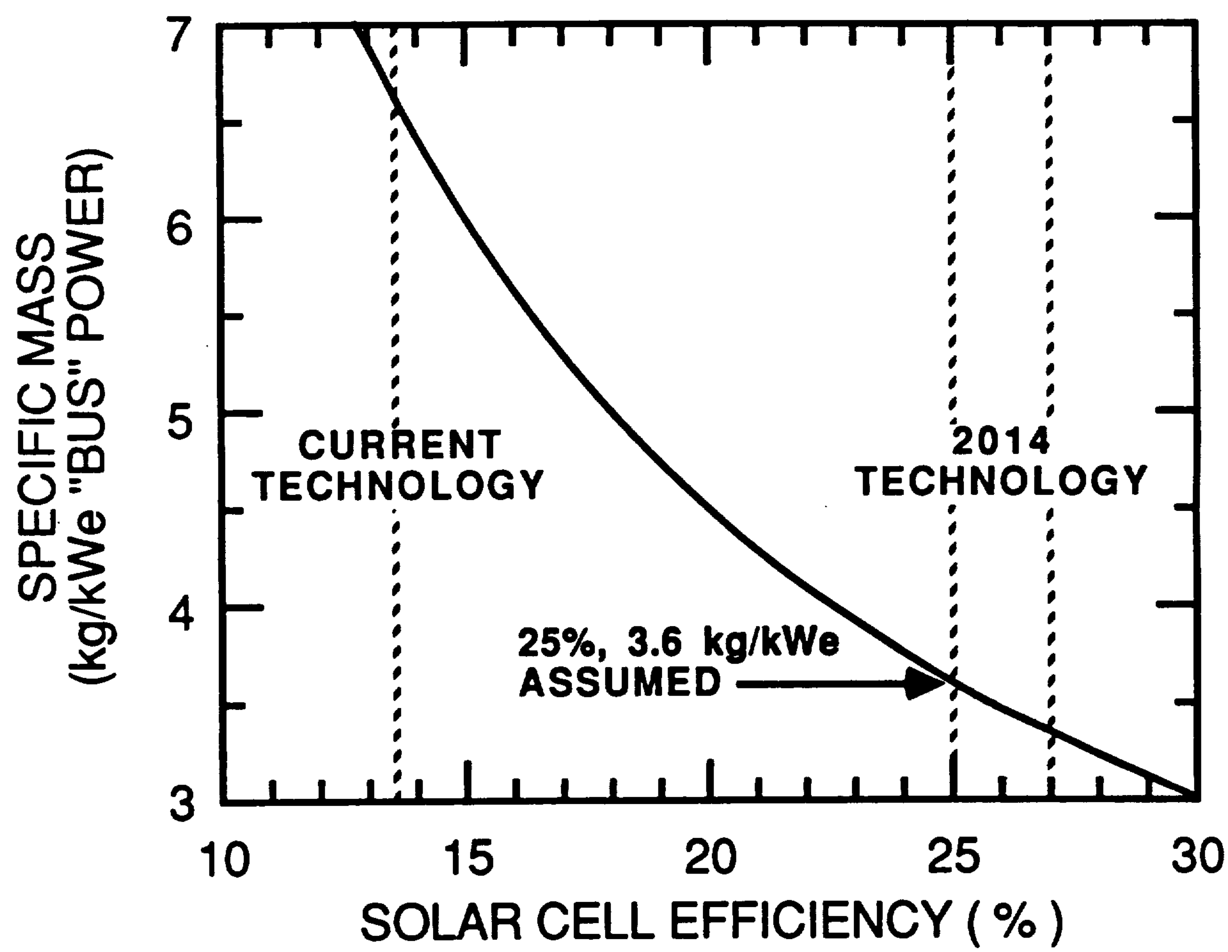

Figure 3-4. Advanced Solar Array Specific Mass vs. Efficiency

3.2.2.2 Power Processor Unit (PPU) Specific Mass and Efficiency. The high-power PPU systems were evaluated by Stanley Krauthmer of JPL[4] and were estimated to have a specific mass of 0.45 kg/kW$_e$ with an electrical efficiency of 97 %. This is lower performance (higher specific mass, lower efficiency) than the PPU for the NEP system

described in the next Section because the output from the solar arrays is limited to 500 V and must be converted to a high voltage for input to the ion engines (discussed below); by contrast, the high-voltage output from the nuclear reactor dynamic power conversion system gives a better match to the needs of the ion engines, so less voltage manipulation is needed in the NEP case. Thus, the mass of the PPU is simply:

$$\text{PPU Mass [kg]} = (0.45\ [\text{kg/kW}_e]) \cdot P_e\ [\text{kW}_e]$$

$$\text{PPU Mass [MT]} = (0.45\ [\text{MT/MW}_e]) \cdot P_e\ [\text{MW}_e]$$

3.2.2.3 Ion Thruster Specific Mass and Efficiency. High-powered advanced ion engines for a 100-MW_e class SEP (and NEP) system were evaluated in earlier studies;[5] details of the sizing of the electron-bombardment ion engines used in this analysis are given in the Appendix. Both ion and magneto-plasma-dynamic (MPD) thrusters are candidates for SEP (and NEP) vehicles. In this study, ion thrusters were selected over MPD thrusters because of the potentially higher specific impulse (I_{sp}), efficiency, and lifetime of ion thrusters, although the two types of thrusters have different strengths and weaknesses. For example, MPD thrusters may provide significant advantages for high-power vehicles because the power-per-thruster of ion thrusters is typically limited to a few MW_e. By contrast, MPD thrusters do not begin to operate efficiently until power levels of one MW_e or higher are reached. Thus, for a 100 MW_e vehicle, the number of MPD thrusters required might be an order-of-magnitude less than the number of ion thrusters, with corresponding savings in system complexity (PPUs, gimbals, feed systems, etc.).

The specific mass and efficiency of the high-performance ion thrusters depend on the specific impulse (I_{sp}) at which they operate, as illustrated in Figs. 3-5 and 3-6. For example, at an I_{sp} of 5000 lb_f-s/lb_m, the thruster has a specific mass of 0.38 kg/kW_e and an efficiency (electric-to-jet) of 79 %; at an I_{sp} of 10,000 lb_f-s/lb_m, the corresponding values are specific mass and efficiency are 0.10 kg/kW_e and 85 %; and at an I_{sp} of 20,000 lb_f- s/lb_m, 0.025 kg/kW_e and 85 %. The scaling equation describing thruster specific mass and efficiency are:[5]

$$\text{Thruster Specific Mass [kg/kW}_e\text{=MT/MW}_e] = 1 / \{ 10^{-7} \cdot (I_{sp}\ [lb_f\text{-s}/lb_m])^2 + 0.163 \}$$

$$\text{Thruster Efficiency} = 0.85 / \{ 1 + 1.8\text{x}10^6 / (I_{sp}\ [lb_f\text{-s}/lb_m])^2 \}$$

For the thruster mass, the power (kW_e) is that power entering the thruster (i.e., leaving the PPU), or $0.97 \cdot P_e$ assuming a 97-% efficient PPU for the SEP system:

$$\text{Thruster Mass [kg]} = (\text{Specific Mass [kg/kW}_e]) \cdot (0.97 \cdot P_e\ [\text{kW}_e])$$

$$\text{Thruster Mass [MT]} = (\text{Specific Mass [MT/MW}_e]) \cdot (0.97 \cdot P_e\ [\text{MW}_e])$$

In sizing the vehicles, it is also necessary to consider thruster lifetime, which is assumed to be 15,000 hours. Additional thrusters are added depending on the total "burn" time required for the mission.

Finally, it should be noted that these scaling equations are based on the use of mercury propellant in the ion thrusters. Mercury propellant was used extensively in earlier ion engine technology programs. However, mercury ion thruster exhaust is generally considered undesirable from a toxicity and spacecraft contamination perspective because it is relatively non-volatile and would coat the vehicle surfaces. Partly for these reasons, much of the recent ion thruster work has focused on the use of volatile inert gasses such as xenon and argon. Interestingly, the high cost and limited production rate of xenon may preclude its use in large electric propulsion vehicles.[6] Also, tankage for the inert gasses,

as either liquids or supercritical gasses, will be high (e.g., a 0.333 tankage factor for supercritical xenon storage).[7] Finally, the contamination effects of high-I_{sp} ion thruster exhaust plumes may make the use of mercury propellant acceptable if the high exhaust velocity (i.e., an I_{sp} of 20,000 lb_f-s/lb_m corresponds to an exhaust velocity of 200 km/s) minimizes plume back-flow. These issues of thruster performance versus contamination effects should be addressed in future studies.

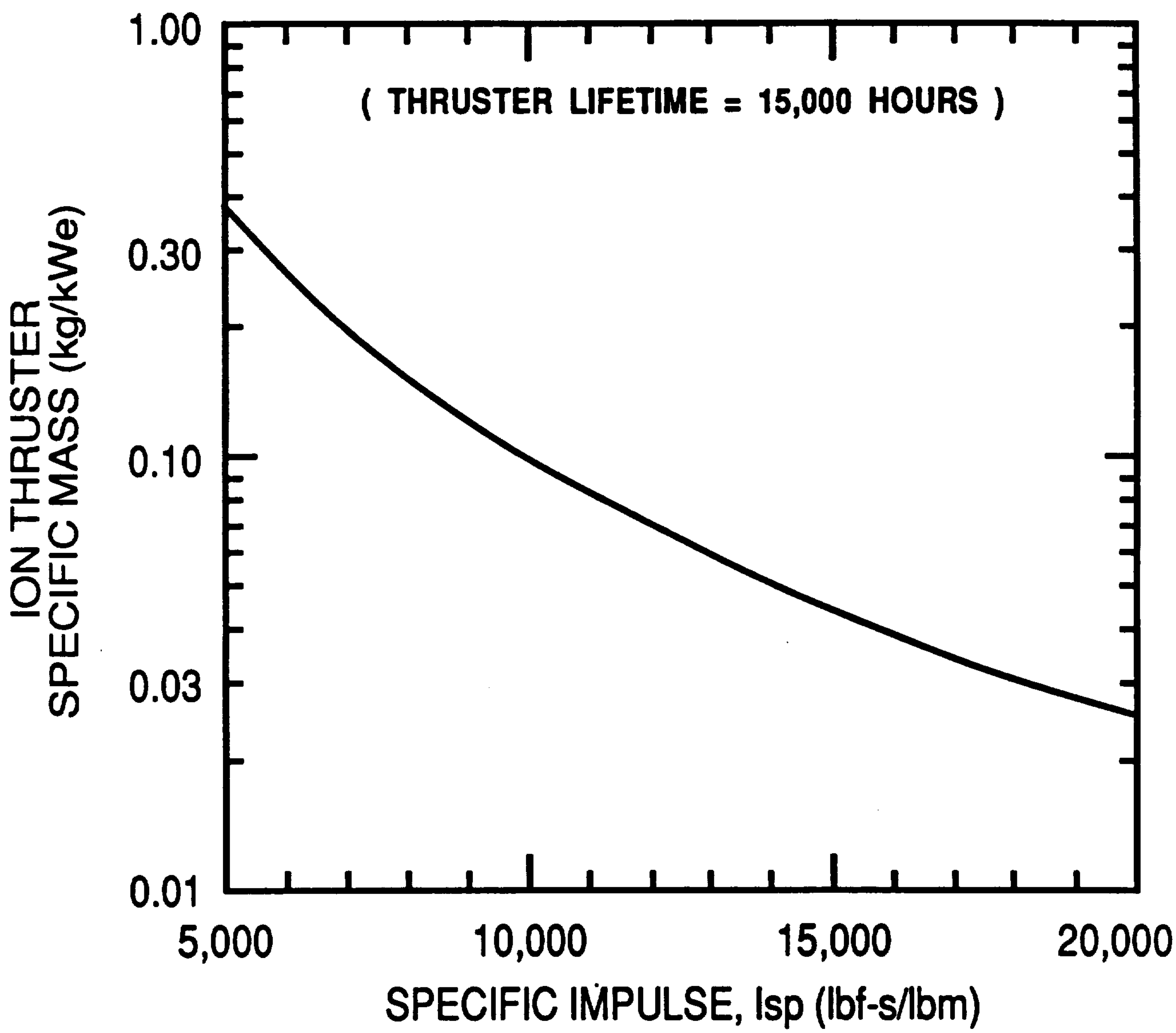

Figure 3-5. Advanced Ion Thruster Specific Mass vs. Specific Impulse

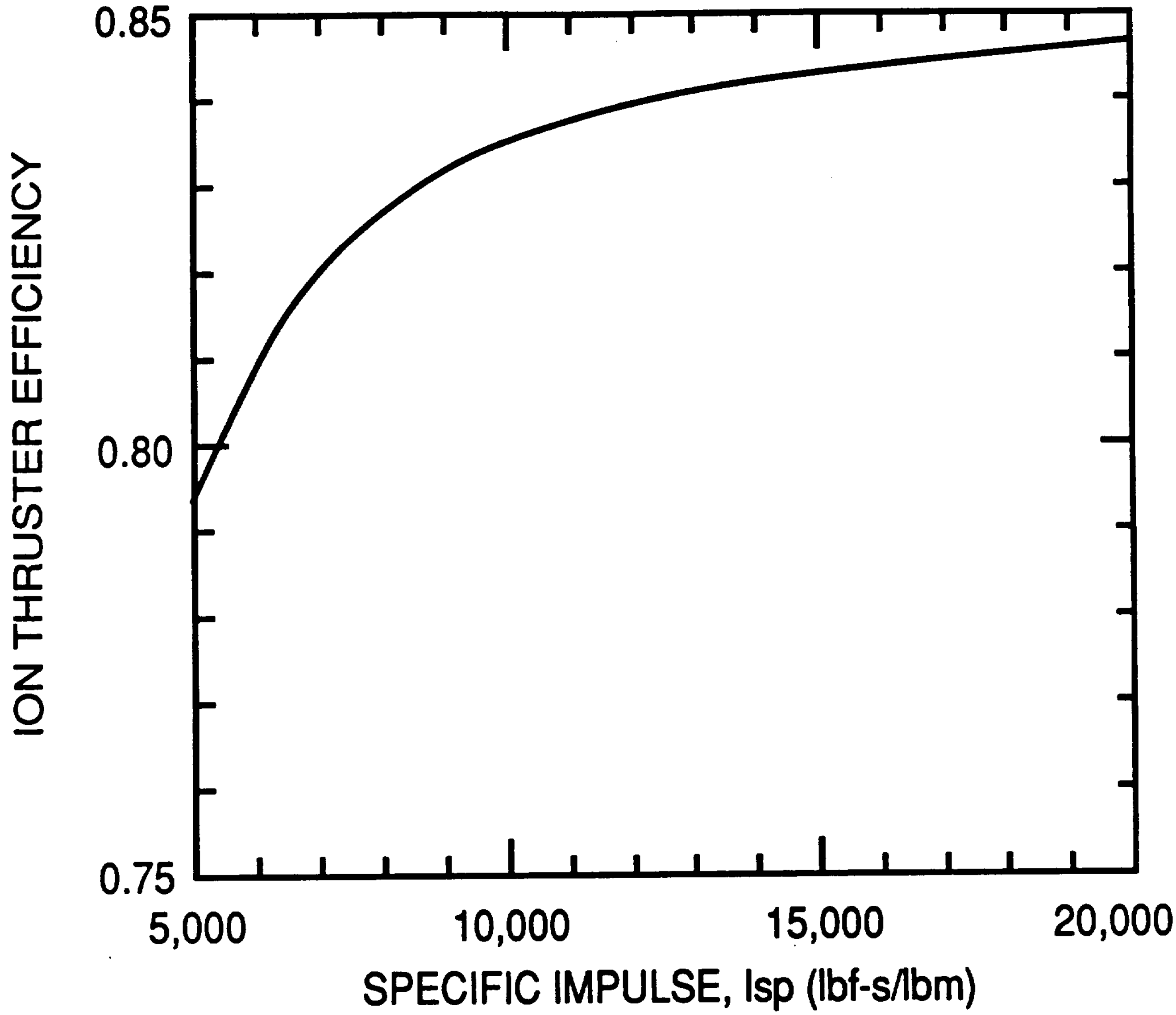

Figure 3-6. Advanced Ion Thruster Efficiency vs. Specific Impulse

3.2.2.4 Final Vehicle Sizing. The overall vehicle sizing including the mercury propellant tankage[5] and the elements listed above (solar arrays, PPUs, and thrusters) is:

$$M_{Dry}\ \text{SEP Vehicle [kg]} = \alpha_{\text{Solar Array}} \cdot P_e + \alpha_{PPU} \cdot P_e + \alpha_{\text{Thrusters}} \cdot P_e \cdot \eta_{PPU} + 1.42 \cdot M_p[\text{kg}]^{2/3}$$

$$M_{Dry}\ \text{SEP Vehicle [MT]} = \alpha_{\text{Solar Array}} \cdot P_e + \alpha_{PPU} \cdot P_e + \alpha_{\text{Thrusters}} \cdot P_e \cdot \eta_{PPU} + 0.142 \cdot M_p[\text{MT}]^{2/3}$$

where the propellant mass (M_p), specific mass (α), and "bus" electric power (P_e) are in the appropriate units and the thruster mass is a function of the PPU efficiency (η).

The total propulsive ("jet") power depends on the total "bus" power (P_e) and the efficiencies of the PPU and thruster (η_{PPU} and $\eta_{Thruster}$, respectively):

$$P_{Jet} = P_e \cdot \eta_{PPU} \cdot \eta_{Thruster}$$

Finally, a propellant resupply tanker (tankage factor = 2 %) is also included in the total initial system mass in low Earth orbit.

3.2.3 Support Infrastructure

As illustrated in Fig. 3-3, the SEP vehicle leaves LEO and travels to Phobos orbit, the payload is released, and the SEP returns to LEO. Because the 100-MW class SEP vehicle is a large structure (0.36 km^2 at 100 MW_e), an orbital maneuvering unit (OMV) is added to each payload to provide for rendezvous and docking in Earth and Mars orbit, as was done in the case of the Solar Sail payloads.

3.2.4 Trajectory Analysis

3.2.4.1 Low-Thrust Trajectory Analysis. The low-thrust SEP (and NEP) trajectories were provided by Carl G. Sauer Jr. of JPL.[8] The results for the various low-thrust concepts (e.g., solar-electric, nuclear-electric, and solar-thermal propulsion) evaluated in this study are listed in the Appendix. The trajectory analysis code has as its input the I_{sp} and initial ("wet") mass (M_0) of the vehicle; the outputs include the final ("dry") mass (M_B) and trip times for the various portions of the trajectory (escape spiral, heliocentric transfer, and capture spiral). The vehicle masses are "normalized" in terms of the propulsive ("jet") power (P_{Jet}), i.e., M_0/P_{Jet} and M_B/P_{Jet}. When using the data in the Appendix, P_{Jet} is based on the power availabe at Earth (1 AU) even for solar-powered concepts; the trajectory analysis code takes into account variations in power with changes in distance from the sun.

3.2.4.2 Mission Analysis. In determining the total system initial mass in low Earth orbit (IMLEO) and trip time, it is necessary to "work backwards" from the final empty vehicle weight in LEO (after returning from Mars) to the initial fully loaded vehicle mass in LEO (prior to the Earth-to-Mars transfer). This involves first calculating the values of M_B/P_{Jet} and then M_0/P_{Jet} for the Mars-to-Earth trajectory (with no payload), and then calculating M_B/P_{Jet} and M_0/P_{Jet} for the Earth-to-Mars trajectory with the Mars-bound payload and propellant for the return trip included in the vehicle's weight. This requires an iterative method, because the vehicle "dry" mass depends on the mass of the total propellant carried as well as the total trip time (i.e., tankage for the propellant and number of thrusters required based on thruster lifetime, respectively). Details of the mission analysis code are described in the Appendix. Inputs to the code include vehicle power and efficiencies (i.e., P_{Jet}), I_{sp}, and number of vehicles (i.e., payload per vehicle). Outputs include propellant mass, vehicle "dry" mass, propellant resupply tanker mass, and trip time.

3.3 RESULTS

Figures 3-7 and 3-8 present the results for a variety of SEP systems in the 100 to 500 MW_e range, with I_{sp}s from 5,000 to 20,000 lb_f-s/lb_m. Specific impulses of 10,000 lb_f-s/lb_m or higher are needed to compete with the IMLEO of the baseline aerobraked chemical O_2/H_2 vehicle option. This is due, in part, to the fact that the SEP system has a specific mass almost three times that of the NEP system discussed in the next Section.

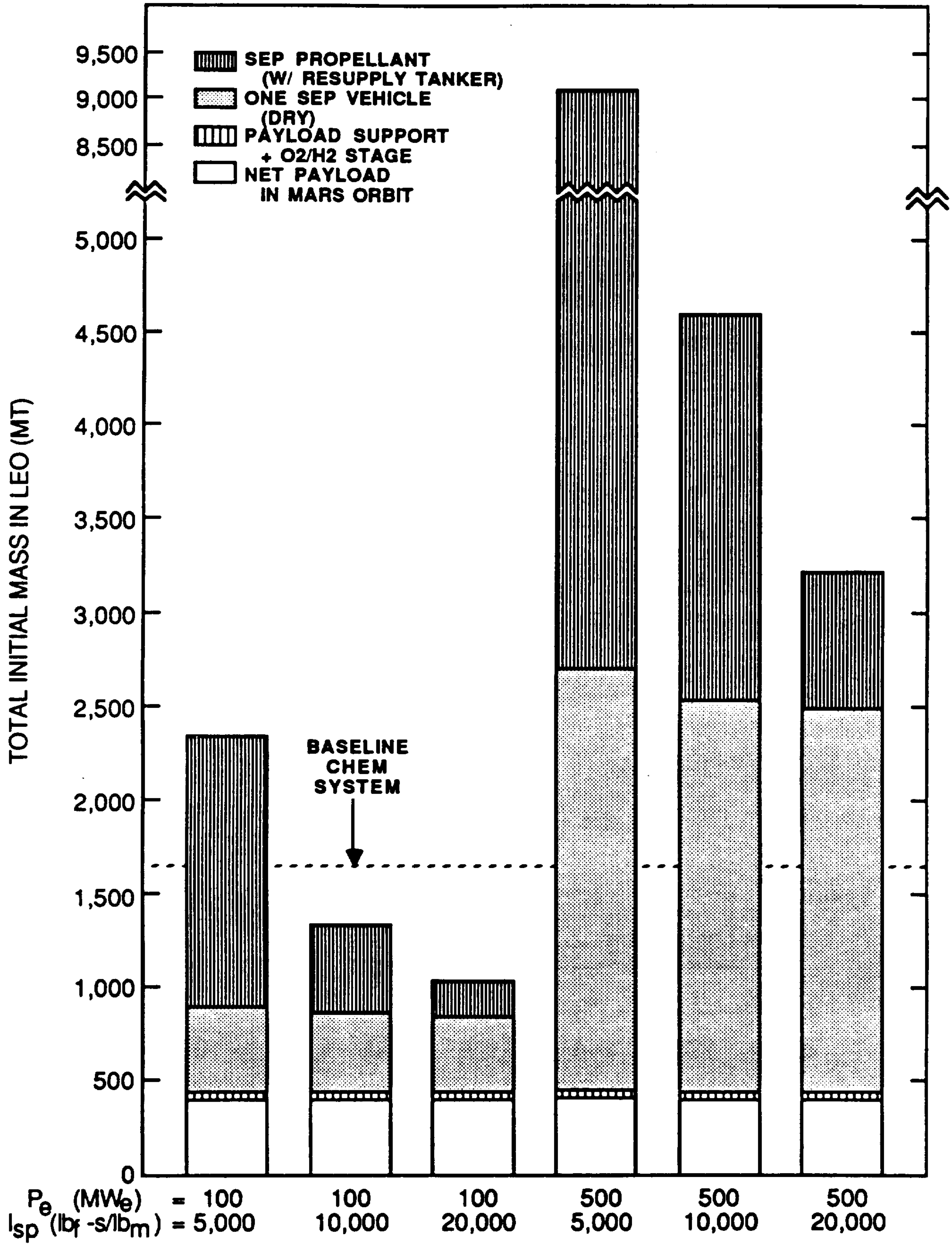

Figure 3-7. Solar Electric Propulsion System Mass Breakdown

However, an SEP vehicle has several operational advantages over an NEP vehicle. For example, the SEP vehicle can operate directly from LEO so an OTV infrastructure, with its associated mass and trip time penalty, is avoided. Also, an SEP vehicle is not nuclear-powered vehicle, so the "political" and operational issues (e.g., shielding versus standoff distance to protect nearby spacecraft) associated with space nuclear power are absent.

Finally, as seen in Fig. 3-8, there is a severe penalty on IMLEO when the cargo is split among several vehicles in an attempt to reduce trip time. This is due to the significant dry weight of these vehicles. For example, placing half the total payload on two vehicles reduces trip time by only about 10%, but almost doubles the IMLEO because two SEP vehicles (and their associated propellant) are now required. Thus, there is little or no advantage to splitting the cargo among many heavy vehicles in an attempt to decrease trip time.

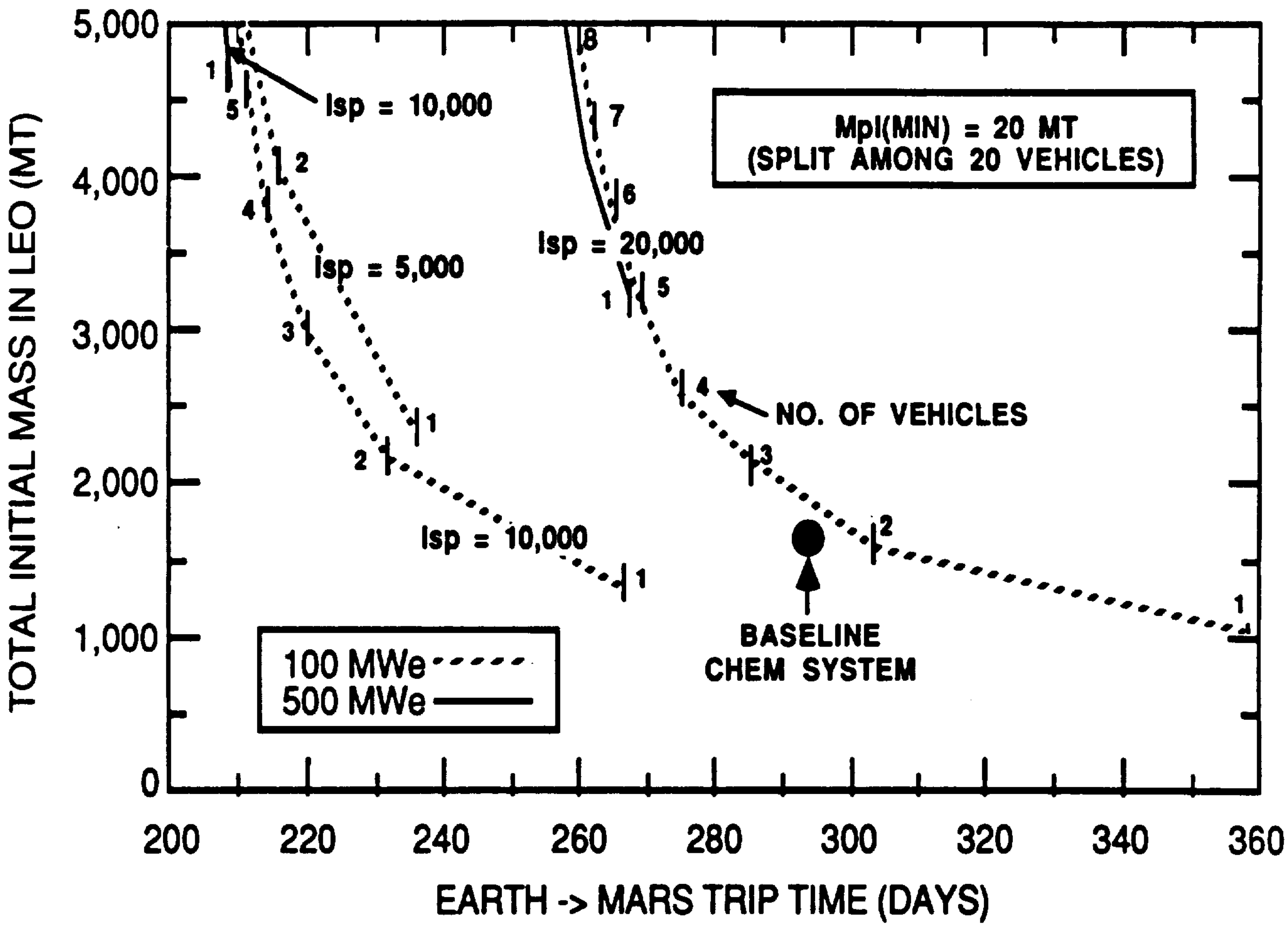

Figure 3-8. Initial Mass in Low Earth Orbit vs. Earth-to-Mars Trip Time for 100-MW Class Solar Electric Propulsion

3.4 CONCLUSIONS

On the basis of IMLEO and trip time, a 100-MW_e class NEP system out-performs a similar-sized SEP system. Also, high I_{sp}s (>10,000 lb_f-s/lb_m) are required by the SEP system to show major improvements in IMLEO over a chemical system. However, although lower in performance than a NEP system, a 100-MW_e class SEP system may be preferred in an overall operational or "political" context. It is recommended that future studies be performed to address both the technical as well as operational requirements of 100-MW_e class SEP and NEP in order to more fully understand their relative strengths and weaknesses.

3.5 REFERENCES

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SECTION 4

100-MW CLASS NUCLEAR ELECTRIC PROPULSION

4.1 INTRODUCTION

A Nuclear Electric Propulsion (NEP) system consists of a nuclear reactor and a thermal-to-electric conversion system, as well as a power processor unit (PPU) and electric thrusters. Unlike solar photovoltaic arrays, which scale approximately linearly with power level (i.e., a roughly constant specific mass), a nuclear power supply has the ability to make use of significant economies of scale. Thus, whereas a solar photovoltaic array in a 100-MW_e class Solar Electric Propulsion (SEP) vehicle has a specific mass of 3.6 kg/kW_e, a similar-power nuclear reactor with a high-temperature, Rankine dynamic power conversion system has a specific mass of only 0.9 kg/kW_e. Therefore, there is the potential for significant mass and trip time savings with NEP over SEP.

This potential benefit, however, is offset by the infrastructure required to base the NEP at a Nuclear Safe Orbit (NSO) of, typically, 700 to 1000 km altitude. This high altitude is required to ensure that, in the event of a catastrophic failure, there will be sufficient on-orbit stay time for any radioactive components to decay to safe levels before re-entering Earth's biosphere. The actual altitude depends on the ballistic coefficient of the vehicle (i.e., mass versus drag) and the levels of harmful nuclear isotopes that must decay to safe levels before air drag causes the vehicle's orbit to decay and re-enter. In this study, a 1000-km NSO is assumed. Also, it was assumed that a combination of stand-off distance and (limited) 4-pi steradian shielding would prevent damage to other vehicles or interference with science experiments (e.g., gamma-ray astronomy) in nearby orbits. However, these issues need to be addressed in detail in future studies of 100-MW_e class space nuclear power systems.

One interesting aspect of NEP operation that has been identified in previous studies is that an NEP vehicle can safely travel once, initially, from low Earth orbit (LEO) to NSO, because the reactor starts out "cold" (little or no harmful nuclear isotope inventory). As the reactor is operated and the vehicle begins to spiral out to NSO, the rate at which harmful nuclear isotopes build up is such that, were the system to fail at that point, the orbital lifetime achieved at that point would exceed the time required for safe decay of the harmful isotope inventory that has been produced to that point. Thus, a NEP vehicle can boost itself the first time to NSO. However, after prolonged operation it cannot return to LEO for periods typically given as several hundred years.

A schematic of a 100-MW_e class NEP vehicle is shown in Fig. 4-1. The vehicle configuration is dominated by the radiators required to radiate waste heat from the thermal-to-electric power conversion system. As will be shown below, the radiators also represent a significant fraction of the vehicle mass. Finally, an Orbit Transfer Vehicle (OTV) infrastructure will be required to transfer payloads and propellants to the 1000-km NSO assumed in this study; however, as seen below, this infrastructure represents a small fraction of the total initial mass in low Earth orbit (IMLEO).

4.2 ASSUMPTIONS

4.2.1 Mission Scenario

For the NEP system, an OTV infrastructure, like that used for the Solar Sails, is used to transport payloads and propellants to the 1000-km NSO. As with the 100-MW_e class SEP, power levels range from 100 to 500 MW_e. Also, payloads can be split

between several NEP vehicles flying in parallel to Mars. The mission scenario is illustrated in Fig. 4-2.

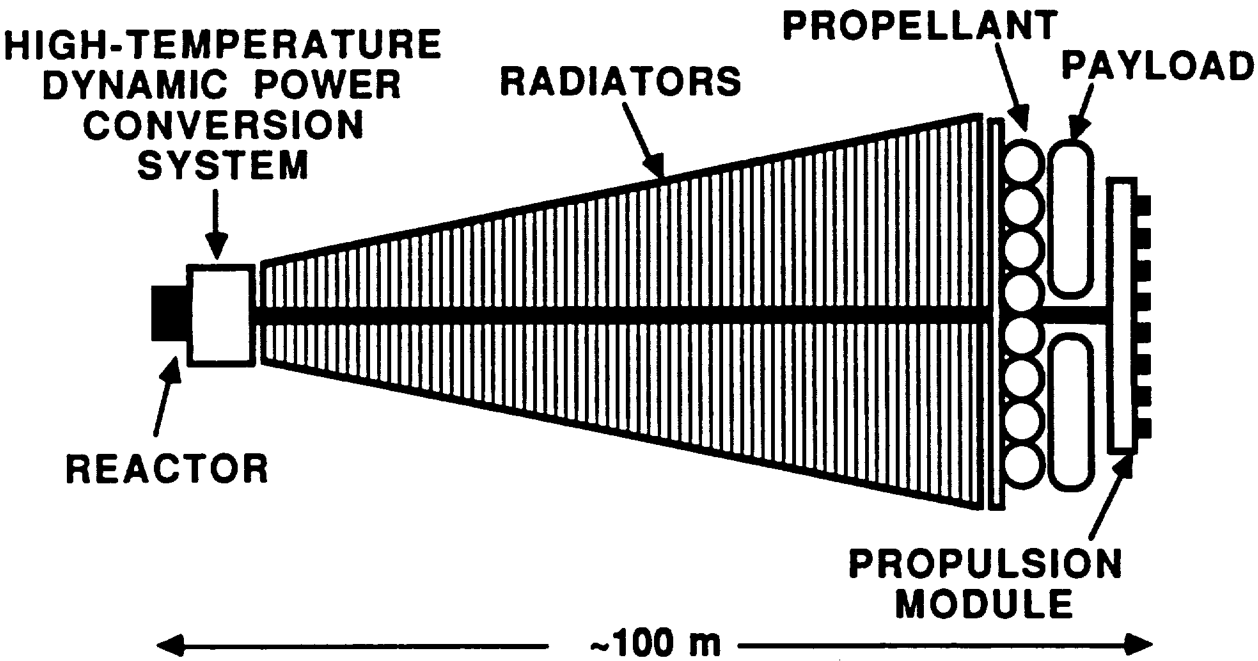

Figure 4-1. 100-MW Class Nuclear Electric Propulsion (NEP) Concept

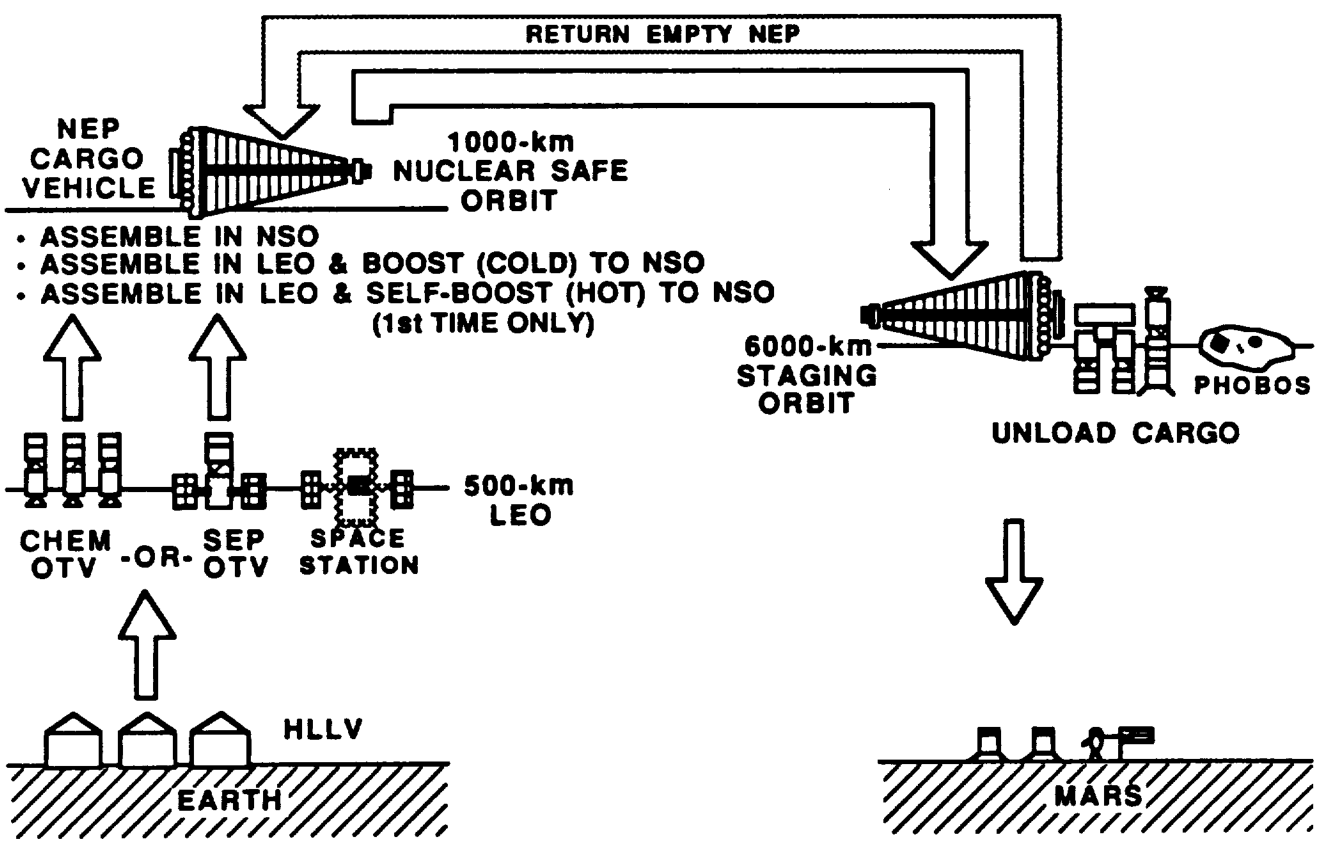

Figure 4-2. 100-MW Class Nuclear Electric Propulsion Mission Scenario

4.2.2 100-MW$_e$ NEP Vehicle Sizing

4.2.2.1 Nuclear Power System Specific Mass. A nuclear fission reactor is used with a high-temperature dynamic conversion system to give a specific mass of 0.92 kg/kW$_e$ at 100 MW$_e$; this decreases to 0.87 kg/kW$_e$ at 500 MW as shown in Fig. 4-3.

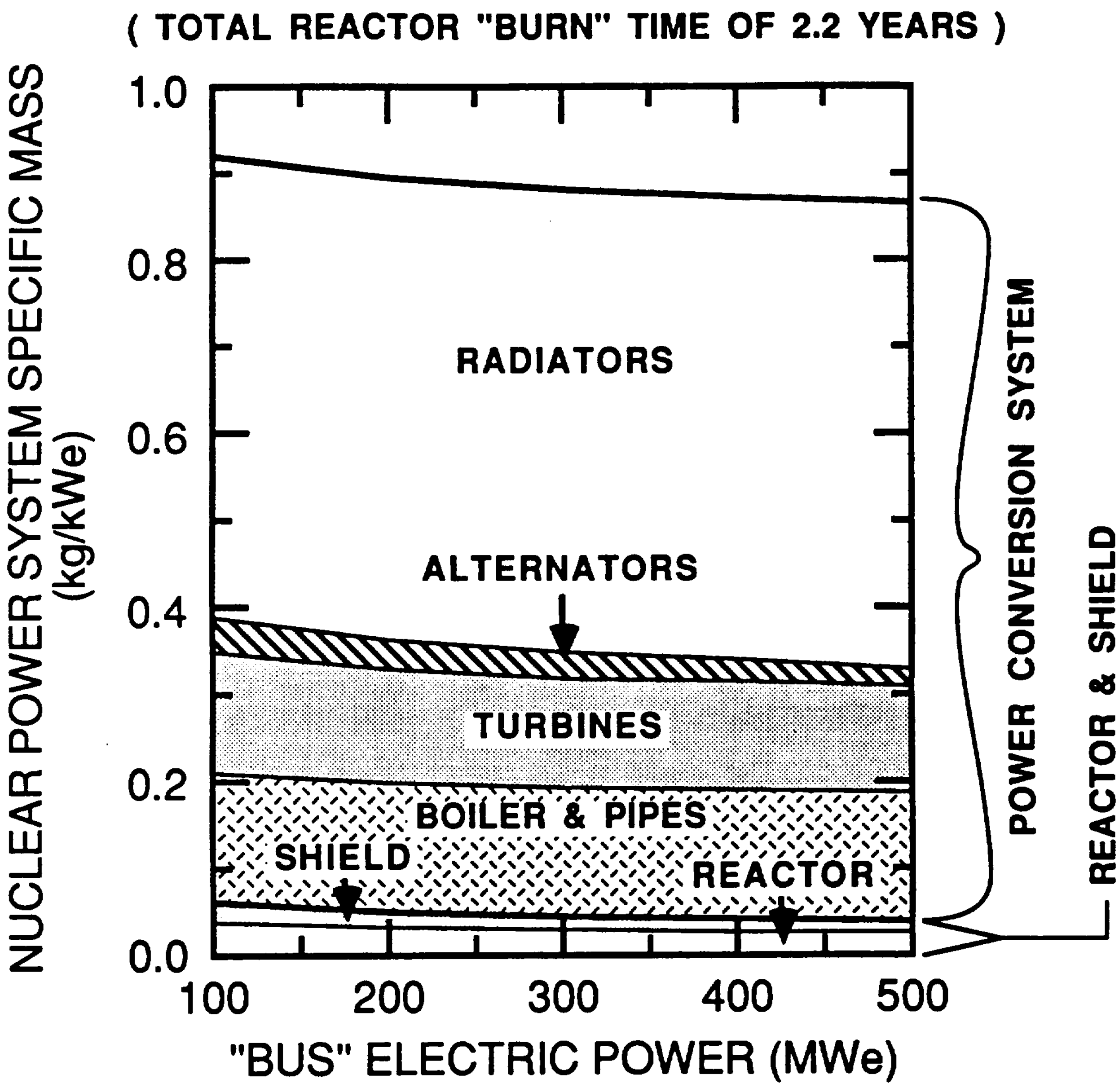

Figure 4-3. NEP Power System (Reactor Plus Conversion System) Specific Mass vs. Power Level

The high-power reactor and power conversion systems were evaluated in a previous study.[2] Details of the nuclear power system sizing are given in the Appendix. The reactor subsystem consists of the reactor vessel, nuclear fission fuel, and radiation shadow shield. The scaling equations for these components expressed as a function of

the "bus" electric power (P_e) are:[2]

$$\text{Reactor Mass [MT]} = 0.0949 \cdot (P_e \, [MW_e])^{0.5} + 0.0693 \cdot (P_e \, [MW_e])^{0.641}$$

$$\text{Nuclear Fuel Mass [MT]} = 2.055 \times 10^{-5} \cdot (P_e \, [MW_e]) \cdot (T_{Burn} \, [\text{Days}])$$

$$\text{Shadow Shield Mass [MT]} = 0.0739 \cdot (P_e \, [MW_e])^{0.7}$$

where T_{Burn} is the reactor operating time at full power.

The Rankine dynamic power conversion system consists of a liquid potassium boiler, turbines, alternators, and waste-heat radiators. The scaling equations are:[2]

$$\text{Boiler Mass [MT]} = 0.15 \cdot (P_e \, [MW_e])$$

$$\text{Turbine Mass [MT]} = 0.223 \cdot (P_e \, [MW_e])^{0.9}$$

$$\text{Alternator Mass [MT]} = 0.129 \cdot (P_e \, [MW_e])^{0.75}$$

$$\text{Radiator Mass [MT]} = 0.530 \cdot (P_e \, [MW_e])$$

4.2.2.2 Power Processor Unit (PPU) and Ion Thruster Specific Mass and Efficiency. The high-power PPU systems were evaluated by Stanley Krauthmer of JPL.[3] The PPU has a lower specific mass, 0.25 kg/kW$_e$, and higher efficiency, 99%, than the corresponding system for the SEP vehicle because the high-voltage output from the reactor's dynamic power conversion system requires less processing for the high-voltage ion thrusters than the low-voltage output from the solar photovoltaic system. The mass of the PPU for the NEP system is:

$$\text{PPU Mass [MT]} = (0.25 \, [MT/MW_e]) \cdot P_e \, [MW_e]$$

The mercury ion thrusters are the same as those used in the 100-MW$_e$ class SEP system as discussed in Section 3.

4.2.2.3 Final Vehicle Sizing. The overall vehicle mass is found by summing the masses of the power system (reactor and conversion system), PPU, thrusters, and propellant tankage, as was done for the 100-MW$_e$ class SEP system discussed in Section 3. Also, as with the SEP system, the total propulsion ("jet") power is a function of the efficiencies of the PPU and thruster. Finally, a propellant resupply tanker (2 % tankage factor) is included in the total initial mass in low Earth orbit.

4.2.3 Support Infrastructure

4.2.3.1 OTV Infrastructure. As mentioned above, it is assumed that the NEP system operates from a 1000-km NSO. Thus, it is necessary to transport propellant and payloads from LEO to NSO using an infrastructure consisting of chemical (O_2/H_2) or 100-kW$_e$ SEP OTVs as was done for the Solar Sails. The high-thrust (chemical) and low-thrust (SEP) Delta-Vs for the transfer from LEO to NSO are 262 and 263 m/s, respectively. Using the chemical and Xe-ion OTVs described in Section 2, this results in a chemical and SEP OTV payload capacity for the LEO to NSO transfer of 632 MT and 464 MT, respectively. For the SEP OTV, the LEO-to-NSO trip time is 684 days while the return trip require only 6 days, assuming that the SEP OTV is in sunlight 64.7 % of the time for both legs of the trip. It is assumed that the "first time" transfer of the NEP from LEO to NSO is made under its own power but with no payload or excess propellant; this typically requires less than a day.

4.2.3.2 Payload Support. As with the high-powered SEP vehicle, this NEP vehicle is large (on the order of 100 m in length), so an Orbital Maneuvering Vehicle (OMV) is added to each payload for rendezvous and docking in Earth and Mars orbit.

4.2.4 Trajectory Analysis

The low-thrust NEP trajectories were provided by Carl G, Sauer Jr. of JPL.[4] The results of the trajectory code are given in the Appendix; the mission analysis methodology is similar to that described in Section 3 for the SEP system. The primary difference between the SEP and NEP mission analysis is the need to add the OTV infrastructure (100-kW_e SEP OTVs, propellant, and propellant resupply tankers) to the NEP system.

4.3 RESULTS

Figures 4-4 and 4-5 illustrate the performance of the 100-MW_e class NEP system for various combinations of power and I_{sp}. The most striking result is the performance of the high-I_{sp} (20,000 lb_f-s/lb_m) NEP system at 100 MW_e. This vehicle has an IMLEO (700 MT) that competes directly with solar sails, yet has an Earth-to-Mars trip time (240 days) that is less than that required for the minimum-energy trajectory chemical system. At an I_{sp} of 10,000 s, a 300-MW_e system has an IMLEO comparable to the chemical system, but with less than one-half the trip time of the chemical system. Even at a modest I_{sp} (5,000 lb_f-s/lb_m), the NEP system can best the chemical system in trip time.

This performance is due to the economies of scale in a 100-MW_e class NEP. For example, the overall total system specific mass (power, PPU, thrusters, tankage) ranges from a low 1.1 kg/kW_e (at 20,000 lb_f-s/lb_m I_{sp} and 500 MW_e power) to only 1.7 kg/kW_e (at 5,000 lb_f-s/lb_m and 100 MW_e). This performance is somewhat offset by the need for an OTV fleet to deliver propellant and payloads to a 1000-km NSO, but as with Solar Sails, this is only a small fraction (1 to 2 %) of the total IMLEO.

4.4 CONCLUSIONS

A 100-MW_e class NEP system has superior mass and trip-time performance. Only advanced solar sails can achieve a lower IMLEO, yet the NEP has a shorter trip time than the chemical system. In fact, the NEP system can make a round-trip in less time, and with a lower IMLEO, than a chemical system making a one-way delivery.

Against this performance advantage must be weighed the intangible "political" issues of space nuclear power. In that respect, a corresponding 100-MW_e SEP system, although lower in performance (higher specific mass), may be preferred. There are also significant technological challenges in the development of long-lived 100 MW_e space reactors, although the Strategic Defense Initiative may address some of these issues. It should be noted that there is a significant terrestrial reactor technology base for stationary and mobile (e.g., submarine) reactors in this size range. Thus, if the technological, political, and operational issues (e.g., shielding versus standoff distance to protect nearby vehicles) can be resolved, the 100-MW_e class NEP promises to be an attractive option for the Mars cargo mission. In fact, its short trip time may make it more suited to the manned portion of the mission, where trip time is a major figure of merit. It is recommended that a detailed study be performed to assess the 100-MW_e class NEP for both piloted and cargo missions so as to identify the mission benefits and technology requirements of this concept.

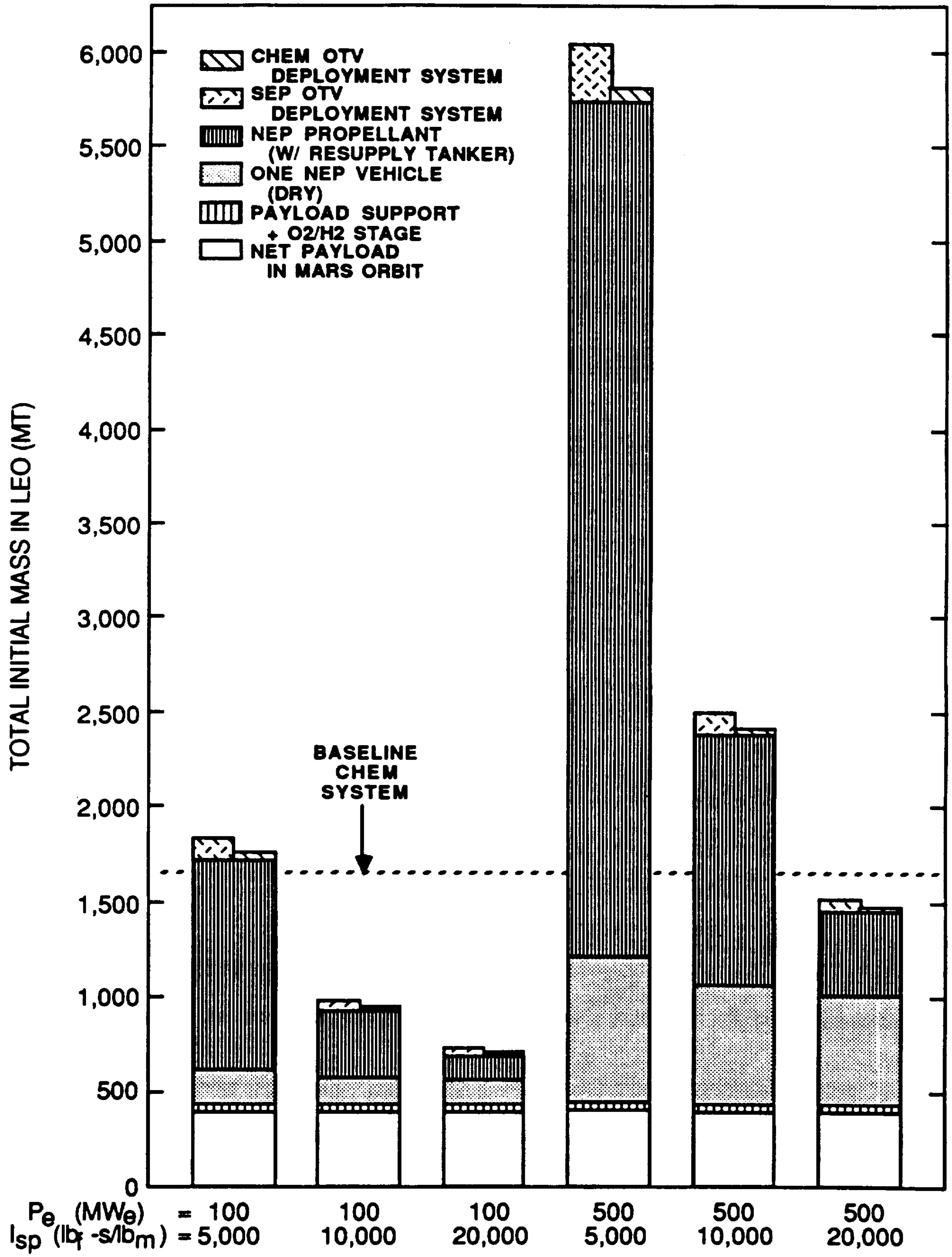

Figure 4-4. Nuclear Electric Propulsion System Mass Breakdown

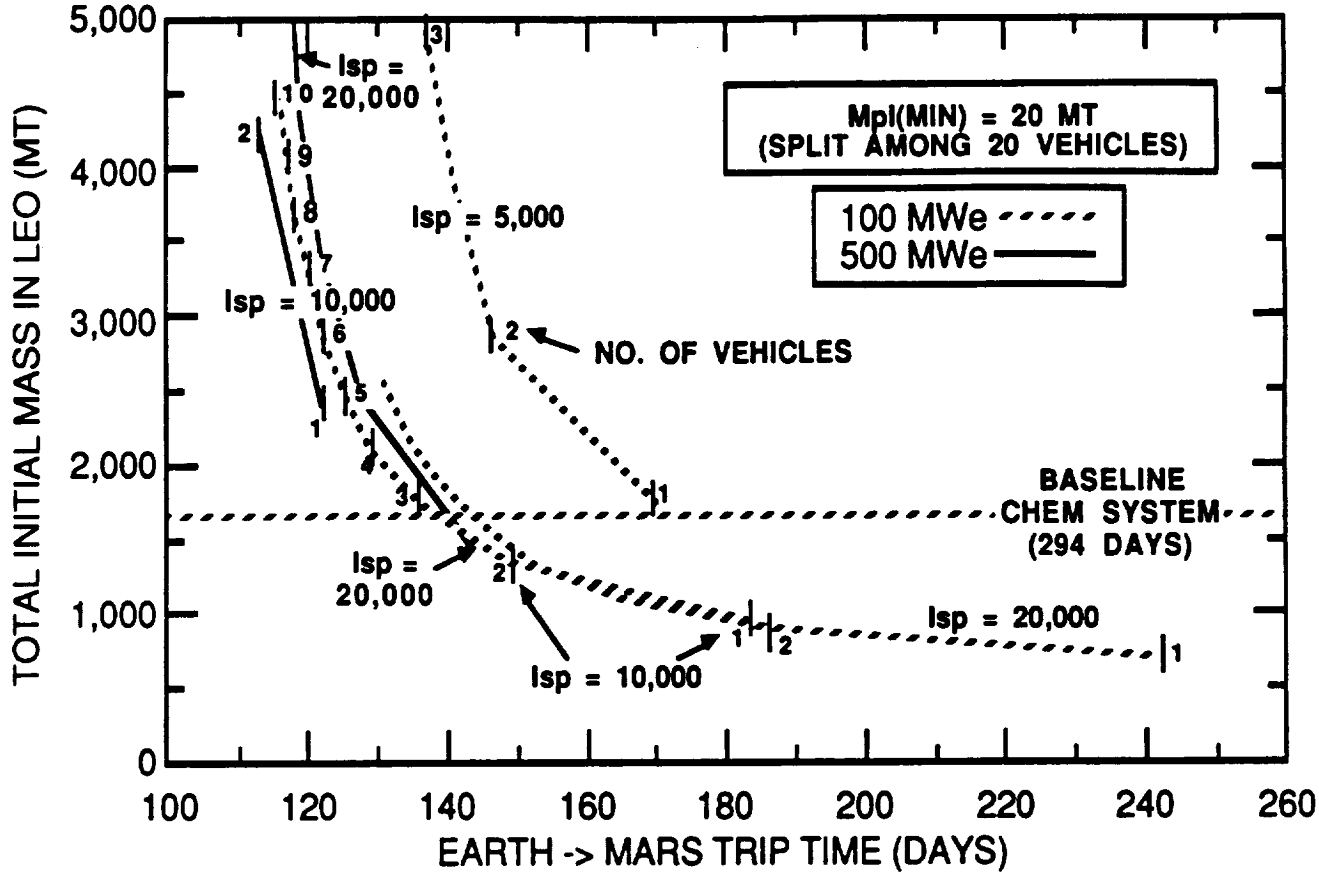

Figure 4-5. Initial Mass in Low Earth Orbit vs. Earth-to-Mars Trip Time for 100-MW Class Nuclear Electric Propulsion

4.5 REFERENCES

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4. Sauer, C.G., Jr., Personal Communication, June 1989.

SECTION 5

MAGNETIC SAILS (MAGSAILS)

5.1 INTRODUCTION

5.1.1 Background

The magnetic sail, or magsail, is a novel concept recently introduced by Robert Zubrin and Dana Andrews.[1-3] A literature survey uncovered no previous description of such a device. Figure 5-1 shows a conceptual diagram of the magsail concept. It consists of a cable of superconducting material, millimeters in diameter, which forms a hoop that is tens to hundreds of kilometers in diameter. The current loop creates a magnetic dipole which diverts the background flow of solar wind. This deflection produces a drag-force on the magsail radially outward from the sun. In addition, proper orientation of the dipole may produce a lift-force which could provide thrust perpendicular to the radial drag-force. The combination of these forces can be used to transport the magsail and cargo on interplanetary or interstellar missions.

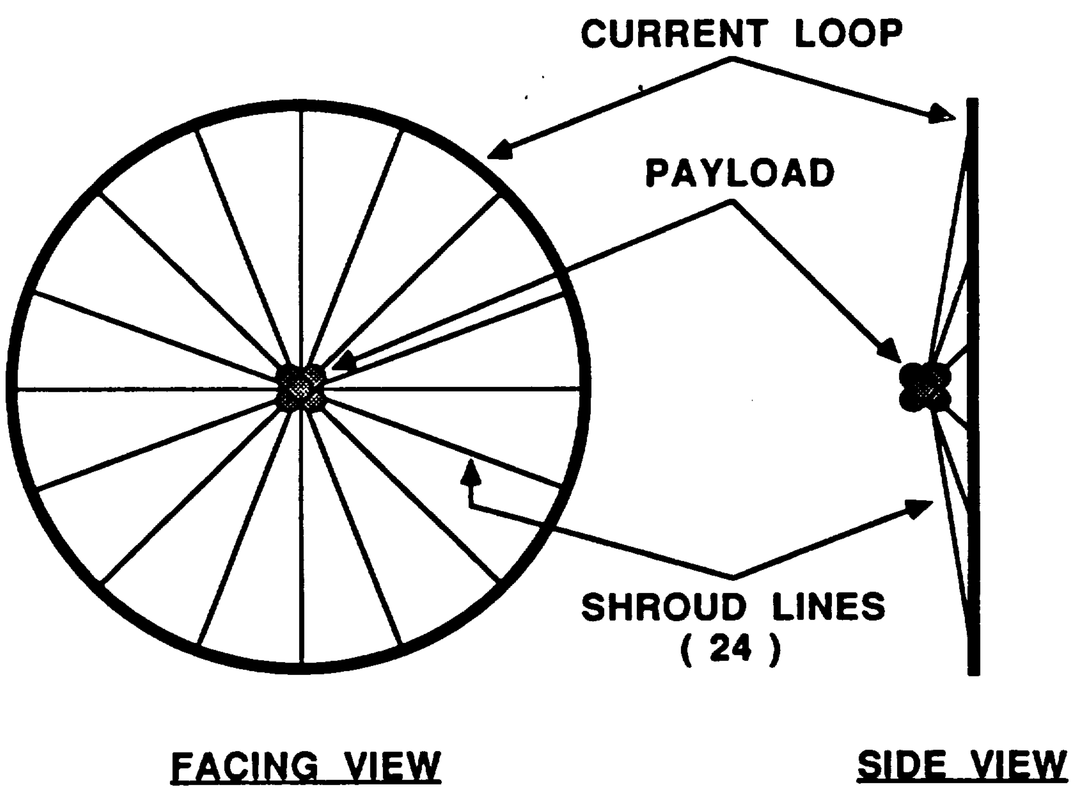

Figure 5-1. Magsail Deployed With Payload, Normal Configuration (Adapted from Reference 3)

5.1.2 Operational and Technical Feasibility Issues

As a relatively new concept, the magsail possesses a number of unresolved operational and technical feasibility issues. Dr. John L. Callas of JPL assisted in the

definition and evaluation of these feasibility issues,[4] which include thermal control, structures, radiation, superconductor technology, attitude control, deployment, planetocentric operation, and interaction with the solar wind. Each of these issues is described in detail below. Solutions to some of these issues may require the application of advanced technology (e.g., superconductors), others may only require innovative engineering (e.g., thermal control).

5.1.2.1 Thermal Control. For the current-carrying cable to remain superconducting, its temperature must be maintained below the critical temperature of the embedded superconductor. Preliminary thermal modeling indicates that in addition to passive reflective coatings, some form of active cooling system will be required to maintain the magsail cable below 100 K in Earth-Mars space.

If the thermal control scheme is based upon a continuous capability to orient the sail (i.e., to maintain a "hot side" and a "cold side" with different absorptivity and emissivity), temperature control during deployment and inflation may be difficult. For example, if the superconductor is inadvertently or intentionally quenched, attitude control is lost and the cable may drift from proper orientation, warming it above the superconductor critical temperature (T_C). This event could pose a catastrophic failure mode because attitude control could not be regained until the cable could again be cooled and powered-up.

5.1.2.2 Structures. An initial baseline design discussed by Zubrin and Andrews[2] describes a magsail 64 km in diameter, with a cable diameter of approximately 5 mm. These dimensions suggests that the structure will be susceptible to vibrational motion; the cable material must be very malleable to survive this motion without fracture. Current high-temperature superconductors are like brittle ceramics in terms of their material properties. Whether or not a superconductor material can be manufactured possessing the proper resiliency and malleability is an important feasibility issue. It may be necessary to enclose the superconducting cable in a flexible sheath of Kevlar (or some other material like Kevlar which is effective for tether applications, but is more appropriate for a low-temperature application than Kevlar), to provide flexible tensile support.

5.1.2.3 Radiation. The magnetic field of the magsail may generate local Van Allen-type radiation belts. These belts may pose a significant radiation hazard for payload or crew in the vicinity of the magsail, though not at the geometric center of the magsail hoop. The background solar wind and cosmic-ray radiation may also induce long-term cumulative radiation damage in the superconducting hoop, degrading the superconducting properties of the material.

5.1.2.4 Superconductor Technology. The baseline magsail designs of Zubrin and Andrews rely upon significant advancements in superconductor technology such that the assumed critical current density of 1 to 2 $\times 10^{10}$ Amps/m^2 must be achieved in bulk form in high-temperature superconductors. Recent findings[5] suggest that Type II superconductors designed for high-critical-temperature operation ($T_C > 77$ K) are susceptible to "giant flux creep" (which creates resistance in the superconductor) in the presence of a magnetic field. The superconductor characteristics and operating environment assumed for current magsail designs describe a demanding combination of conduction current density, critical temperature, and magnetic flux density. If no solution is found to the problem of giant flux creep, subsequent reduced superconductor conduction current density and critical temperature will significantly reduce magsail performance.

It is also necessary to design the magsail system to survive a "quench", in which the superconducting material loses its ability to conduct current without resistance. A

quench may be caused by a rise in temperature above the superconductor critical temperature, or a rise in the magnetic flux density above the critical field of the superconductor. In addition, there may be situations in which it will be desirable to significantly reduce or eliminate the current in the superconducting cable (e.g., for navigation, or to release charged particles trapped in induced radiation belts). Quench capability could be provided by an external resistor bank.

5.1.2.5 Attitude Control. One potential attitude control scheme is similar to that proposed for use on solar sails in which articulated control vanes (separate small superconductor loops for the magsail) are used to modulate the center of pressure of the sail while the center of mass remains fixed. A second approach, again proposed for solar sails, would be to shift the center of mass of the magsail could by moving part of the payload out along a shroud line while the center of pressure remains fixed. The difference in location between the center of pressure of the magsail and the center of mass would induce a torque and small angular acceleration. For example, assuming that 50 MT of payload could be offset 10 % of the hoop radius, the resulting torque could change the orientation of the baseline magsail by 90° in 10 to 12 hours. Several issues arising from this scheme remain unresolved. Local variations in the solar wind density may cause a random perturbation of the center of pressure which complicates the application of this attitude control scheme. In addition, the slow response time (e.g., 10 to 12 hours to rotate 90°) may make it difficult to execute a planetocentric "pumping" orbit-raising maneuveur if thrust vectoring is required.

5.1.2.6 Deployment. The size and electrical current in the superconducting magsail cable imply significant energy storage. For example, the energy stored in the cable of a 64-km diameter, 10^{-5} Tesla magsail is approximately 8×10^{10} Joules. A continuously operating 10-kW solar array would require approximately 93 days to energize the cable to full power. This large energy storage suggests two potential problems. First, if magsail deployment and "inflation" require a large amount of time, the magsail may lack attitude control during this period, which could lead to a subsequent loss of thermal control, as well as unusual mechanical stresses. Second, it may be difficult to modulate the current in the magsail cable in the manner required for a "solar-pumping" maneuver described below. One possible method for rapid sail deflation would be to redirect part of the cable electric current to a radiative resistor bank, although this may aggravate the difficult magsail thermal control problem.

5.1.2.7 Planetocentric Operation. Thus far, it is not known if the magsail can be operated near a planet's magnetosphere. In their analyses, Zubrin and Andrews have constrained magsail operation to heliocentric space: "For our reference spacecraft, starting in very high Earth orbit and about to orbit the sun at Earth radius...".[2] Clearly, a magsail cannot be used within a planet's magnetosphere (between the magnetopause and the planet's surface) because there is no solar wind there. The minimum distance from the center of the Earth to the Earth's magnetosphere is 10 Earth radii, or 64,000 km.[6] Other planets have significantly varying magnetosphere sizes based on the planet's magnetic field strength.

It will be difficult for the magsail to operate in planetocentric orbits of even higher altitude (above the magnetopause), because in order to gain altitude in the orbit the magsail must execute a "solar-pumping" maneuver analogous to that originally conceived for solar sail orbit raising and escape. In a planetocentric solar-pumping maneuver, the solar sail is feathered such that solar photon pressure is minimized when the sail is heading sunward. The sail is then re-oriented to maximize solar photon pressure when it is flying away from the sun. In this way, the apogee of the sail orbit is incrementally boosted to achieve higher-energy orbits or escape. However, unlike a solar sail, re-orienting the magsail hoop does not significantly modulate the radial solar wind

drag-force. In order for the magsail to execute a solar-pumping maneuver, the drag-force "thrust" (and possibly "lift") would be modulated during each orbit. In order to reduce or eliminate the radial drag force on the upwind leg, the electric current in the magsail cable could be reduced or eliminated. This current-modulation scheme suggests several operational issues. The magsail may lose attitude control, as described above. In addition, the circular shape of the superconducting cable is a result of the solenoidal hoop-stress imparted to a current-carrying cable in an ambient magnetic field; if the magsail were quenched, the hoop may lose its shape as a result of gravity gradient or other perturbation forces. The magsail would be recharged before thrust could be generated on the downwind side. As suggested above, if the cable recharge is constrained by the onboard magsail power supply, then recharging the cable to full power may be time consuming (perhaps beamed power could be utilized). A potential solution to this problem is to execute the solar-pumping maneuver at a reduced magsail energy level to allow quicker magsail inflation and deflation.

5.1.2.8 Modeling of the Solar Wind-Magsail Interaction. Both a particle model and a fluid model have been proposed for calculation of the magsail drag-force radial thrust and lift-induced tangential thrust. A particle-based model of the solar wind-magsail interaction was developed by Callas[4] which roughly confirmed the results of Zubrin's particle model.[1,3] Callas' model predicts a thrust of approximately 200 N for the 64-km diameter, 10^{-5} Tesla magsail. Both Callas and Zubrin have concluded that a plasma fluid model is probably most appropriate for modeling the radial and possible tangential thrust of the magsail. Zubrin's plasma fluid model predicts a minimum (quiet solar wind) thrust of 538 N and an average lift-to-drag ratio of 0.28. In the Mars cargo mission analysis, it is shown that the initial mass in LEO is sensitive to the estimated magsail radial thrust, so this parameter is allowed to vary from 200 to 500 N. Perpendicular thrust (positive lift-to-drag) was not considered in the orbital analysis and mission performance. Furthur work is needed to fully understand the radial and tangential thrust characeristics of the magsail.

5.2 ASSUMPTIONS

5.2.1 Systems Analysis

With several exceptions, the technology assumptions and magsail design described in Ref. 2 are assumed for this study. The density of the 5-mm diameter magsail cable is assumed to be 5000 kg/m^3 (copper oxide). The assumed maximum current density is approximately 10^{10} Amps/m^2. The assumed maximum magnetic field strength on the normal-axis of the hoop is 10^{-5} Tesla.

Evaluation of the material properties of superconductors (i.e., brittleness) leads to an assumption that it will be necessary to add a tether-material sheath to the superconducting cable. The mass of the shroud lines and superconductor sheath are estimated using near-term tether materials such as Kevlar 29 with a density of approximately 1500 kg/m^3. This material does not maintain high tensile strength and malleability at the superconductor critical temperatures (77 K < T < 135 K), but is representative of the density of tether materials. Twenty-four shroud-lines, at 3.5 mm in diameter and 32 km in length apiece, have a mass of approximately 11 MT. A sheath of tether material, surrounding the superconductor, with a thickness of 3.5 mm, has a mass of approximately 49 MT.

Miscellaneous subsystems could include a solar-photovoltaic power supply, an attitude control system (e.g., center-of-gravity shift mechanism), cable-current modulation system, payload mounting structure, and passive and active refrigeration systems. The total mass of these subsystems is assumed to be 40 MT. Therefore, the total mass of the magsail (not including cargo) is 200 MT.

5.2.2 Mission Analysis

The mission requirement is to transport 400 MT of cargo from low Earth orbit (LEO) to a 6000-km circular Mars orbit. Figure 5-2 shows a conceptual magsail Mars cargo mission scenario. Both chemical (O_2/H_2) and 100-kW Xe-ion Solar Electric Propulsion (SEP) orbit transfer vehicles (OTVs) as described in Section 2 were evaluated for boosting the magsails and Mars cargo from LEO to the operational magsail orbit.

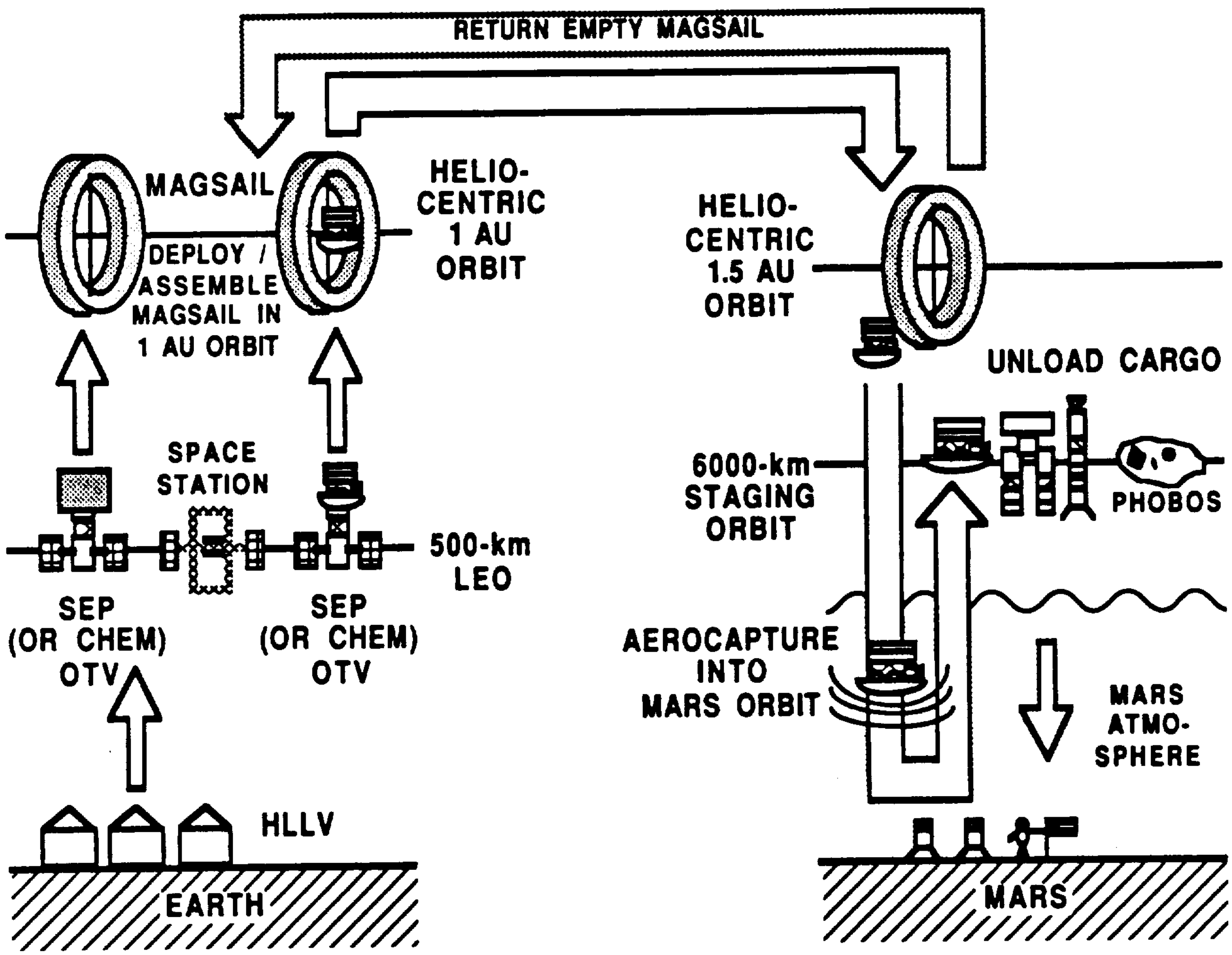

Figure 5-2. Magsail Mission Scenario

The magsail accomplishes only the heliocentric portion of the Mars cargo delivery mission. The cargo is then released in a close-approach to Mars, as shown in Fig. 5-3. A single-stage aerobraked chemical O_2/H_2 propulsion vehicle (Isp = 470 lb_f-s/lb_m) similar to the second stage of the baseline chemical vehicle described in Section 1 is attached to the payload. This vehicle accomplishes rendezvous and docking in Earth orbit with the LEO-to-1 AU OTV and in very high (heliocentric) Earth orbit with the magsail (Delta-V = 50 m/s), and performs a circularization maneuver (Delta-V = 650 m/s) following aerobraking at Mars. The variation in vehicle mass with payload mass is shown in Fig. 5-4.

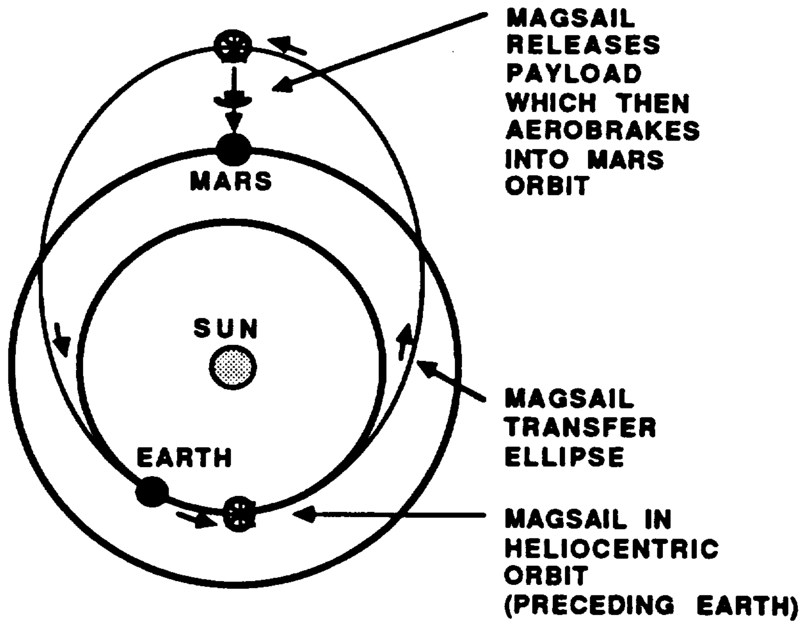

Figure 5-3. Earth-to-Mars Magsail Cargo Mission Trajectory (Adapted from Reference 2)

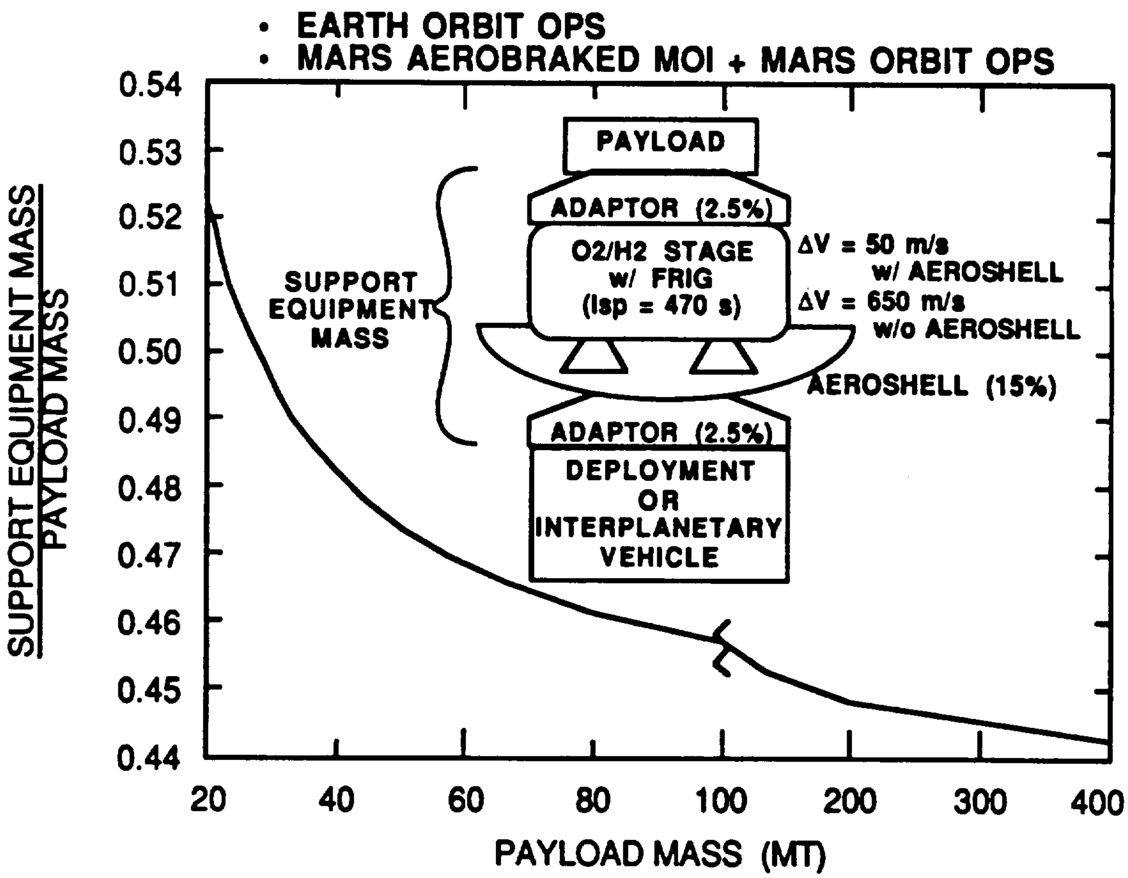

Figure 5-4. Effective Payload Weight Due to Adding an Aerobraked Chemical (O_2/H_2) Stage for Earth and Mars Orbital Operations and Aerobraking Into Mars Orbit

It is assumed that a magsail can only be operated effectively in heliocentric space (outside the gravitational influence of any planetary bodies). Thus, the components of the magsails are transported from LEO to 1 AU and then assembled in a 1-AU heliocentric orbit slightly ahead of the Earth. Thus, unlike a Solar Sail, both high- and low-thrust OTVs can be used because the magsail is not "inflated" in LEO. For the chemical OTV, the one-way high-thrust Delta-V and trip time from LEO to the heliocentric position (C_3 = 0 km^2/s^2) is 3.153 km/s and about one day, respectively, assuming the performance parameters given in Table 2-2 and a one-way (no return) deployment scenario. For the Xe-ion SEP OTV, the one-way low-thrust Delta-V and trip time from LEO to 1 AU is estimated to be 7.612 km/s and 747 days, respectively. In this case, a "stretched" Xe-ion SEP OTV is used in a round-trip deployment scenario (the SEP OTV returns to LEO empty). The dry mass of the "stretched" SEP OTV is 4.933 MT and the propellant capacity is twice that given in Table 2-2 with all other parameters (I_{sp}, power, etc.) the same. Propellant resupply tankers are also included in the OTV infrastructure, as discussed in Section 2.

The magsail is loaded with payload (and aerobraked chemical vehicles, etc.) and the thrust then found which results in a 284-day Hohmann-type transfer ellipse to Mars' heliocentric orbit. Based on Zubrin's estimates of the drag-force (538 N) that the magsail can produce, the magsail has insufficient performance to circularize in Mars orbit, so the payload is dropped off for aerocapture into Mars orbit. The magsail returns to Earth in approximately 284 days. If future analyses show that the magsail drag-force is sufficiently high, or if the magsail is loaded with a smaller amount of cargo, the magsail drag-force may be sufficient to circularize the loaded sail in a heliocentric orbit at 1.52 AUs (Mars' distance from the sun). The circular velocity of the loaded magsail will be several kilometers/second less than Mars' orbital velocity. This allows the magsail to "loiter" before dropping off the Mars cargo, and eliminates the need to wait for Hohmann-type transfer opportunities every 2.2 years. However, based on current estimates of the drag-force, the allowable cargo mass per magsail is sufficiently reduced that this is not considered an effective option. For example, a drag-force of 618 N is required to circularize a magsail with no payload at 1.52 AU; this force increases to 712 N for a magsail loaded with a minimum-size Mars cargo element (20 MT payload plus 10.4 MT O_2/H_2 stage).

5.3 RESULTS

5.3.1 Initial Mass in LEO and Trip Time

Figure 5-5 shows the initial mass in LEO for the Mars cargo mission using magsails of different drag-force thrust levels. The magsail's one-way trip time is not dependent upon thrust level, and is 284 days for the magsail heliocentric transfer plus 747 days for the SEP-OTV LEO-to-1 AU transfer, for a total of 1031 days. The total trip time is significantly reduced for the chemical OTV, but at the expense of a significant increase in mass. This figure also shows the substantial mass of the OTV infrastructure required to transfer the Mars cargo from LEO to the magsail. Also, unlike the other advanced propulsion options requiring an OTV infrastructure (e.g., solar sails), the trip time of the OTV required to deploy the magsail is included in its total trip time. This is primarily due to the need to transfer to a 1 AU heliocentric orbit. A "fast" manned set-up or repair flight is relatively "inexpensive" in terms of mass and trip time for a high Earth orbit (HEO), such as the 2000-km solar sail deployment orbit, but very "expensive" for a 1-AU orbit. Finally, note that a one-magsail system requires a drag-force significantly in excess of the available force (538 N) estimated by Zubrin.

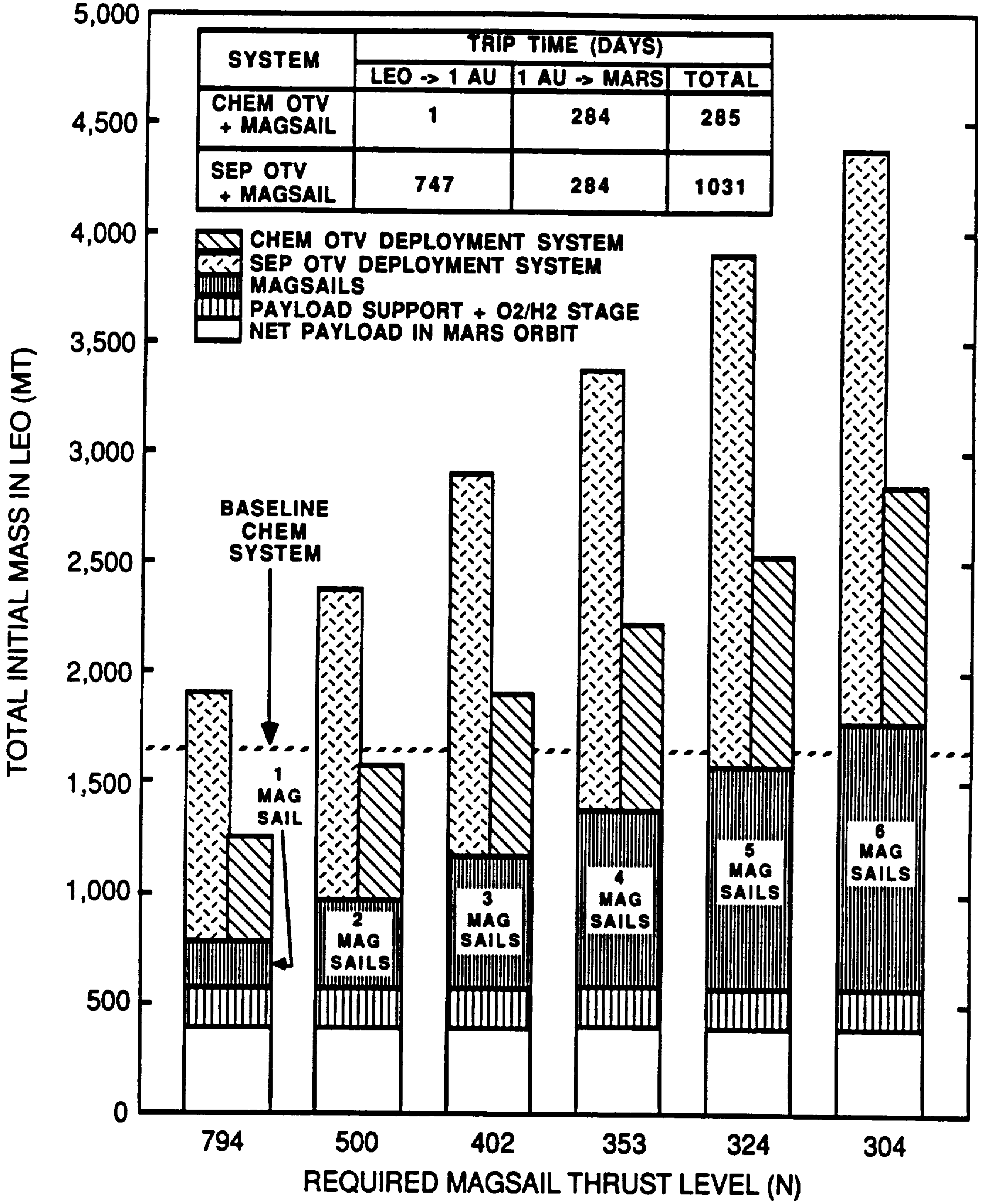

Figure 5-5. Initial Mass in Low Earth Orbit vs. Thrust Level for Magsail Propulsion

5.3.2 Comparison to Solar Sails

Solar sails and magsails share several general features. For example, both are large, gossamer structures, both have "infinite" specific impulse, and both require an OTV infrastructure for operation. This section compares and contrasts solar sails and magsails.

5.3.2.1 Size, Mass, Thrust, and Characteristic Acceleration. Zubrin and Andrews have suggested that useful magsail diameters are tens to hundreds of kilometers. It is possible that smaller sails (100's of meters in diameter) could also have limited mission applications. A typical size (constrained by Shuttle payload volume) for deployable square solar sails is 2 km x 2 km. Optimistically, the mass of a 64-km diameter magsail is at least 200 MT. Estimates of solar sail areal density for near-term deployable systems range from 5 to 7 g/m^2. A square solar sail measuring 2 km on a side would have a mass of approximately 20 MT. The characteristic radial drag-force thrust of the 64-km diameter, 10^{-5} Tesla magsail is estimated to be at least 500 N. The thrust of a 2x2-km solar sail is approximately 36 N. These thrust and mass estimates produce characteristic accelerations (acceleration at 1 AU, no payload) for the magsail and solar sail of 2.5 mm/s^2 and 1.8 mm/s^2, respectively. Thus, these are both low thrust-to-weight vehicles.

5.3.2.2 Operational Regime (Mission Applications). Solar sails can probably be used as low as a 2000-km altitude HEO, but they are prevented from operating any lower by atmospheric drag. As discussed above, magsails may not be usable in planetocentric space. The significant reduction in IMLEO for solar sails is due primarily to their capability to operate in HEO; thus they do not require a substantial infrastructure for LEO-to-HEO cargo transport.

Missions considered to be well-suited to the capabilities of near-term solar sails are a Mercury orbiter mission, high-inclination solar orbiter mission, and a multiple near-Earth asteroid rendezvous mission.[7] Magsails will probably find application in very high-energy interplanetary or interstellar missions.[1]

5.4 CONCLUSIONS

This preliminary study has attempted to define the primary technical and operational feasibility issues for magsails. Significant feasibility issues arise in the areas of thermal control, structures, radiation, superconductor technology, attitude control, reliability and redundancy, deployment, and planetocentric operation.

In addition, one measure of the performance of the concept (radial drag-force thrust) was estimated utilizing a particle-based model of the magsail-solar wind interaction. These calculations validated the particle-based results of the original authors, Zubrin and Andrews. However, Zubrin's plasma fluid dynamic model of the magsail-solar wind interaction produces an estimate of both the radial drag-force thrust and a tangential lift-force thrust that is considered to be more accurate than the particle-based model. Although no attempt was made to validate the results of Zubrin's plasma fluid dynamic model, his performance results are considered to be reasonable, and were used in our subsequent Mars cargo mission analysis.

Even if magsail feasibility issues can be overcome without compromising the performance assumed in this study, magsails are not as effective as several other advanced concepts, such as solar sails and nuclear electric propulsion, for the Mars cargo mission based on trip time and initial mass in Earth orbit. However, this is a relatively low-energy mission; the magsail may find more effective applications in the realm of very high-energy outer-planet or interstellar missions.

Future study should address the issues of performance, scale, and feasibility. Prior to any further system or mission analysis, it is necessary to develop an improved model of the magsail-solar wind interaction in order to more accurately assess the radial and perpendicular thrust of the concept. This model would also show the size range over which the magsail has an effective performance. Magsails may be scalable down to the size of solar sails and still provide enabling capability. Finally, if the potential performance of the concept is sufficiently interesting, a systems study could resolve the host of operational and technical feasibility issues.

5.5 REFERENCES

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SECTION 6

MASS DRIVERS AND RAIL GUNS

6.1 INTRODUCTION

Mass drivers (MD) and rail guns (RG) can be used as electric propulsion thrusters.[1,2] In both concepts, a propellant "pellet" is accelerated in a "bucket" or "container" that couples to an externally applied electromagnetic field. The propellant "pellets" are accelerated to high velocities (e.g., 12 km/s corresponding to an I_{sp} of 1200 lb_f-s/lb_m) and fired from the vehicle to produce thrust.

As shown in Fig. 6-1, the two concepts take different approaches to accelerating the propellant. The mass driver consists of many solenoid magnets which are energized in series to pull a payload bucket which contains its own magnet to couple to the externally applied fields. Very large mass drivers can be used to directly catapult vehicles from bodies such as the Moon which lack an atmosphere. Because any material can be placed in the payload bucket, a mass driver, when used as a reaction engine, can use any material as propellant. In this study, the option of using lunar-produced materials (e.g., lunar soil, oxygen, etc.) for propellant was considered as a means of reducing the initial mass in LEO. In general, mass drivers are large and complex, but have a high electric-to-jet power efficiency (70 to 90 % overall).

The rail gun is currently under consideration for use as a kinetic-energy weapon by the Strategic Defense Initiative Office. Although smaller and simpler than a mass driver, rail guns have a lower efficiency (45 % for the vehicle considered here) than a mass driver. Conceptually, the rail gun consists only of a power supply and two electrically energized rails. A "bucket" with a conductive armature is placed on (between) the rails; current flow through the armature produces a Lorentz force which causes the bucket to accelerate down the rails. Erosion of the rails by the bucket armature is a serious problem that currently limits rail guns to a small number of firings; major improvements in lifetime are required because use as a reaction engine might require on the order of 10^8 firings for a Mars mission.

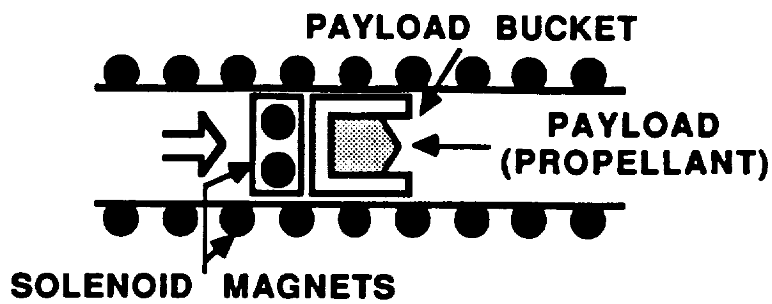
6.2 ASSUMPTIONS

In this study, the mass driver and rail gun vehicles are Solar Electric Propulsion (SEP) vehicles with a mass driver or rail gun "thruster". The vehicles are sized to give a "thruster" exhaust velocity of the propellant pellets of 12 km/s; this corresponds to an I_{sp} of 1224 lb_f-s/lb_m. For calculation purposes, the I_{sp} was assumed to be 1200 lb_f-s/lb_m so as to take into account losses or inefficiencies in the thruster system.

6.2.1 Mission Scenario

Because the MD and RG are SEP vehicles, they can operate from LEO, as shown in Fig. 6-2. Also, because any material can be used as propellant, the option of using lunar materials is considered. It is assumed that oxygen propellant is used which is supplied from either the Earth or the Moon). The oxygen is stored as a liquid and frozen into pellet form just prior to loading in the bucket. The use of solid oxygen as propellant has two advantages. First, and most important, it is a volatile pellet that will evaporate (sublime) and not present a collision hazard to other spacecraft. A second advantage, when used in a mass driver bucket, is that the solid oxygen can be used to cool the high-temperature superconducting magnet in the mass driver bucket.

MASS DRIVER



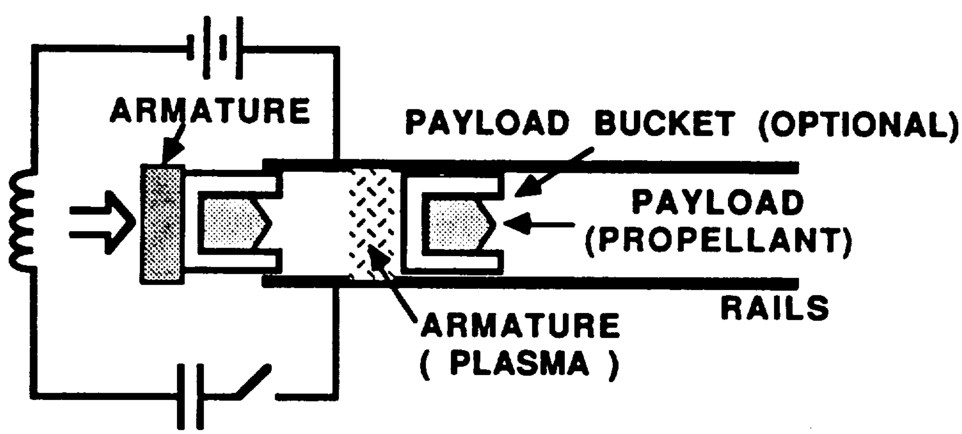
ADVANTAGES

- HIGH EFFICIENCY
- REUSABLE BUCKETS

ISSUES

- LARGE, COMPLEX SYSTEM
- LOW ACCELERATION

RAIL GUN



ADVANTAGES

- SIMPLE
- SDI INTEREST

ISSUES

- MODERATE EFFICIENCY
- RAIL EROSION
- NON-REUSABLE BUCKETS

Figure 6-1. Mass Driver and Rail Gun Propulsion Concepts

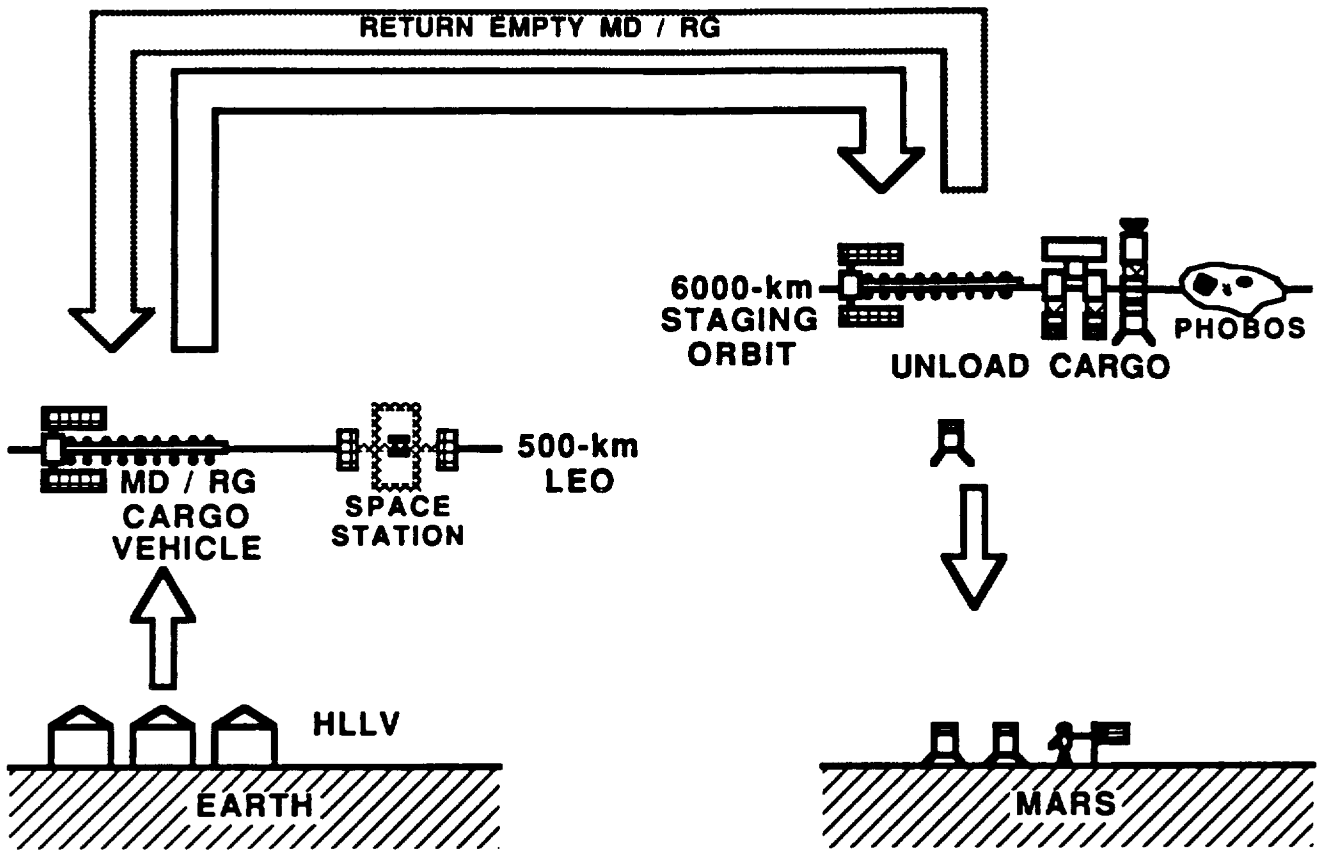


Figure 6-2. Mass Driver and Rail Gun Solar Electric Propulsion Mission Scenario

6.2.2 Vehicle Sizing

6.2.2.1 Systems Common to Both Vehicles. Subsystems common to both the mass driver and rail gun vehicles include the solar electric power system, a refrigerator for active cooling of liquid oxygen stored during transit and for freezing the liquid to produce solid oxygen pellets, and propellant tankage.

The MD and RG vehicles employ the same type of solar photovoltaic array used in the 100-MW_e SEP vehicle (see Section 3), with a specific mass of 3.6 kg/kW_e.[3] However, the MD and RG systems operate at a "bus" power level of 1 to 100 MW_e. Because a specific mass of 3.6 kg/kW_e may not be valid for systems of less than 100 MW_e, use of this low a specific mass for the MD and RG vehicles should be considered a limiting case of "optimistic" performance.

Sorption refrigerators, like those discussed in Section 1, are used for storage of liquid oxygen (LO_2) and for freezing the LO_2 to produce solid oxygen "pellets" for firing in the "thrusters". The mass of the liquid oxygen refrigerator is a function of the total propellant mass (M_p) stored. The mass of the refrigerator used to freeze the liquid oxygen is a function of the propellant mass flow rate and thus "jet" power (P_{Jet}) and I_{sp}. For the liquid oxygen refrigerator (90 K), the cooling load (W_{cool}) for a single propellant tank is:[4]

$$W_{cool}\,[W] = 0.01248 \cdot M_p^{2/3} \quad [1\ LO_2\ \text{tank, all masses in kg}]$$

For the refrigerator used to freeze the LO_2, the total cooling (freezing) load at an I_{sp} of

1200 lb_f-s/lb_m is :

$$W_{freeze}\,[W] = 198 \cdot P_{Jet}\,[MW]$$

where P_{Jet} is a function of the "bus" electric power (P_e) and the overall propulsion system efficiency, as discussed below.

The LO_2 refrigerator is a 2-stage (thermoelectric cooler, O_2 sorption cooler) sorption refrigerator designed for temperatures of 85 to 90 K; its mass is:[5]

$$M_{Frig}\,[85\text{-}90\,K] = 30.6 + 17.7 \cdot W_{cool} \quad \text{[all masses in kg]}$$

which, when combined with the LO_2 cooling load is:

$$M_{Frig}\,[85\text{-}90\,K] = 30.6 + 0.220 \cdot M_p^{2/3} \quad \text{[all masses in kg]}$$

$$M_{Frig}\,[85\text{-}90\,K] = 0.0306 + 0.0220 \cdot M_p^{2/3} \quad \text{[all masses in MT]}$$

A liquid-hydrogen temperature (20 K) refrigerator, like that described in Section 1, is used for freezing the LO_2. Its scaling equation is:[5]

$$M_{Frig}\,[20\,K] = 45.9 + 21.1 \cdot W_{cool} \quad \text{[all masses in kg]}$$

which, when combined with the LO_2 freezing load is:

$$M_{Frig}\,[20\,K] = 45.9 + 4177.8 \cdot P_{Jet}\,[MW] \quad \text{[all masses in kg]}$$

$$M_{Frig}\,[20\,K] = 0.0459 + 4.1778 \cdot P_{Jet}\,[MW] \quad \text{[all masses in MT]}$$

Finally, the LO_2 propellant tankage on the vehicles and in the propellant resupply tankers has a tankage factor of 1 % of the mass of propellant stored.

6.2.2.2 Rail Gun Vehicle. The rail gun SEP was sized based on a recent study of rail guns for launching small (1-kg sized) spacecraft from LEO.[6] Because a rail gun system is simpler than a mass driver, it was possible to develop a point design, based on exhaust velocity only, such that the rail gun "thruster" has a fixed mass of 126.151 MT and a total efficiency (electric-to-jet power) of 45 %, as shown in Table 6-1.

In the rail gun, the propellant "pellet" and "bucket" are combined into a single unit (by the loader) and ejected as propellant; a small mass of armature material would also be required and was assumed to be included in the mass contingency.

Finally, a solar photovoltaic power system, refrigerators, and propellant tankage are added to give a final RG vehicle "dry" mass scaling equation of:

$$\begin{aligned} M_{Dry}\ \text{RG Vehicle [MT]} = {} & 126.228 + \alpha_{Solar\ Array} \cdot P_e\,[MW_e] \\ & + 4.1778 \cdot \eta_{Total} \cdot P_e\,[MW_e] \\ & + 0.0220 \cdot M_p^{2/3} + 0.01 \cdot M_p \\ & \text{[all masses in MT]} \end{aligned}$$

where $\alpha_{Solar\ Array}$ is the specific mass of the solar photovoltaic power system (3.6 kg/kW_e=MT/MW_e) and η_{Total} is the total system efficiency (45 % electric-to-jet power).

The total efficiency is simply the product of the energy storage system efficiency (50 %) and rail gun launcher efficiency (90 %) such that:[6]

$$P_{Jet} = \eta_{Storage} \cdot \eta_{Launcher} \cdot P_e = \eta_{Total} \cdot P_e$$

Note that although the launcher efficiency is high, the energy storage system efficiency is low. This is due, in part, to the need to rapidly store all the energy required for a single "shot" and then discharge all this energy in a short (24 ms), intense burst to the rail gun launcher. By contrast, the mass driver energy storage system discussed below can have a much higher efficiency (95 %) because it operates more in a load-leveling mode without short, intense bursts of energy.

Table 6-1. Rail Gun Thruster Components

Component	Ref. 6	This Study
General		
Length (m)	100	144
Acceleration (m/s^2)	$5x10^5$	$5x10^5$
(10^3 Gees)	51	51
Pellet Velocity (km/s)	10	12
I_{sp} (lb_f-s/lb_m)	1000	1200
Pellet Mass (kg)	1	1
Shot Time On (ms)	20	24
Pellet Kinetic Energy (MJ)	50	72
$\eta_{Storage}$	0.5	0.5
$\eta_{Launcher}$	0.9	0.9
Energy Needed per Shot (MJ)	111.1	160.0
System Masses (MT)		
Rails	2.900	4.176
Insulator	0.800	1.152
Kevlar	1.700	2.448
Aluminum Tube	0.900	1.296
Structure & Cable	2.000	2.880
Sun Shade	0.100	0.144
Homopolar Generator	28.880	40.181
Inductor	34.500	48.000
Switches	0.200	0.200
Pellet Loader	(N/A)	0.100
Pre-Boost Gas Gun	0.100	0.100
Command/Data/Telecom	0.050	0.100
Attitude Control System	0.100	0.144
Subotal	72.330	100.921
Contingency (25 % as per Ref. 6)	18.083	25.230
Total	90.413	126.151

6.2.2.3 Mass Driver Vehicle. The mass driver SEP was sized based on a study which considered the use of a mass driver OTV to deliver large (4000 MT) payloads to geosynchronous Earth orbit (GEO).[7] This vehicle had a power level (≈ 13 MW_e) comparable to those considered in this study. The MD OTV had a mass driver "thruster" specific mass of 2 to 20 kg/kW_e and an assumed total efficiency (electric-to-jet power) of 80%; this range of specific masses is reflected in the analyses below.

Figure 6-3 illustrates that the total efficiency is a function of the accelerator, de-accelerator, and energy storage system efficiencies, and the relative masses of the propellant "pellet" and bucket. For example, there is the electrical-to-kinetic energy efficiency of the accelerator section in accelerating the bucket and propellant pellet to a velocity of 12 km/s. The kinetic energy of the empty bucket is recovered in the de-accelerator section because the bucket is slowed and returned to the start of the accelerator section for re-loading and re-use. The de-accelerator section is assumed to have the same efficiency as the accelerator section, but because the bucket is empty, less energy is available for recovery. Also, the energy storage system has its own charge/discharge cycle efficiency. However, unlike the rail gun which must discharge all of its energy in a short, intense pulse, the mass driver energy storage system can operate more efficiently in a load-leveling mode as a full bucket is accelerated and an empty bucket is de-accelerated. Thus, the total efficiency of 80 % assumed here could correspond to an efficiency of 95 % in the accelerator, de-accelerator, and energy storage systems, and an empty bucket mass approximately equal to the ejected pellet mass, as shown in Fig. 6-3. The use of high-efficiency (99 %) superconducting magnets in the accelerator and de-accelerator systems, combined with a 95-% efficient energy storage system, would permit bucket masses two to three times the pellet mass.

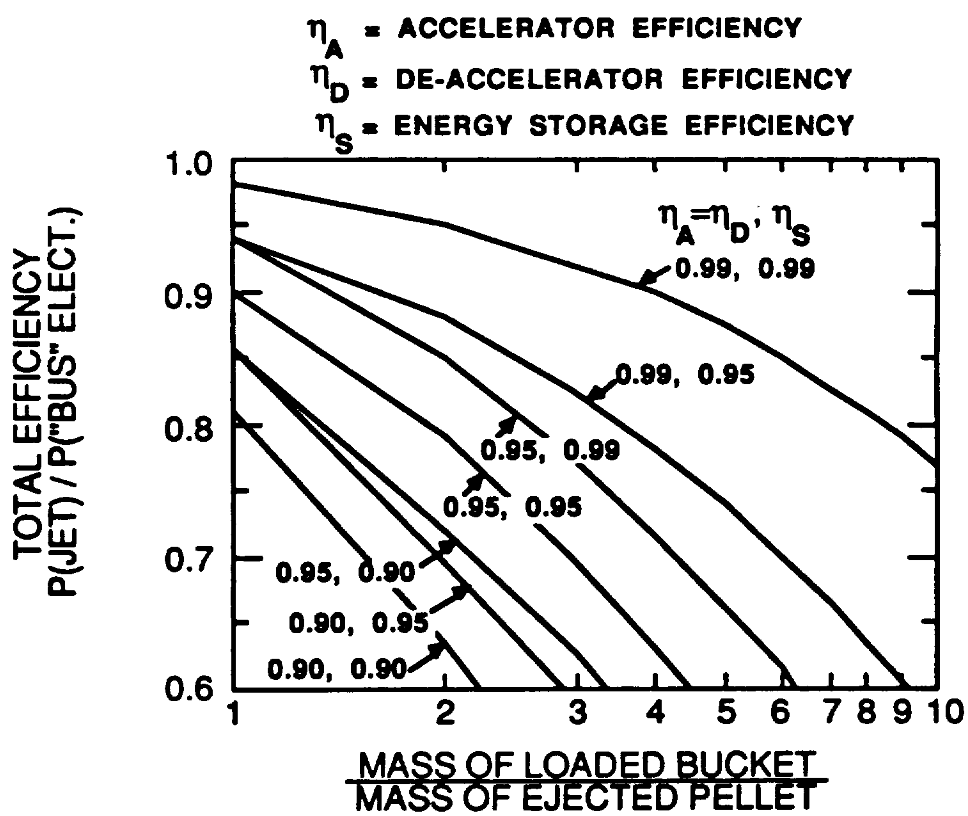


Figure 6-3. Mass Driver Efficiency

The "thruster" specific mass was derived from the data presented in Ref. 7 for a LEO-to-GEO OTV with a delivery time of 180 days and an exhaust velocity of 12 km/s. The total system specific mass and "bus" electric power ranged from 10 to 30 kg/kW_e and 12.7 to 14.0 MW_e, respectively. The solar power system mass (3.6 kg/kW_e) and the propellant tankage ($0.01 \cdot M_p$) mass were subtracted from the total vehicle "dry" mass to give a total "thruster" system (structure, accelerator, de-accelerator, energy storage, switches, radiators, loader, command/data/telecommunications, and attitude control systems) specific mass of 2.64 to 18.57 kg/kW_e based on a total system efficiency of 80 %.

When the various subsystems (including refrigerators) are combined, the MD vehicle "dry" mass scaling equation becomes:

$$M_{Dry}\ \text{MD Vehicle [MT]} = 0.077 + \alpha_{Solar\ Array} \cdot P_e\,[MW_e]$$
$$+ 4.1778 \cdot \eta_{Total} \cdot P_e\,[MW_e]$$
$$+ \alpha_{Thruster} \cdot P_e\,[MW_e] + 0.0220 \cdot M_p^{2/3} + 0.01 \cdot M_p$$
$$\text{[all masses in MT]}$$

where $\alpha_{Solar\ Array}$ is the specific mass of the solar photovoltaic power system (3.6 $kg/kW_e = MT/MW_e$), $\alpha_{Thruster}$ is the specific mass of the mass driver "thruster" system (2 to 20 $kg/kW_e = MT/MW_e$), and η_{Total} is the total system efficiency (80 % electric-to-jet power) such that:

$$P_{Jet} = \eta_{Total} \cdot P_e$$

6.2.3 Trajectory Analysis

As with the 100 MW_e SEP vehicle, Carl G. Sauer Jr. of JPL provided the low-thrust trajectory analysis.[8] The results of the analysis for an SEP vehicle with an I_{sp} of 1200 $lb_f\text{-}s/lb_m$ are listed in the Appendix.

6.2.4 Support Infrastructure

6.2.4.1 Lunar Oxygen Production Infrastructure. The use of lunar materials requires an infrastructure to produce and deliver the lunar material to LEO. For this study, it was assumed that the required infrastructure (lunar processing factory and Moon-to-LEO transportation system) was already in place and that the lunar propellant (liquid oxygen) would be "free" for the MD/RG vehicles. Thus, no additional mass penalty was incurred by the MD/RG system for the need to produce and deliver lunar propellants. Note that such a lunar propellant production system might be possible using a variety of processing schemes and transportation options. Several transportation options are possible which require no propellants from the Earth. For example, the liquid oxygen could be launched from the Moon using a mass driver catapult and mass "catcher" in Earth orbit.[2] Another option would be the use of a nuclear thermal rocket using lunar-produced liquid oxygen propellant (rather than the usual hydrogen propellant).[9] Alternatively, if sufficient quantities of hydrogen were found on the Moon, chemical (O_2/H_2) OTV-type vehicles could transport the lunar materials to LEO without requiring any propellant from Earth for their operations.[9]

6.2.4.2 Payload Support. As with the 100-MW_e SEP (and NEP), the option of splitting the payload among several MD/RG vehicles is considered. Also, like the SEP and NEP systems, it is assumed that the MD/RG vehicles are too large to dock directly at the LEO or

Mars-orbit base, so an Orbital Maneuvering Vehicle (OMV) is added to each payload for Earth and Mars orbital rendezvous and docking.

6.3 RESULTS

Figures 6-4 through 6-7 summarize the results for the mass driver and rail gun systems. The first, and fundamental, result is that the I_{sp} of an MD/RG system is too low for a low-thrust round-trip Mars mission; the total IMLEO using propellant from Earth is excessive. By contrast, an MD/RG system using "free" lunar propellant can exhibit significant reductions in IMLEO, especially for mass drivers with very low "engine" specific masses (2 kg/kW_e excluding tankage, refrigerations, and power supply).

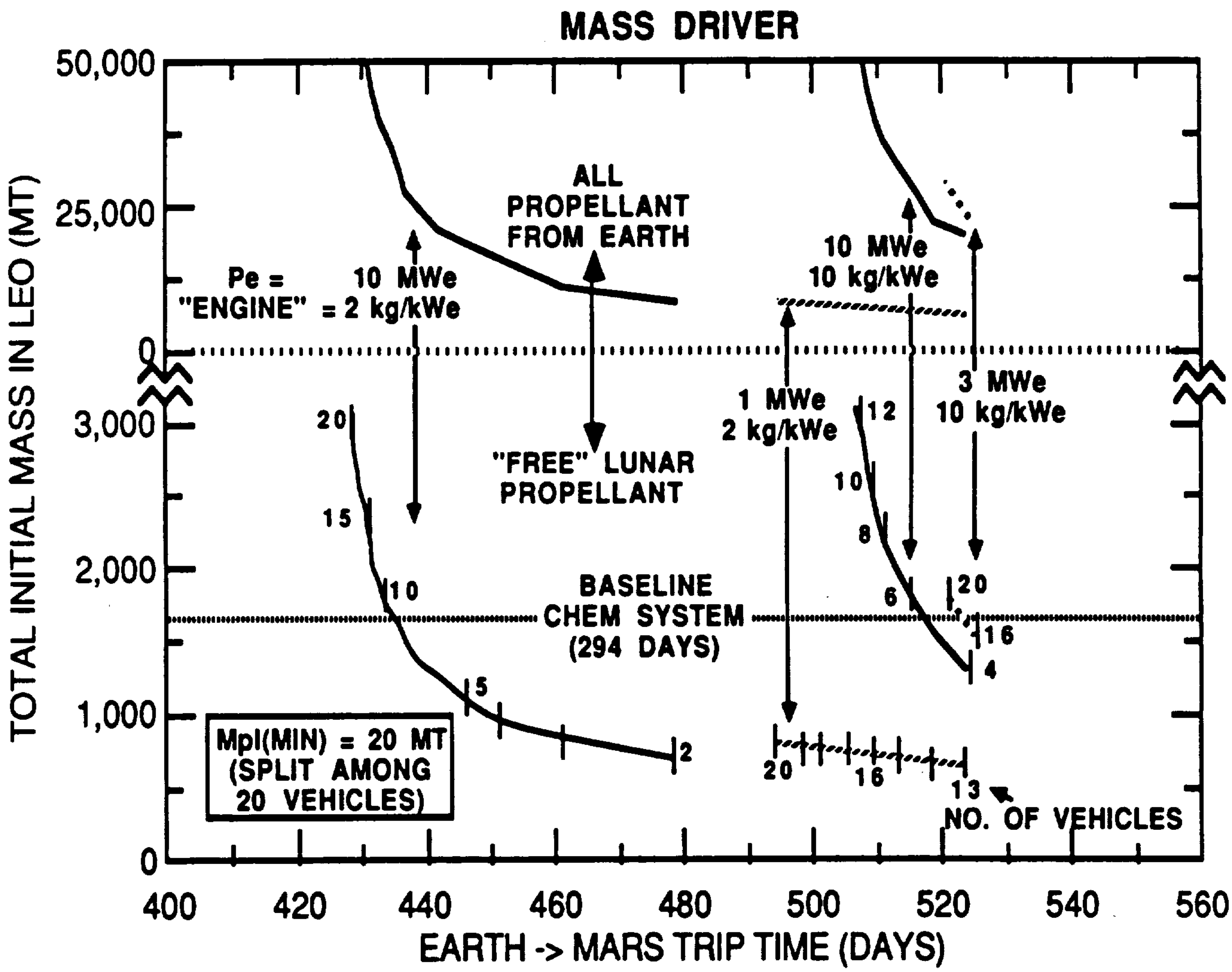


Figure 6-4. Initial Mass in Low Earth Orbit vs. Earth-to-Mars Trip time for Mass Driver Propulsion

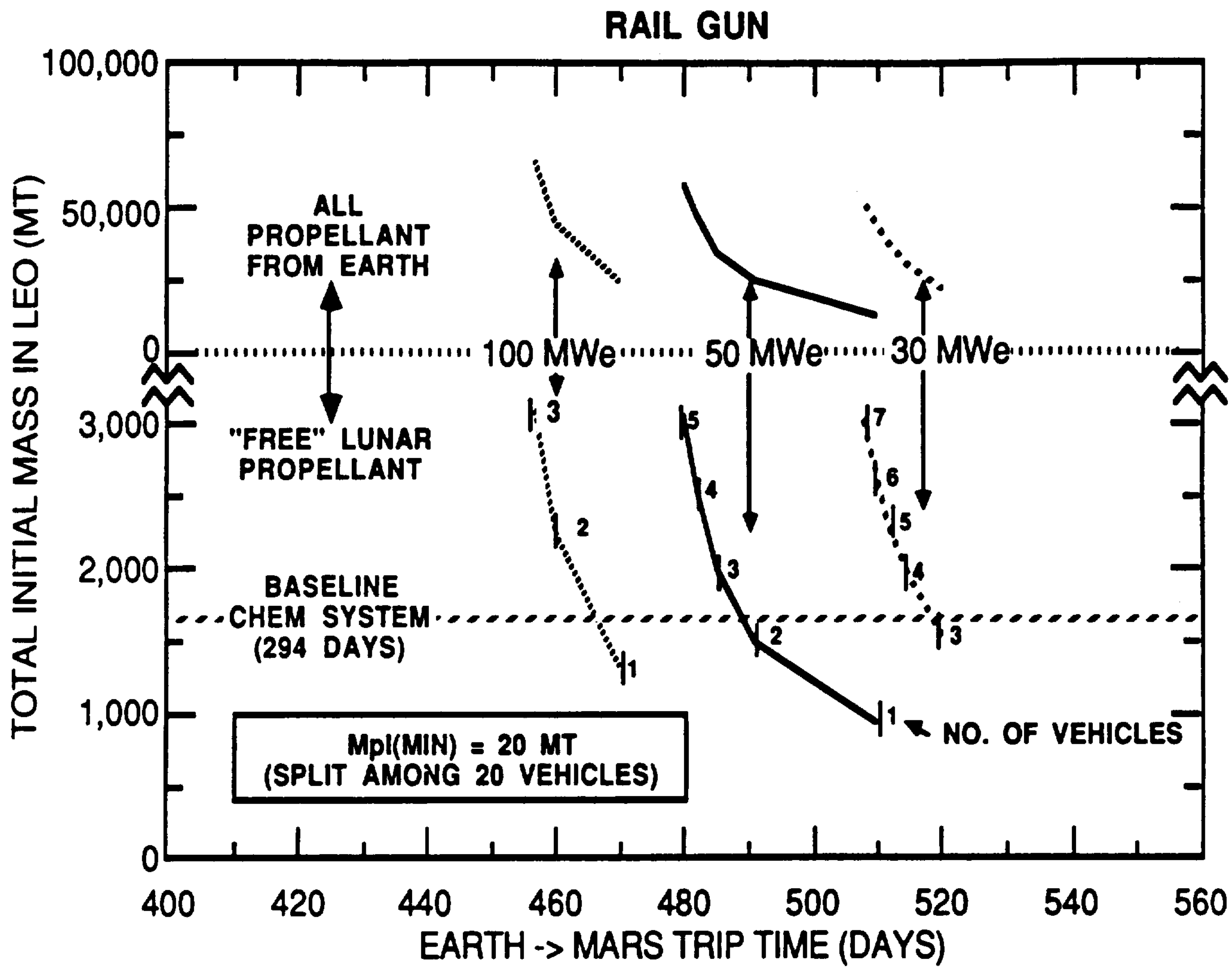


Figure 6-5. Initial Mass in Low Earth Orbit vs. Earth-to-Mars Trip Time for Rail Gun Propulsion

However, this low IMLEO is somewhat misleading because copious amounts of lunar oxygen are required. Thus, the attractiveness of the MD/RG system depends on the "cost" of producing and transporting lunar materials to LEO. In this analysis, it was assumed that the lunar material would be "free"; however, the large amounts of propellants required for this mission using MD/RG systems could easily outstrip the capacity of a lunar facility designed to support only cis-lunar operations. Thus, a Mars cargo mission using MD/RG systems could require construction of a dedicated lunar oxygen production and transportation system, which would be infrastructure chargeable to the IMLEO of the MD/RG system.

A second important result is that the high efficiency of the mass driver (80 % versus 45 % for the rail gun) allows an MD vehicle to operate with smaller power supplies than a RG system. For example, mass drivers can operate at power levels as low as 1 MW_e; by contrast, rail guns operate at 30 MW_e and above, with an optimum power (minimum IMLEO) around 50 MW_e.

A third result, illustrated in Figs. 6-6 and 6-7, is that the performance of these

vehicles is strongly dependent on the assumed tankage factor. For example, in this study, a tankage factor of 1 % was assumed; by contrast, a 5 % tankage factor for liquid oxygen has been assumed in previous studies.[9,10] In fact, the MD and RG vehicles, even with "free" lunar oxygen propellant, are not competitive with the baseline aerobraked chemical (O_2/H_2) vehicle option for even a 2 % LO_2 tankage factor.

Finally, the Earth-to-Mars trip times for both MD and RG systems are around 500 days, or about 70 % greater than the baseline chemical vehicle option.

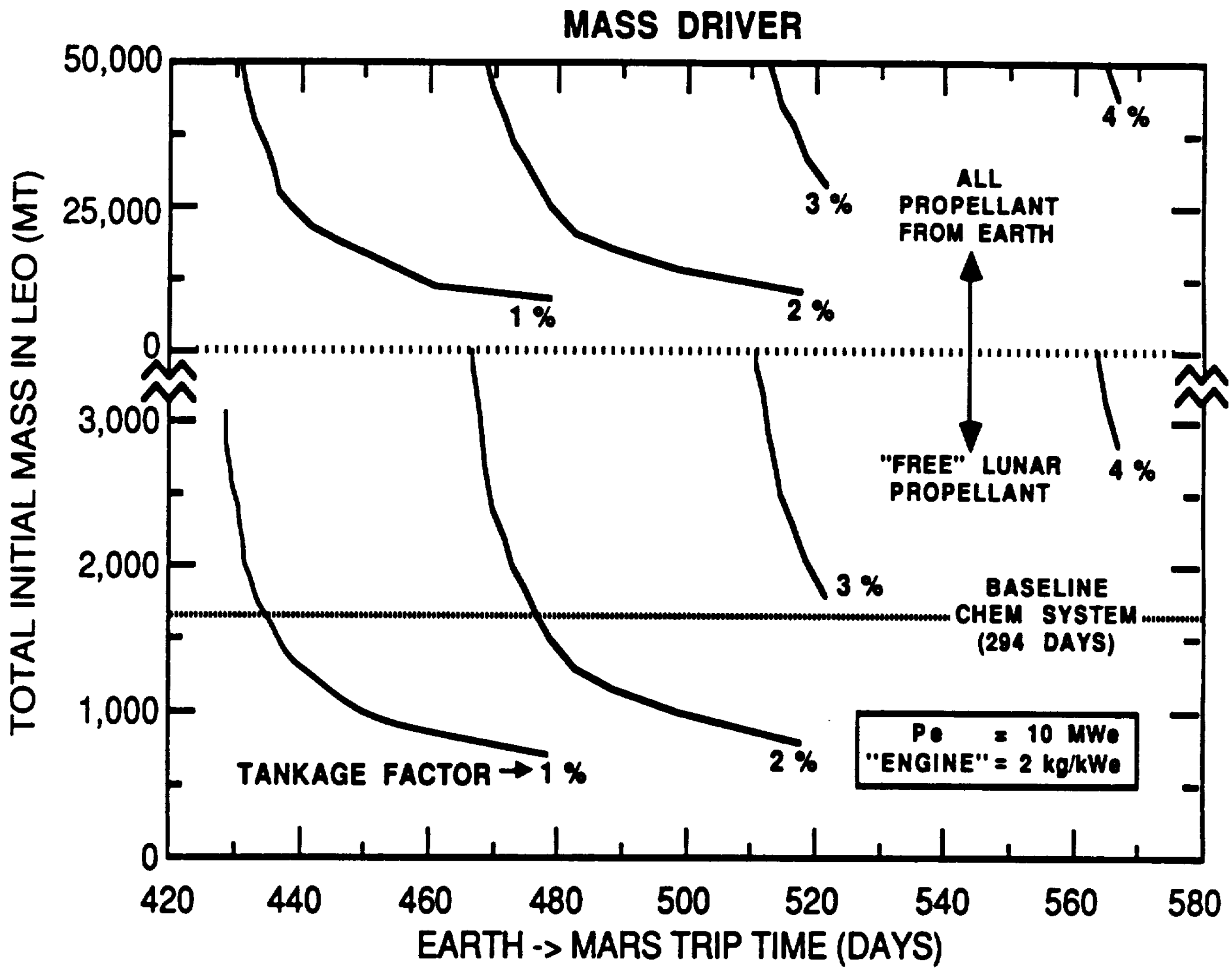


Figure 6-6. Initial Mass in Low Earth Orbit vs. Earth-to-Mars Trip time for Mass Driver Propulsion as a Function of the Propellant Tankage Factor

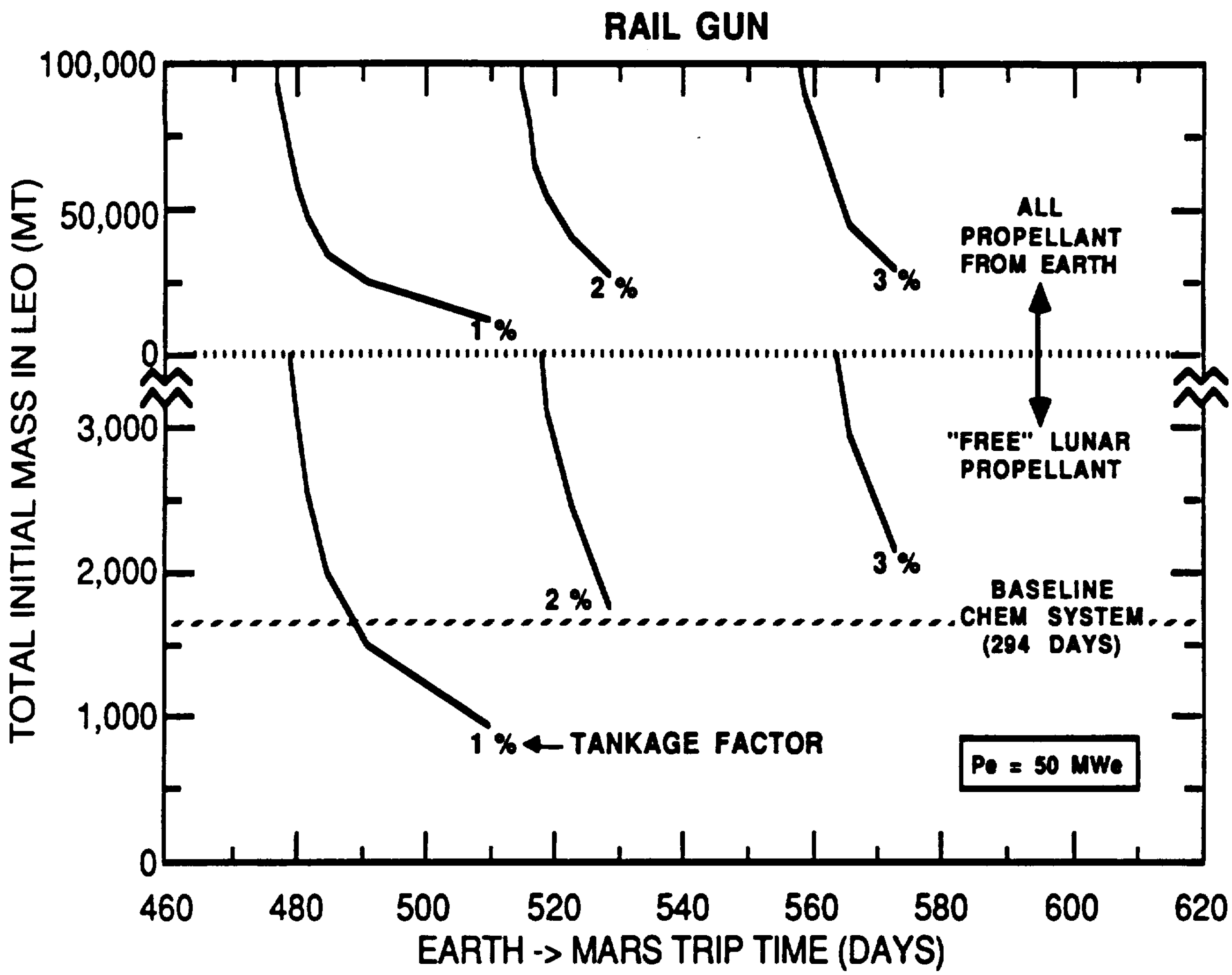


Figure 6-7. Initial Mass in Low Earth Orbit vs. Earth-to-Mars Trip Time for Rail Gun Propulsion as a Function of the Propellant Tankage Factor

6.4 CONCLUSIONS

Although MD/RG systems have a low I_{sp} for a round-trip Mars mission, their ability to use extraterrestrial materials (e.g., lunar oxygen) as propellant mass can result in significant savings in IMLEO if the propellant is "free". Because the quantities of propellant are large for the MD/RG systems, a dedicated lunar production and transportation system (to deliver the lunar material to LEO) may be required. The requirements of a large-scale lunar oxygen production/transportation infrastructure should be addressed in future studies, not only for MD/RG systems, but also for chemical (O_2/H_2) and other systems (e.g., oxygen-propellant nuclear-thermal rockets). Similarly, a Phobos propellant production facility should be evaluated for propellant production for the Mars-to-Earth leg of the trip.

The high efficiency of the MD system greatly improves its performance as compared to the RG system. Thus, although the RG is currently receiving attention from the Strategic Defense Initiative Office, the RG does not appear to be attractive for the Mars

cargo mission. However, the superior IMLEO of the MD system is dependent upon its assumed specific mass and efficiency; a detailed design study should be performed to assess the performance parameters of the various subsystems (especially the "thruster" specific mass and the propellant tankage factor) in the MD vehicle. One area that should be considered is the use of high-temperature superconducting materials for the solenoid coils and in the bucket magnet to improve performance.

6.5 REFERENCES

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SECTION 7

BEAMED-ENERGY PROPULSION

7.1 INTRODUCTION

In beamed-energy propulsion, a remote transmitter source, which could be on the ground or in space, transmits power via an intense beam of electromagnetic radiation (near-visible or microwave wavelengths) to a spacecraft. There, the beam is collected and used to power the propulsion system. In beamed-energy propulsion, there is the potential for significant weight reduction, and thus improved performance on the spacecraft, because the heavy power supply is left back at the transmitter station.

Two different wavelength regions (near-visible and microwave) are considered. These can then be used directly in a thermal propulsion system, or indirectly in an electric propulsion system, by first converting the incoming beamed energy into electricity,[1] as shown in Fig. 7-1. Thus, a total of four beamed-energy propulsion concepts were considered in this study, as shown in Fig. 7-2; these are discussed next.

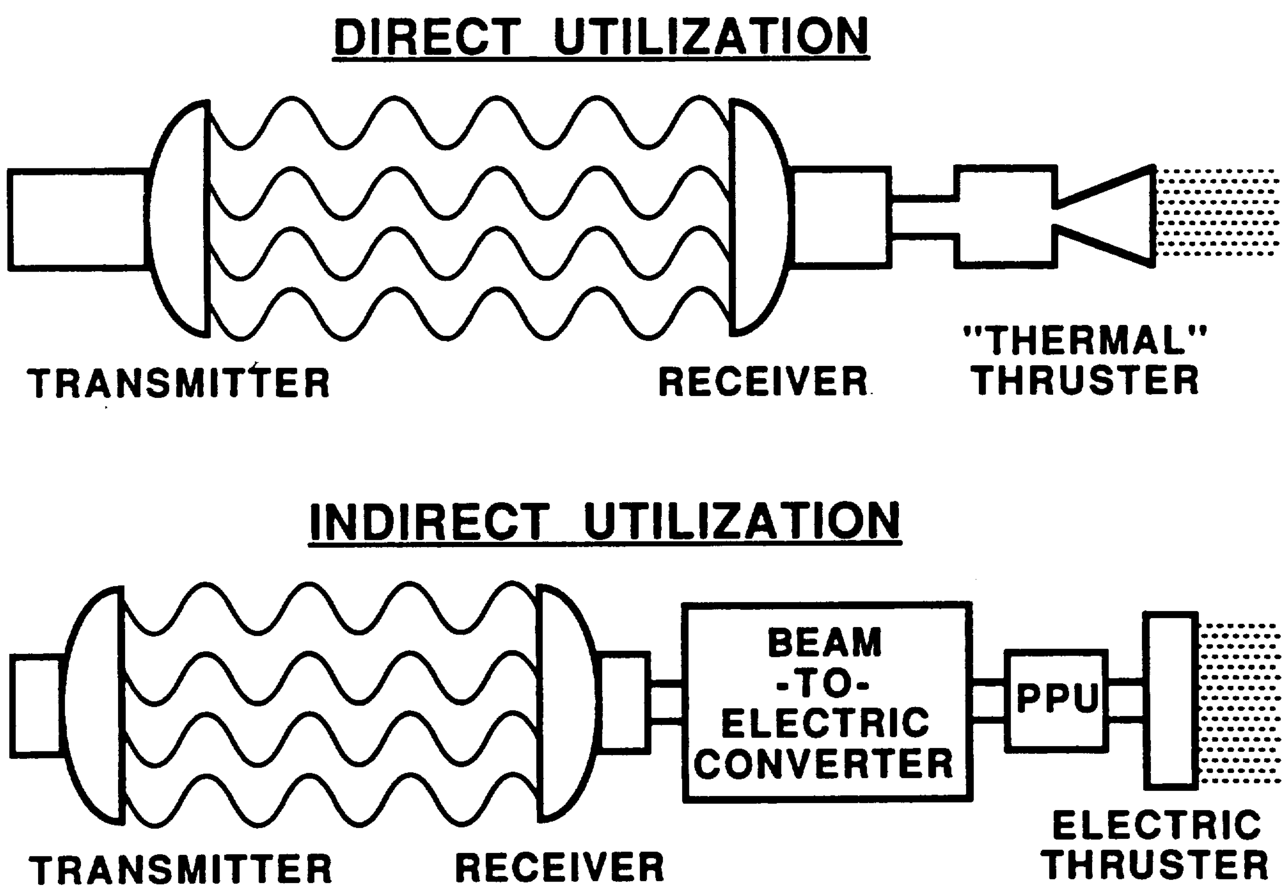


Figure 7-1. Beamed Energy Utilization Concepts for Propulsion

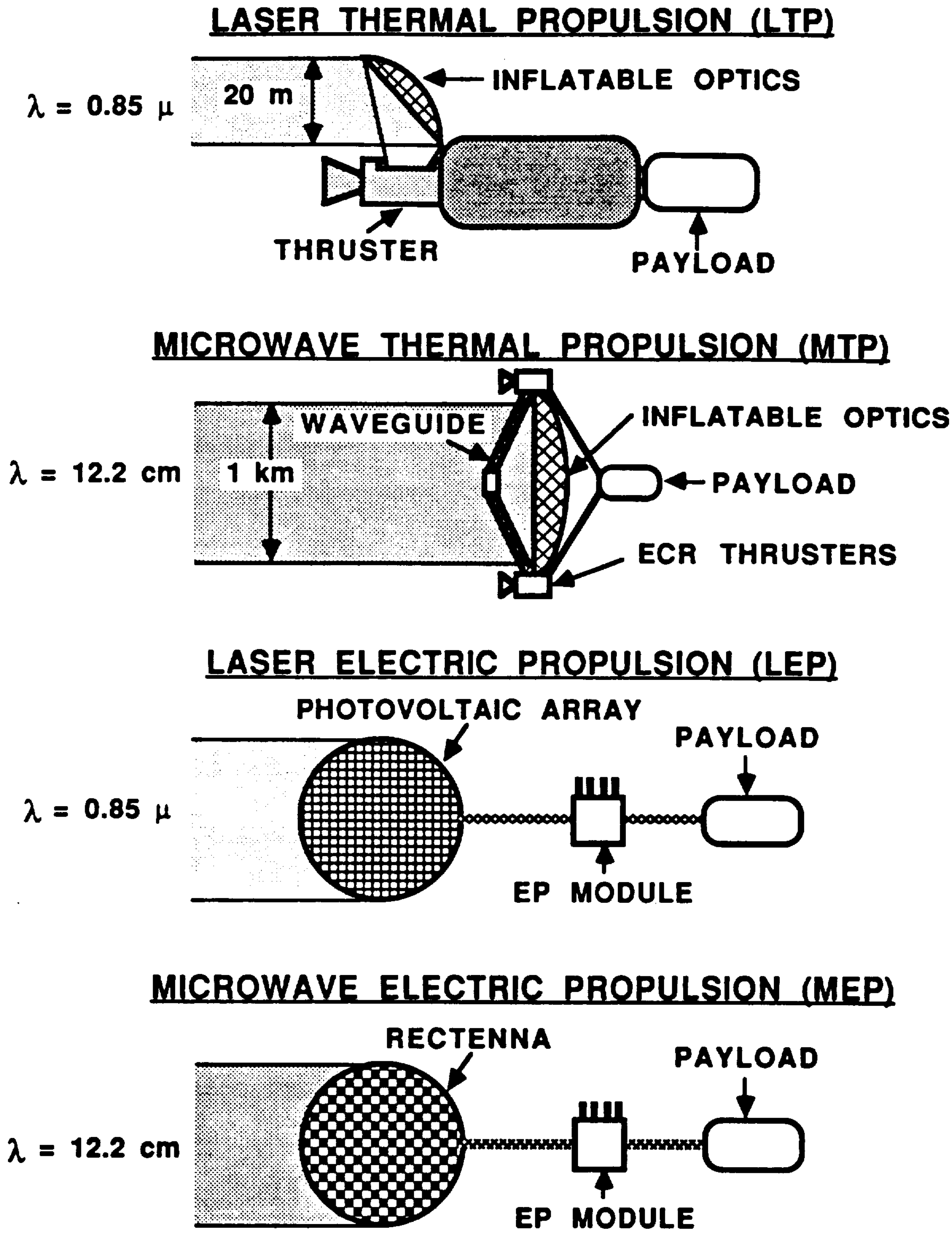


Figure 7-2. Beamed Energy Propulsion Vehicle Concepts

7.1.1 Beamed-Energy Propulsion Concepts

The first system is Laser Thermal Propulsion (LTP); this concept is similar to the Solar Thermal Propulsion (STP) concept to be discussed in Section 8. In the LTP

concept, visible or infrared (VIS/IR) laser light is focused into a thruster to heat a propellant such as hydrogen. Because the beam "spot" intensity is higher in laser thermal propulsion, it is possible to couple the beam energy directly into the propellant to permit a higher I_{sp} than that from solar thermal propulsion (e.g., 1500 versus 1200 lb_f-s/lb_m, respectively).[2]

The second system is Microwave Thermal Propulsion (MTP). One type of MTP thruster is that can be used is an analog of the LTP thruster in that microwave energy is focused into a thruster to excite and heat a propellant. However, a different microwave energy coupling mechanism can be employed, involving electron-cyclotron resonance (ECR); this has the potential for providing I_{sp}s comparable to electric propulsion concepts (e.g., around 5000 lb_f-s/lb_m).[1] The ECR thruster is used in this study on the MTP system.

The third system is Laser Electric Propulsion (LEP). In this concept, a "solar" photovoltaic cell array is illuminated with laser light. The laser light is tuned to the correct wavelength to excite the photovoltaic array (e.g., 0.85 micrometers [microns] for gallium arsenide cells). The laser beam can have a much higher intensity than the 1.345 kW/m^2 of sunlight at 1 AU, thus resulting in an effectively lower solar array specific mass. Note that this is basically a Solar Electric Propulsion (SEP) system, but with laser light instead of sunlight.[3-5]

The final system is Microwave Electric Propulsion (MEP). In this concept, a rectenna (rectifying antenna) is used to convert microwaves to electricity, which is then used to power electric thrusters as in an SEP system.[1]

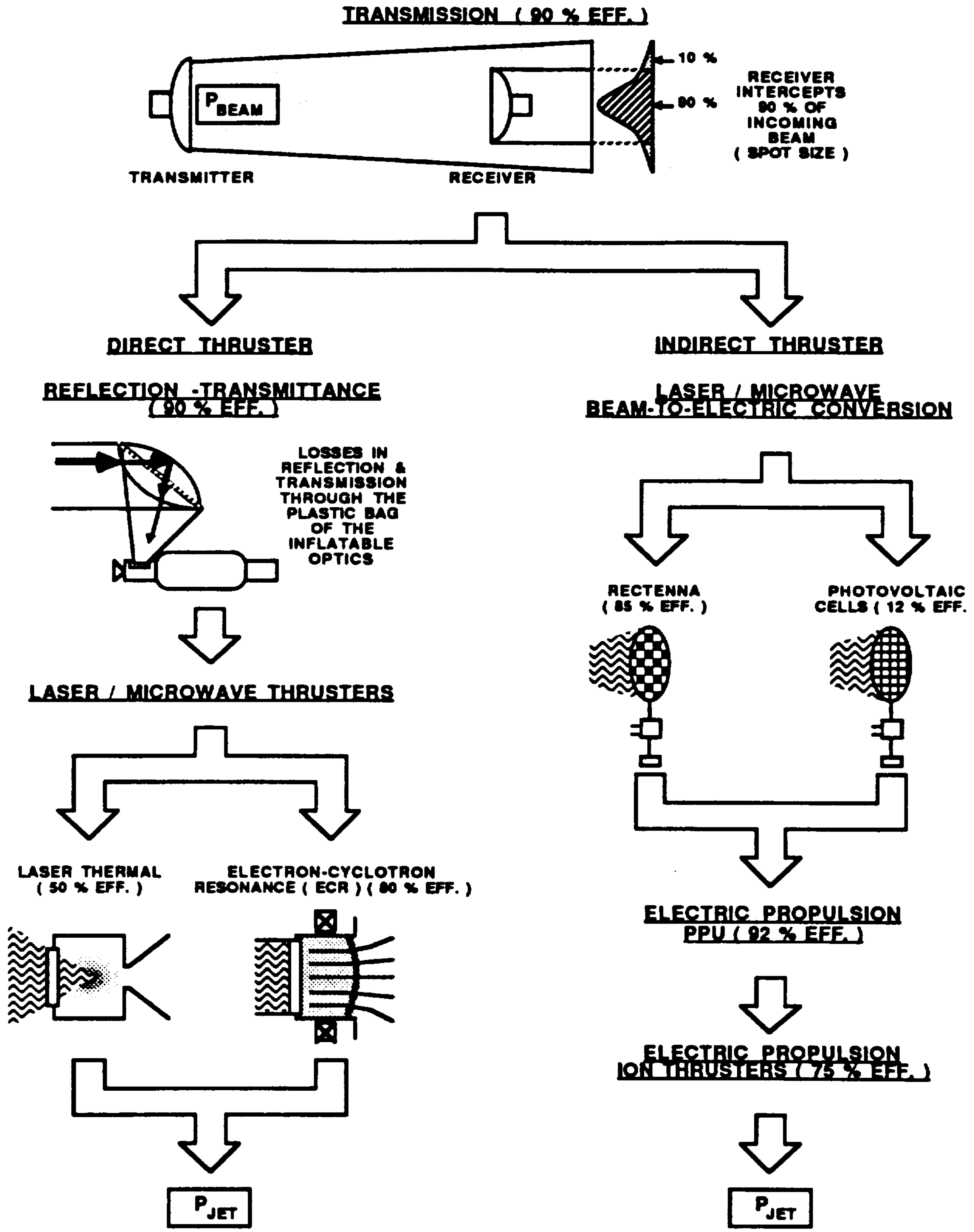
Each vehicle will have different efficiencies for collection of the beamed energy (P_{Beam}) and conversion of the beamed energy into propulsive "jet" power (P_{Jet}), as illustrated in Fig. 7-3. Beam powers were allowed to vary parametrically from 1 to 100 MW, although past studies have considered near-term steady-state laser beam powers only up to 10 MW.[2] (Note that lasers for strategic defense may operate at very high power levels, but are required to operate for only short periods of time; by contrast, lasers for orbit raising must operate continuously for years.) Also, lasers operating in the 10-MW range would require power supplies in the 100- to 1,000-MW range, because most lasers are on the order of 1 to 10 % efficient;[2] the requirement for a 100-MW power supply for a space-based laser would represent a second "new" technology (in addition to the laser transmitter). Thus, beam powers in excess of 10 MW should not be considered as a near-term option for missions early in the next century, although major improvements resulting from research by the Strategic Defense Initiative Office could change this assumption.

7.1.2 Optics Considerations

For the ideal limit of diffraction limited optics, the product of the diameters of the transmitter and receiver optics varies as the product of the wavelength and the beaming distance:[1]

$$\eta = 1 - \exp\{(-A_t \cdot A_r)/(\lambda^2 \cdot L^2)\}$$

where η is the fraction of beam power which finally impinges on the receiver under the influence of diffraction, A_t and A_r are the areas of the transmitter and receiver, respectively, λ is the wavelength of the transmitted beam, and L is the distance between the transmitter and receiver. Note that this relationship represents an ideal limit for values of η less than about 95 %; the effect of non-idealities, such as beam jitter, may significantly increase the size of the optics required.[2]



VIS-IR (LASER THERMAL)

$P_{JET} = 0.9 \cdot 0.9 \cdot 0.5 \cdot P_{BEAM} = 0.40 \cdot P_{BEAM}$

MICROWAVE (ECR)

$P_{JET} = 0.9 \cdot 0.9 \cdot 0.8 \cdot P_{BEAM} = 0.65 \cdot P_{BEAM}$

VIS-IR (PHOTOVOLTAIC CELL)

$P_{JET} = 0.9 \cdot 0.12 \cdot 0.92 \cdot 0.75 \cdot P_{BEAM} = 0.08 \cdot P_{BEAM}$

MICROWAVE (RECTENNA)

$P_{JET} = 0.9 \cdot 0.85 \cdot 0.92 \cdot 0.75 \cdot P_{BEAM} = 0.53 \cdot P_{BEAM}$

Figure 7-3. Beamed Energy Propulsion Efficiencies

The variation in transmitter and receiver size with wavelength and beaming distance is shown in Fig. 7-4, which assumes interception of 90 % of the beam distribution from diffraction-limited optics. It is seen that microwave-based systems are limited to beaming distances corresponding to geosynchronous Earth orbit (GEO). For example, a 10-km diameter transmitter is required for a 1-km diameter receiver for GEO distances at 12.2-cm wavelengths (2.45 GHz). These dimensions represent a practical upper limit to space-based receivers and ground-based transmitters.

For near-visible light wavelengths (e.g., 0.85 microns), beaming to lunar distances may be possible, but this requires receivers on the order of 25 m in diameter. This places severe demands on the adaptive optics transmitters,[2] although much of the required technology is being pursued by the Strategic Defense Initiative Office.

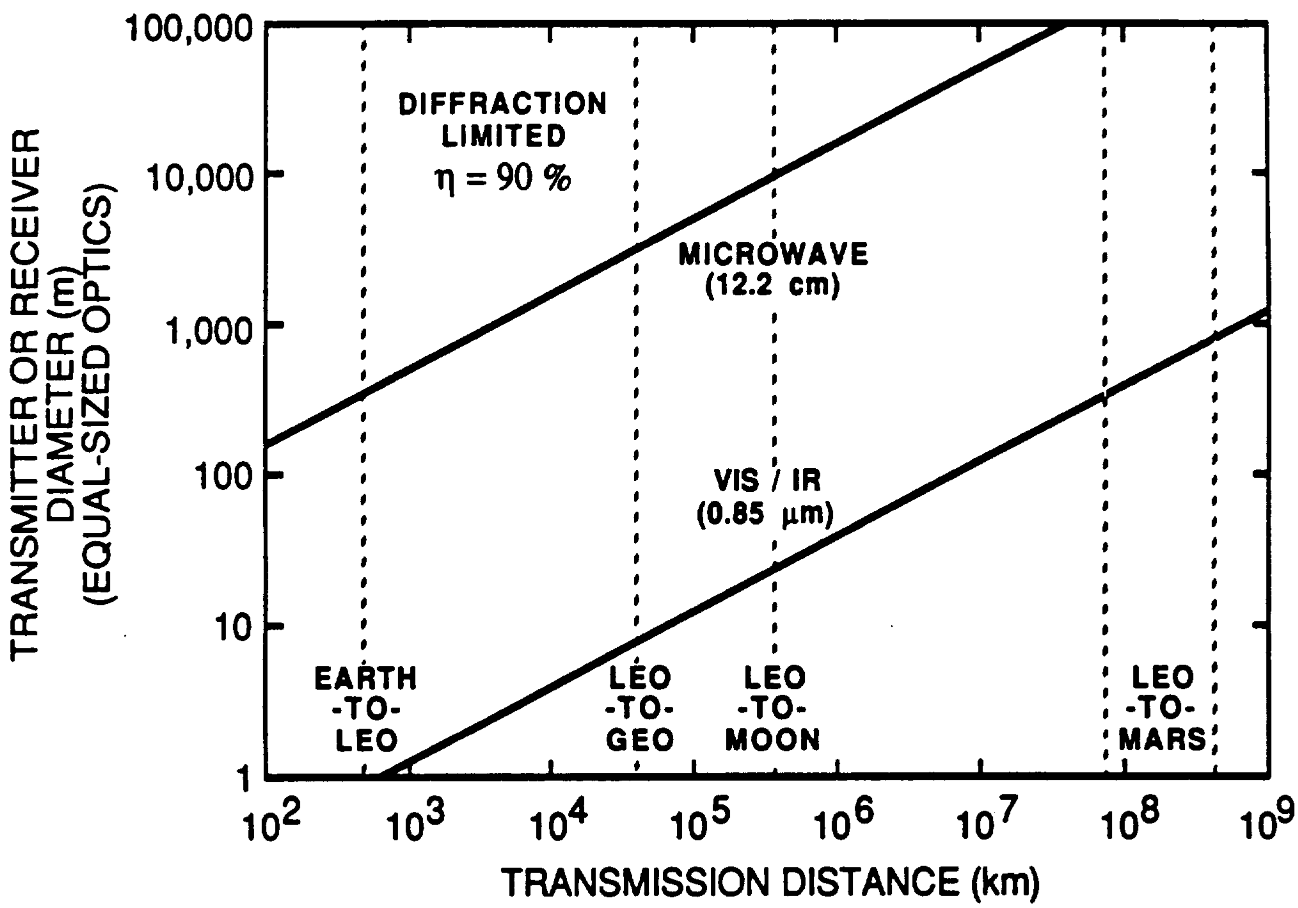


Figure 7-4. Diameter of Transmitter or Receiver Optics vs. Transmission Distance and Wavelength (Equal-Sized Optics)

7.2 ASSUMPTIONS

7.2.1 Mission Scenarios

As illustrated in Figs. 7-5 to 7-8, the beamed-energy propulsion vehicles were used as Orbit Transfer Vehicles (OTV) to carry the cargo and an aerobraked chemical O_2/H_2 stage to a high orbit (GEO altitude). From there, the chemical stage injects the cargo and

itself to Mars. Departure from a high orbit can reduce the required Delta-V for trans-Mars injection; the high-I_{sp} beamed-energy propulsion system is used to provide much of the total Delta-V that is required to escape the Earth's deep gravity well and still meet the transmission distance limitations discussed above.

Note that the laser electric propulsion system can be used either in an OTV mode or in a laser-augmented SEP mode to perform the whole mission. In this option, shown in Fig. 7-9, a laser beam is used near the Earth to increase the power of the system for a "fast" Earth escape (or capture on the return trip). Outside the range of the laser beam, the vehicle reverts back to its normal SEP mode for the remainder of the trip to Mars.

Finally, all the OTVs are assumed reusable with the exception of the LTP vehicle. It is assumed disposable (one-way delivery mode only), because it is a relatively small, "simple" thermal rocket vehicle. Note that the MTP vehicle is also a "thermal" rocket vehicle, but its large size (1-km diameter receiver optics) dictated its reuse. Assumptions specific to each vehicle are discussed next.

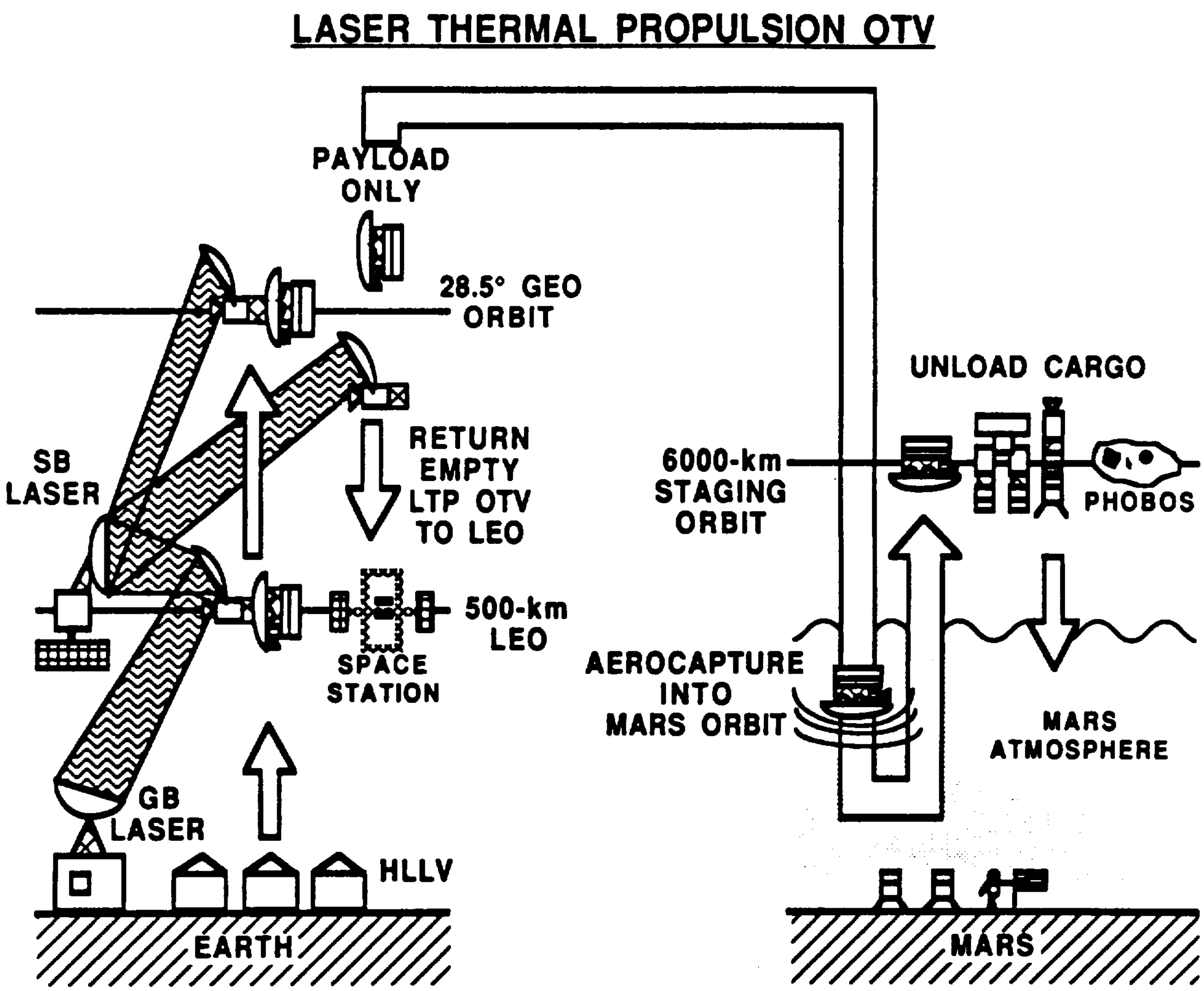


Figure 7-5. Laser Thermal Propulsion OTV Mission Scenario

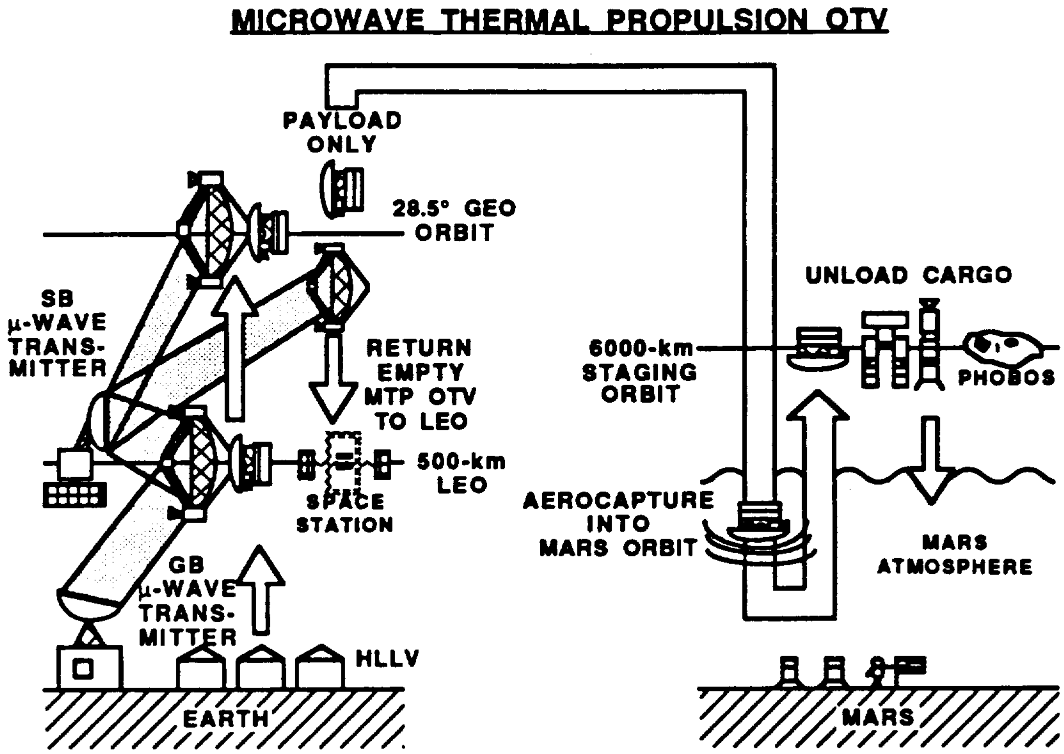


Figure 7-6. Microwave Thermal Propulsion OTV Mission Scenario

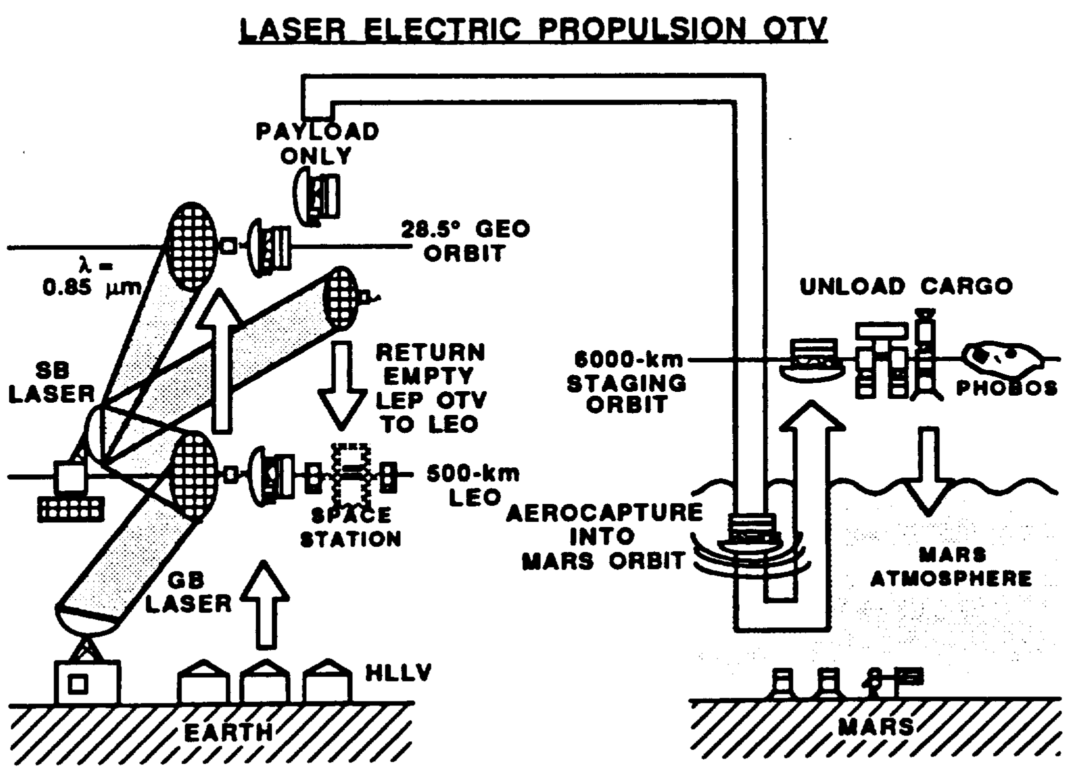


Figure 7-7. Laser Electric Propulsion OTV Mission Scenario

MICROWAVE ELECTRIC PROPULSION OTV

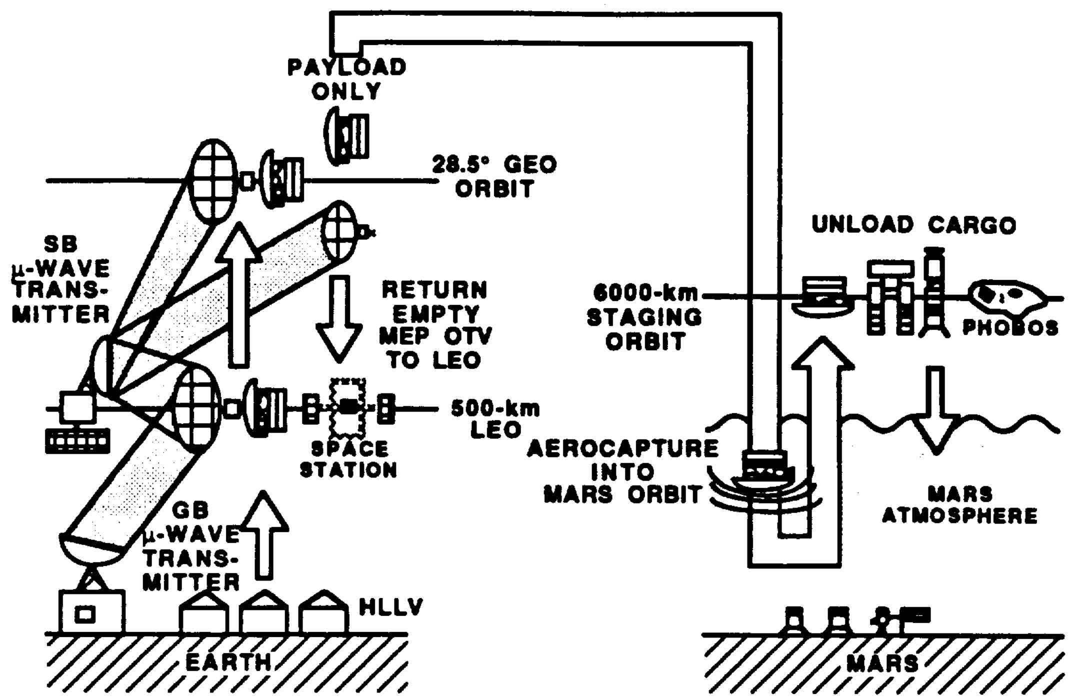


Figure 7-8. Microwave Electric Propulsion OTV Mission Scenario

LASER-AUGMENTED SOLAR ELECTRIC PROPULSION

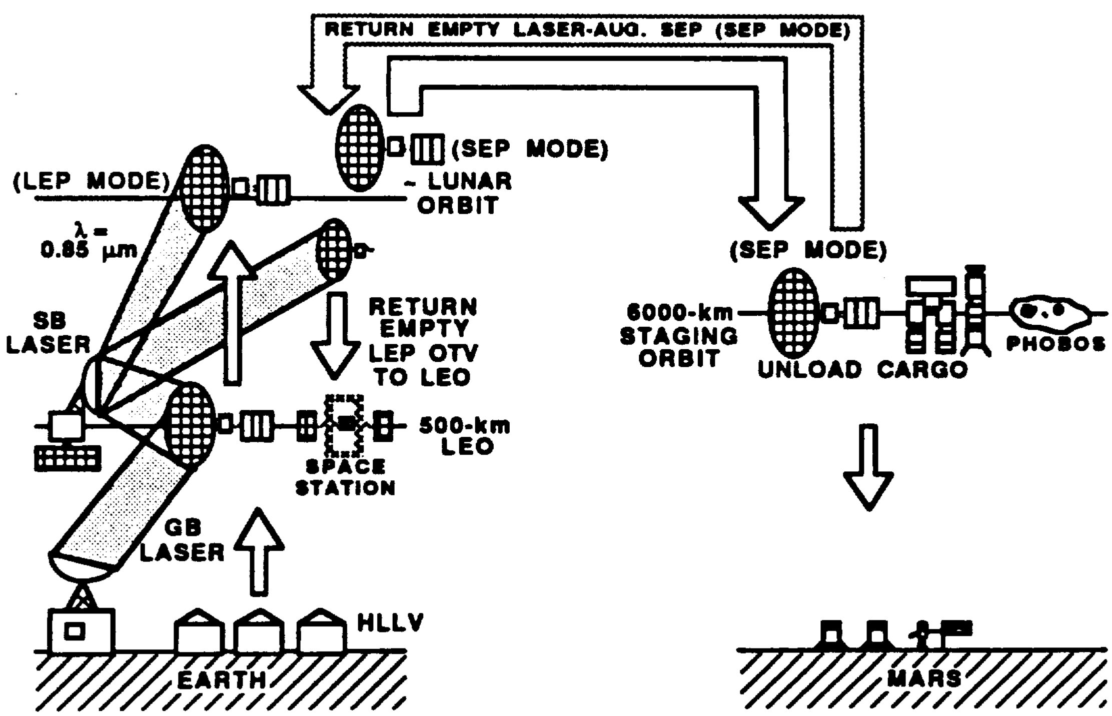


Figure 7-9. Laser-Augmented Solar Electric Propulsion Mission Scenario

7.2.2 Laser Thermal Propulsion (LTP) Vehicle Sizing

7.2.2.1 Inflatable Optics. The LTP vehicle operates at a wavelength of 0.85 microns. This wavelength was chosen to be compatible with that selected for the LEP vehicle, which is in turn driven by the requirements of the its photovoltaic arrays (see below). The optics for all the vehicles are sized assuming a "spot" size that intercepts 90 % of the incoming beam.

An inflatable parabolic reflector, like that shown in Fig. 7-10,[6] is used to collect the incoming laser beam and focus the beam into a laser-thermal thruster; it is also used on the STP vehicle. Inflatable optics like these have been investigated for use as radar antennas and for solar- and laser-thermal propulsion systems. The areal density of the inflatable optics, including gas pressurization systems, rip-stop seams, structural torus, and reflecting membranes ranges from 0.08 to 0.03 kg/m^2 for diameters from 10 to 1,000 m.[6-9] A value of 0.03 kg/m^2, appropriate for reflectors larger than 10 m in diameter, is assumed in this study.

The inflatable optics are sized for a 20-m diameter receiver "spot" size on the OTV[2] which requires 3.468-m diameter diffraction-limited transmitter optics for GEO distances. However, the inflatable optics are at a 45° angle relative to the incoming laser beam. Thus, the actual shape of the inflatable optics is an ellipse with a semi-major axis √2 times the semi-minor axis, for a total area √2 times that of the circular "spot" area. The mass of the inflatable optics is thus:

$$M_{\text{Optics}}\,[\text{kg}] = 1.414 \cdot \text{pi} \cdot (\text{20-m Diameter}/2)^2 \cdot (0.03\ \text{kg/m}^2) = 13.3\ \text{kg}$$

The efficiency of the optics is a measure of the energy lost due to reflection and absorption by the transparent membranes. Estimates of this efficiency range from 80 to 92 %.[10-12] A value of 90 % is assumed in this study.

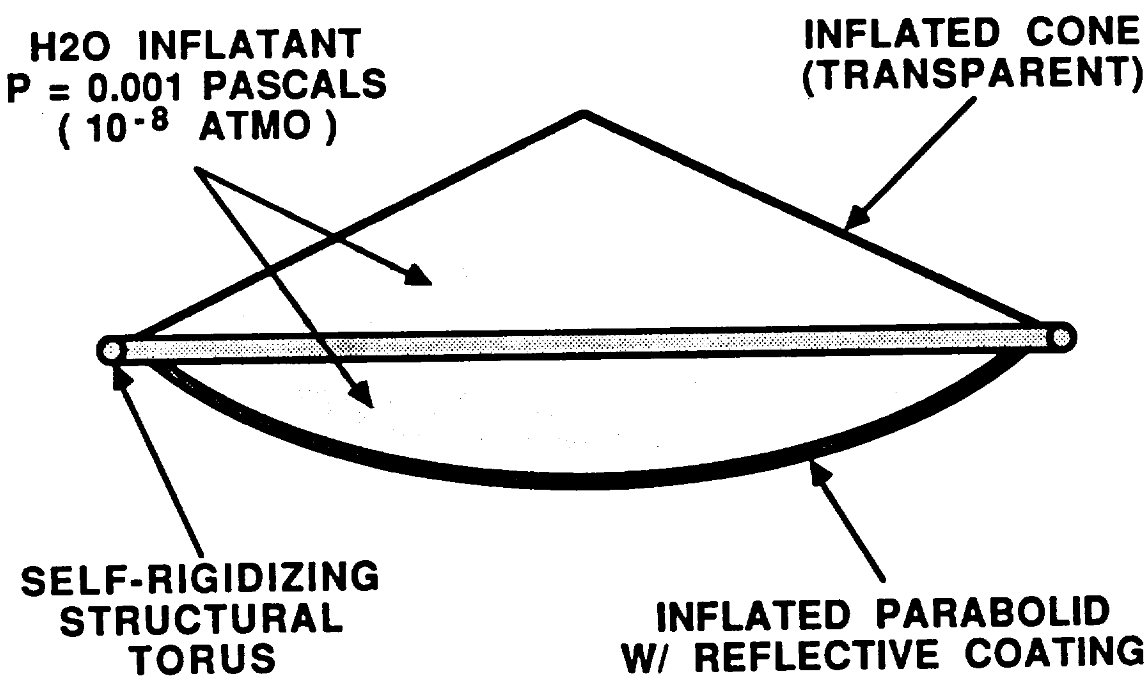


Figure 7-10. Inflatable Optics Concept (From Ref. 6)

7.2.2.2 Laser Thermal Thruster. There are four general mechanisms for coupling the energy in the laser beam to the propellant working fluid (typically hydrogen).[2] The simplest one uses a heat exchanger to absorb the light and then heat the propellant; this method is limited by the heat exchanger's material temperature limits to specific impulses of around 1100 lb_f-s/lb_m. Higher values of specific impulse can be achieved by directly focusing the laser beam into the thruster, as shown in Fig. 7-11, and seeding the propellant with opaque solid or liquid absorbant particles. This may permit specific impulses of 1200 to 1500 lb_f-s/lb_m; however, this introduces the risk of a seedant particle sticking to the window and causing a failure due to a localized hot spot. This can be avoided by using a gaseous seedant which has a molecular resonance absorption line that coincides with the wavelength of the laser. Because the gas temperature is not limited by the need to maintain the seedant in a solid or liquid state, an I_{sp} of 1500 lb_f-s/lb_m may be achievable. Finally, if the laser beam is focused to an intensity of about 25 kW/cm^2, inverse Bremsstrahlung absorption can occur due to breakdown in the gas at these high intensities. This could result in an I_{sp} as high as 2500 lb_f-s/lb_m; however, a value of 1500 lb_f-s/lb_m is considered more realistic. In this study, an Isp of 1500 lb_f-s/lb_m was assumed which could be achieved by either of the latter two mechanisms discussed above. Finally, estimates of thruster efficiency (laser beam-to-jet power) of 10 to 90 % have been considered in previous studies;[2,11,13] a 50-% efficient thruster was assumed here.

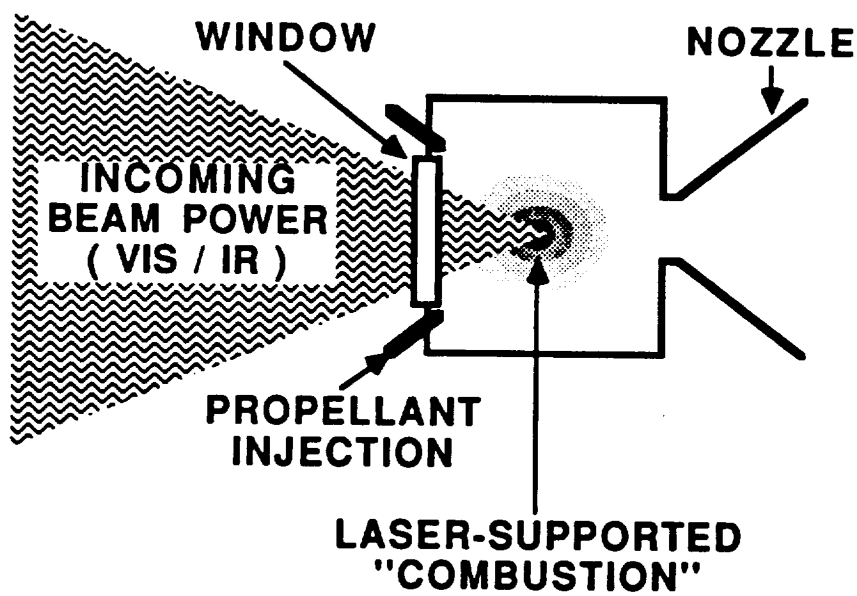


Figure 7-11. Laser Thermal Thruster

The mass scaling equation for the laser-thermal thruster[2] is based on one derived for a solar-thermal thruster[10] because of the similarities between the two types of thrusters. The scaling equation is:

$$M_{Thruster}\,[kg] = 0.125 \cdot (\,Thrust\,[N]\,)^{1.15}$$

where the thrust is a function of the "jet" power (P_{Jet}) and I_{sp}:

$$Thrust\,[N] = 2 \cdot (\,P_{Jet}\,[W]\,) / \{\,g_c \cdot (\,I_{sp}\,[\,lb_f\text{-s}/lb_m]\,)\,\}$$

Thus, the thruster mass becomes:

$$M_{Thruster}\,[MT] = 2203.4 \cdot \{ P_{Jet}\,[MW] / (g_c \cdot I_{sp}\,[lb_f\text{-}s/lb_m]) \}^{1.15}$$

where g_c is 9.8 m/s^2. The jet power depends on the total beam power collected and on the efficiencies (η) of the thruster and inflatable optics:

$$P_{Jet} = P_{Beam} \cdot \eta_{Spot} \cdot \eta_{Optics} \cdot \eta_{Thruster} = P_{Beam} \cdot 0.9 \cdot 0.9 \cdot 0.5$$

7.2.2.3 Other Systems. The vehicle has a structural weight, including attitude control thrusters and tankage for the liquid hydrogen (LH_2) propellant, of:

$$M_{Tankage} = 1572.5 + 0.2584 \cdot M_p \qquad \text{[all masses in kg]}$$

where M_p is the useable propellant mass and the tankage factor of 0.2584 is based on a LH_2 tank model of a 4-m diameter cylindrical tank with ellipsoidal ends operating at 20 psia (30 psia maximum).[14]

Because liquid hydrogen propellant is used, an active refrigeration system is added to the vehicle's dry weight. The mass scaling equation for a three-stage 20-K sorption refrigerator,[15-17] like that described in Section 1 for the baseline aerobraked chemical (O_2/H_2) stage, is:

$$M_{Frig}\,[20\,K] = 45.9 + 21.1 \cdot W_{cool} \qquad \text{[all masses in kg]}$$

where W_{cool} is the refrigeration load in Watts. For a single LH_2 tank:[14]

$$W_{cool} = 0.08226 \cdot (M_p)^{2/3} \qquad \text{[1-tank } LH_2 \text{ stage, all masses in kg]}$$

Thus, the refrigerator mass for a 1-tank LH_2 stage is:

$$M_{Frig}\,[20\,K] = 45.9 + 1.736 \cdot (M_p)^{2/3} \qquad \text{[all masses in kg]}$$

7.2.2.4 Final Vehicle Sizing. The final LTP vehicle "dry"weight (in kg) is the sum of the tankage, refrigerator, thruster, and optics masses:

$$\begin{aligned} M_{Dry}\ \text{LTP OTV Vehicle w/ } LH_2 \text{ Frig} =\ & 1618.4 + 0.2584 \cdot M_p + 1.736 \cdot (M_p)^{2/3} \\ & + 0.27739 \cdot \{ P_{Jet}\,[W] / (g_c \cdot I_{sp}\,[lb_f\text{-}s/lb_m]) \}^{1.15} \\ & + 1.414 \cdot pi \cdot (20\text{-m Dia.} / 2)^2 \cdot (0.03\ kg/m^2) \\ & \text{[all masses in kg]} \end{aligned}$$

The corresponding equation in metric tons is:

$$\begin{aligned} M_{Dry}\ \text{LTP OTV Vehicle w/ } LH_2 \text{ Frig} =\ & 1.6184 + 0.2584 \cdot M_p + 0.1736 \cdot (M_p)^{2/3} \\ & + 2203.4 \cdot \{ P_{Jet}\,[MW] / (g_c \cdot I_{sp}\,[lb_f\text{-}s/lb_m]) \}^{1.15} \\ & + 1.414 \cdot pi \cdot (20\text{-m Dia.} / 2)^2 \cdot (0.03\ kg/m^2) / 1000 \\ & \text{[all masses in MT]} \end{aligned}$$

Finally, a LH_2 propellant resupply tanker, with a tankage factor of 0.2584, is included in the total initial mass in LEO.

7.2.3 Microwave Thermal Propulsion (MTP) Vehicle Sizing

7.2.3.1 Inflatable Optics. The MTP vehicle employs large (1-km diameter) inflatable optics to focus microwaves into waveguides which transmit the beam to two thrusters.[1] As with the LTP system, the inflatable collector has an areal density of 0.03 kg/m^2, a "spot" size efficiency of 90 %, and a reflection/transmission efficiency of 90 %. For GEO distances, a 1-km diameter receiver using 12.2-cm wavelength (2.45 GHz, S-band) microwaves requires transmitter optics almost 10 km in diameter. Note that the 12.2-cm wavelength has been chosen in past studies of microwave power beaming due to its excellent transmission through the atmosphere, which is of particular importance for a ground-based transmitter for a space-based MTP OTV. Higher frequencies (e.g., 5.8 or 22.125 GHz) could be employed, but this introduces problems with transmission through the atmosphere.[1]

A 1-km diameter reflector is the largest size that has been studied for inflatable optics.[6] This large size is driven by the need for reasonable-sized optics for both the transmitter and receiver. However, because of the large optics on the vehicle, it is necessary to have a vehicle configuration in which the large receiver optics are placed at the center of the vehicle, as shown in Fig. 7-2. This has the advantage of allowing the optics to be rotated and oriented such that it is always perpendicular to the incoming beam, rather than at a 45° angle as for the LTP optics. However, this has the disadvantage of requiring two sets of thrusters and waveguides to "pipe" the microwaves from the focal point of the collector optics to each of the two thrusters.[1] By contrast, the smaller (20-m diameter) receiver of an LTP vehicle can be placed off to the side of the vehicle and a single centrally-located thruster used.

The scaling equation for the inflatable optics is:

$$M_{Optics}\,[\mathrm{kg}] = \mathrm{pi} \cdot (\,1{,}000\text{-m Diameter} / 2\,)^2 \cdot (\,0.03\ \mathrm{kg/m^2}\,) = 23{,}562\ \mathrm{kg}$$

In fact, it will be shown that the mass of the inflatable optics dominates the total vehicle "dry" weight.

The waveguide is assumed to also be an inflatable structure with the same areal density as the inflatable optics. A waveguide with a square cross section 12.2-cm on a side is assumed. The optics are assumed to have a focal length of one-half the optics diameter (i.e., the focal length equals the radius of the optics), so the length of each waveguide is $\sqrt{2}$ times the radius of the optics. Thus, the two waveguides have a total mass of:

$$M_{Waveguide}\,[\mathrm{kg}] = 4 \cdot (\,0.122\ \mathrm{m}\,) \cdot 1.414 \cdot (\,1000\ \mathrm{m}\,) \cdot (\,0.03\ \mathrm{kg/m^2}\,) = 20.7\ \mathrm{kg}$$

7.2.3.2 Electron Cyclotron Resonance (ECR) Thruster. The ECR thruster[1,18] circumvents the thermal/I_{sp} limits of the laser-thermal thruster by employing electron-cyclotron resonance plasma acceleration to couple the incoming microwave energy to the propellant (e.g., hydrogen) and produce thrust. Also, in contrast to electric propulsion thrusters, it is an electrodeless device, and thus holds the promise of long operating lifetimes and the ability to use a variety of propellants, including those derived from extraterrestrial sources (e.g., lunar oxygen). Figure 7-12 shows a conceptual schematic of an ECR thruster in which circularly-polarized microwave radiation ionizes the propellant in an energizing zone. The applied diverging magnetic field (which decreases in intensity

in the downstream direction) causes the free electrons in the plasma to move in cyclotron orbits about the field lines. The radii of these orbits are determined by the magnetic field strength. This electron-cyclotron motion produces a magnetic dipole field opposed to the applied field, which in turn produces a net body force on the plasma that accelerates the plasma out of the thruster.

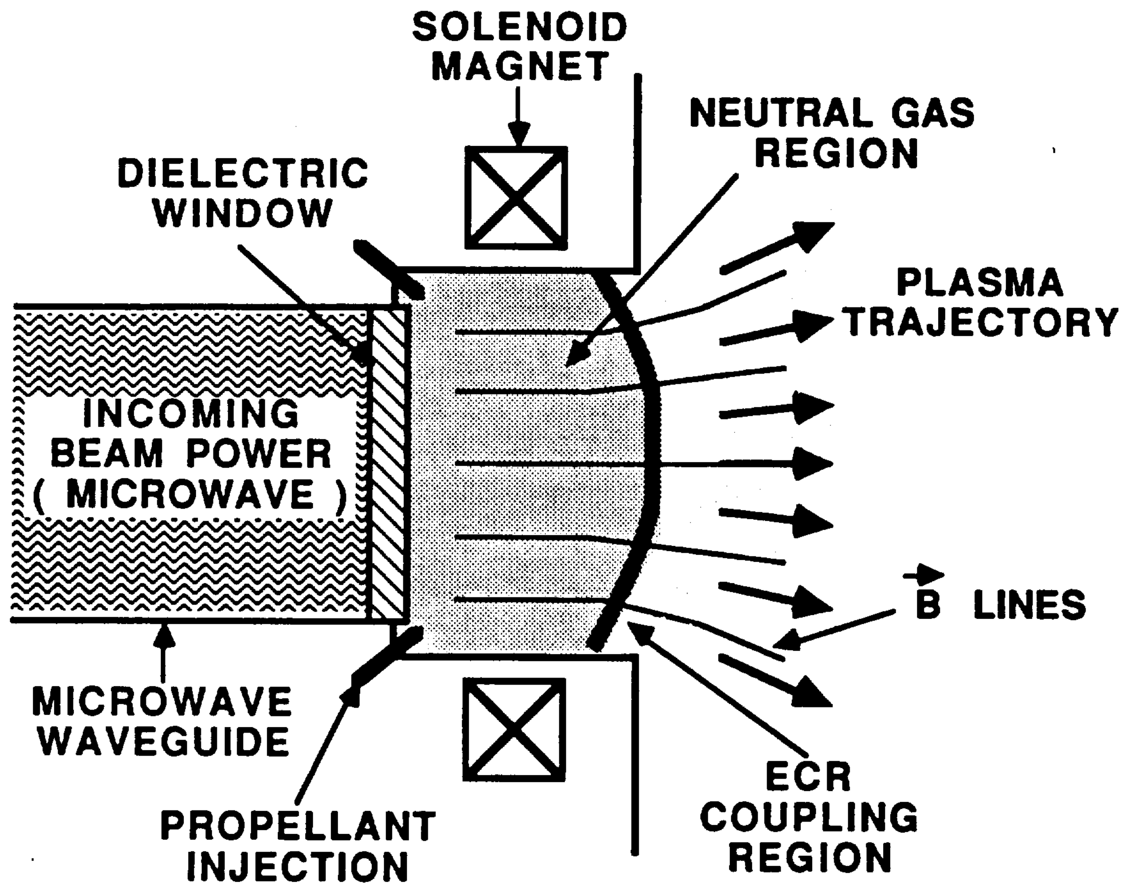


Figure 7-12. Electron Cyclotron Resonance (ECR) Thruster

This thruster concept is still in the basic research stage,[18] but it holds the promise of achieving high specific impulses and efficiencies (50 to 85 %). For this study, an I_{sp} of 5000 lb_f-s/lb_m and an efficiency (microwave beam-to-jet power) of 80 % were assumed.[1] The mass of an ECR thruster is estimated to be:

$$M_{Thruster}\ [kg] = 5.57 \cdot (\ P_{Jet}\ [kW]\ )^{2/3}$$

Each engine is sized to use one-half the total power, because two engines are required at opposite ends of the vehicle. Thus, the assumed scaling equation, modified for the case of two thrusters, is:

$$M_{Thrusters}\ [MT] = 2 \cdot 0.557 \cdot \{\ (\ P_{Jet}\ [MW]\ )\ /\ 2\ \}^{2/3}$$

As with the laser-thermal thruster, the jet power of the ECR thruster depends on the total beam power collected and on the efficiencies (η) of the thruster and inflatable optics:

$$P_{Jet} = P_{Beam} \cdot \eta_{Spot} \cdot \eta_{Optics} \cdot \eta_{Thruster} = P_{Beam} \cdot 0.9 \cdot 0.9 \cdot 0.8$$

7.2.3.3 Other Systems. The same liquid hydrogen tankage, refrigerators, etc. assumed for the LTP vehicle are used in the MTP vehicle.

7.2.3.4 Final Vehicle Sizing. As in the LTV vehicle, the total "dry" mass of the MTP vehicle is the sum of the optics (including waveguides), thrusters, tankage, and refrigerators:

$$M_{Dry\ MTP\ OTV\ Vehicle\ w/\ LH_2\ Frig} =$$

$$1.6184 + 0.2584 \cdot M_p + 0.1736 \cdot (M_p)^{2/3}$$

$$+ 2 \cdot 0.557 \cdot \{ (P_{Jet} [MW]) / 2 \}^{2/3}$$

$$+ \text{pi} \cdot (1{,}000\text{-m Diameter} / 2)^2 \cdot (0.03 \text{ kg/m}^2) / 1000$$

$$+ 4 \cdot (0.122 \text{ m}) \cdot 1.414 \cdot (1000 \text{ m}) \cdot (0.03 \text{ kg/m}^2) / 1000$$

[all masses in MT]

For the specific case of 1-km diameter inflatable optics and waveguides (23.5827 MT), this reduces to:

$$M_{Dry\ MTP\ OTV\ Vehicle\ w/\ LH_2\ Frig\ \&\ 1\text{-km Optics}} =$$

$$25.2011 + 0.2584 \cdot M_p + 0.1736 \cdot (M_p)^{2/3}$$

$$+ 2 \cdot 0.557 \cdot \{ (P_{Jet} [MW]) / 2 \}^{2/3}$$

[all masses in MT]

Note that the mass of the inflatable optics tends to dominate the "dry" mass of the vehicle for 1-km diameter optics. Smaller optics could provide major savings in vehicle "dry" weight, but only at the cost of increased transmitter optics sizes or higher microwave frequencies (shorter wavelengths), although the use of high-frequency microwave radiation introduces problems with transmission through the atmosphere. Finally, as with the LTP vehicle, a LH_2 propellant resupply tanker, with a tankage factor of 0.2584, is included in the total initial mass in LEO.

7.2.4 Laser Electric Propulsion (LEP) Vehicle Sizing

7.2.4.1 Laser-to-Electric Power Conversion (Photovoltaic Array). The LEP vehicle is basically an SEP vehicle. The photovoltaic array requires a laser beam wavelength of 0.85 microns which corresponds to the band gap of the gallium arsenide arrays assumed here.

For this discussion, photovoltaic cells are divided into two categories, solar and unconventional.[3-5] Conventional solar photovoltaic cells are those that have been developed for use on satellites and spacecraft and are optimized for efficient conversion of normal solar radiation. "Unconventional" photovoltaic cells are used in this context to describe a conceptual class of photovoltaic devices that would be optimized for very efficient conversion of very high intensities of incident radiation of a specific wavelength. This discussion is limited to conversion of near-visible (ultraviolet-to-infrared) laser light

sources to electricity; conversion of microwaves to electricity is discussed below. The peak of the sun's spectral emissive power occurs from 400 to 700 nanometers (nm) in the visible spectral region. Implicit in the artificial division of photovoltaic cells into "solar" and "unconventional" is the assumption that the bandgap of the unconventional cell semiconductor is significantly larger or smaller than 400 to 700 nm (i.e., ultraviolet radiation ranges from 10 to 390 nm, and infrared radiation ranges from 770 to 10^6 nm). This assumption of wavelength mismatch means that the unconventional photovoltaic cell conversion efficiency for solar radiation will be substantially lower than that of conventional solar photovoltaics.

The primary advantages of using conventional photovoltaics (e.g. silicon and gallium arsenide solar cells) are their lower relative cost and continuing power generation when laser-augmentation is not available due to beam eclipse or excessive transmission distances. As discussed above, transmission of beamed power to interplanetary distances, such as from the Earth to a spacecraft in Mars orbit, is extremely challenging because of inherent transmission loss mechanisms (e.g. diffraction-limited beam spreading). Advanced optical relay station technology could enable this capability, but it is assumed that within the specified time-frame of interest for the Mars cargo mission, significant laser-augmentation will be available only to vehicles in Earth or lunar orbit. Also, although solar photovoltaic cells are a relatively expensive component of most spacecraft, they will probably cost less than unconventional cells because the technology and manufacturing facilities are well-developed for conventional cells. Thus, the principal disadvantage of unconventional photovoltaics is that the power output falls essentially to zero outside the range of the laser source.

To summarize, for operation in a beamed-energy LEO-to-GEO LEP OTV mission and especially in a laser-augmented LEP/SEP LEO-to-Mars orbit cargo mission, conventional solar photovoltaics are chosen as superior to unconventional photovoltaics for reasons of cost, flexibility, and overall mission power generation capability when laser augmentation is not available. However, unconventional photovoltaics may offer significant advantages when the primary consideration is performance as a LEO-to-GEO or lunar orbit beamed-energy OTV with an infrastructure that permits continuous laser illumination of the vehicle.

Unconventional photovoltaics may be particularly attractive when combined with high-intensity laser sources, because they may be able to operate at much higher intensities, temperatures, and efficiencies than conventional photovoltaics. By contrast, the conversion efficiency of conventional photovoltaic (PV) cells is strongly dependent upon the operating temperature of the cell.[5,19,20] Greater incident laser intensity (Watts/m^2) will increase the equilibrium temperature during laser heating of the PV array. As the PV cell temperature rises, the conversion efficiency decreases. Therefore, it is necessary to find the incident laser intensity which maximizes array output power (watts/m^2).

A simplified thermal model of the PV array is modelled by the following assumptions: the emissivity of the laser-facing side of the PV array is 0.5, the emissivity of the sun-facing side of the PV array is 0.9, the sun-facing side of the PV array is constantly in sunlight, none of the incident sunlight is converted to electrical power (it is just re-radiated), the PV array reaches equilibrium temperature in a short amount of time compared to the total amount of time during which the laser beam is illuminating the array, and the efficiency of the PV cells is given by the following formula:

$$\text{Efficiency} = (72.9 - 0.1023 \cdot T\,[\text{K}]) / 100 \qquad [\text{for } 273\text{ K} \leq T \leq 713\text{ K}]$$

Given the simplified thermal model described above, the Stefan-Boltzmann

equation was numerically solved over a range of laser intensities to give the equilibrium array temperature shown in Fig. 7-13. Figure 7-14 shows the corresponding PV cell efficiency versus incident laser intensity. This information allows the calculation of PV array output power (Watts of "bus" electric power/m^2) versus incident laser intensity, as shown in Fig. 7-15. This figure indicates that the maximum "bus" electric power is obtained at a laser intensity of approximately six "suns", where one "sun" is approximately 1345 Watts/m^2.

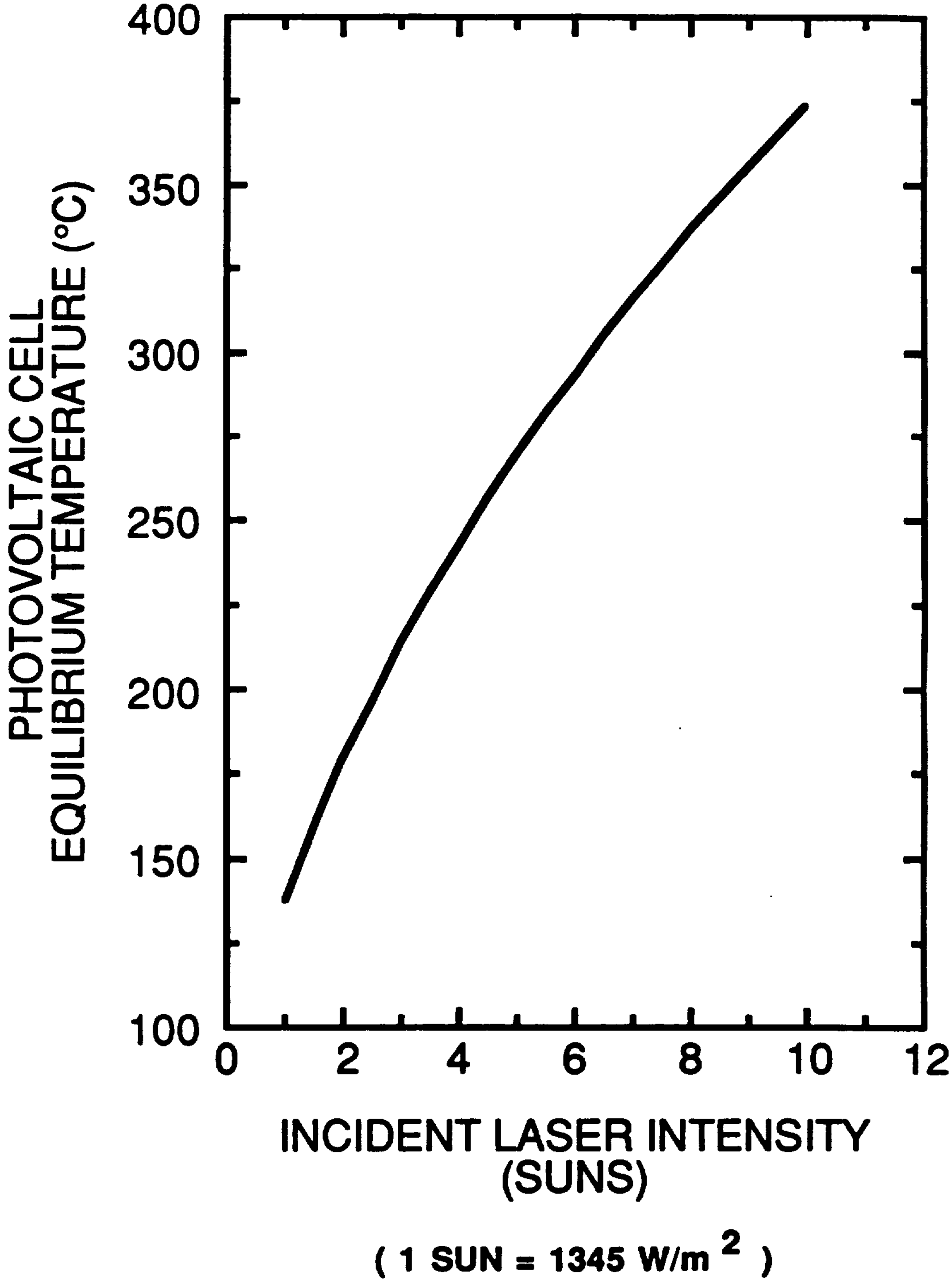


Figure 7-13. Photocell Temperature vs. Laser Intensity

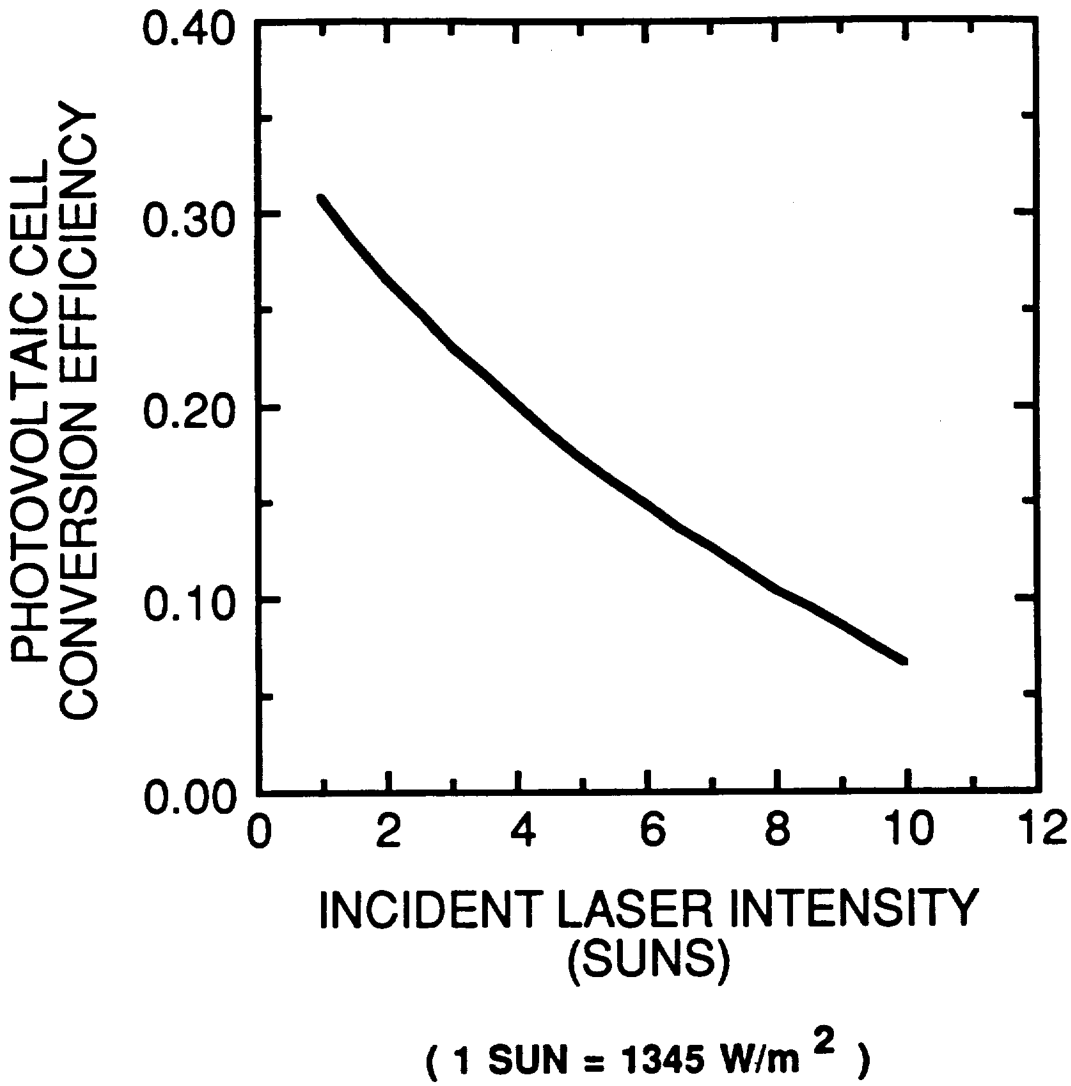


Figure 7-14. Photocell Efficiency vs. Laser Intensity

At this intensity, the PV cells have an efficiency of 15 % as compared to the 25 % efficiency assumed for the solar PV cells described in Section 3 for the 100-MW_e class SEP vehicle. When combined with the 17.7 % miscellaneous distribution losses assumed for the PV arrays in Section 3, the overall efficiency is 12.3 % for the laser-illuminated PV cells. However, because the laser light intensity (W/m^2) is six times that of sunlight, the laser-illuminated PV array has a "bus" electric power output 3.6 times that of an equal-area solar-powered PV array. Assuming the same mass-to-area ratio (1.0 kg/m^2) as that for the solar PV array in Section 3, the corresponding specific mass of a laser-powered PV array is 1.0 kg/kW_e of " bus" power, as compared to a specific mass of 3.6 kg/kW_e for a solar PV array. However, this specific mass is based on the values derived for a 100-MW_e array as discussed in Section 3; the use of the same class of array at power levels of a few megawatts should be considered a lower limit to the LEP array specific mass specific mass and thus an "optimistic" limit on performance.

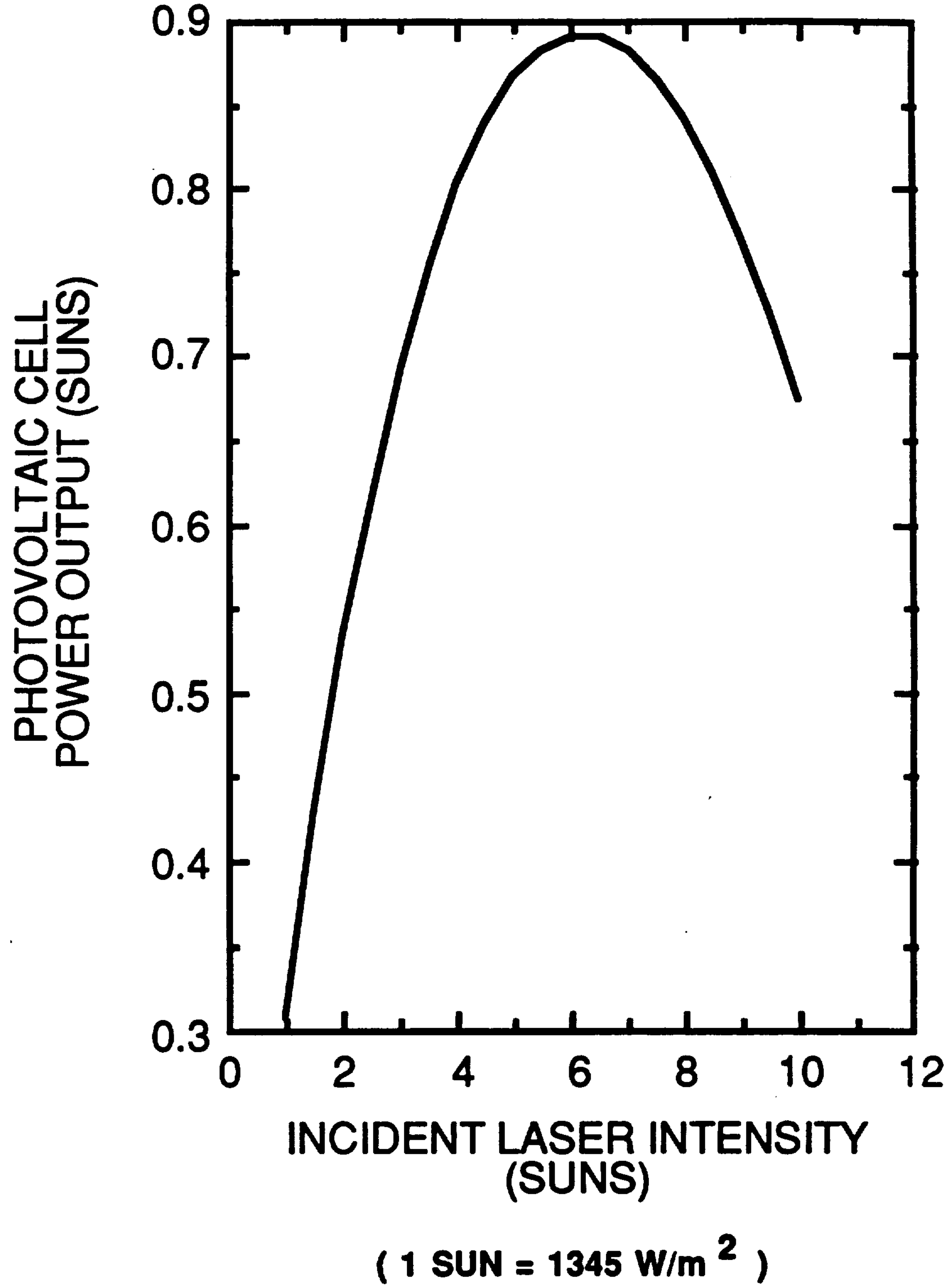


Figure 7-15. Photocell Power Output vs. Laser Intensity

Finally, it is assumed that there is no radiation damage incurred on the PV cells during traversal of the Van Allen radiation belts because of both a rapid passage through the radiation belts (due to the higher power during laser augmentation) and a laser-provided annealing process.[21]

7.2.4.2 Electric Propulsion System. Conventional electric ion thrusters are used with a Power Processing Unit (PPU). However, because the "bus" electric power available to the thrusters is two to three orders of magnitude less in the beamed-energy LEP (and MEP) vehicles than in the 100-MW_e class SEP vehicle discussed in Section 3, the scaling relationships derived for the high-power thruster and PPU were not used for the LEP or MEP vehicles. Instead, a variety of sources[16,22-27] were surveyed to obtain an estimate of thruster and PPU performance at power levels of 30 to 100 kW_e per thruster, as illustrated in Fig. 7-16 and Table 7-1. Based on these results, a thruster plus PPU specific mass of 5.0 kg/kW_e is assumed.[26] The PPU has an assumed 92 % efficiency while the thruster has an efficiency (electric-to-jet power) of 75 %.[26]

Table 7-1. Representative Ion Thruster and PPU Performance Parameters

Power Level (kW_e)	Thruster Specific Mass (kg/kW_e)	Thruster Efficiency	PPU Specific Mass (kg/kW_e)	PPU Efficiency	Reference
10	8.5	0.69	6.0	0.96	22
5	7.04	0.68	7.4	0.90	23
10	5.57	0.76	5.3	0.93	23
3-30	5.0[a,b]	0.73-0.81		0.90-0.92	26
30	3.83	0.79	3.1	0.96	23
50	1.78		4.03		16
222	0.537	0.81	1.0	0.97	25
500	0.175	0.91	0.853	0.98	23
570	2.3[a,b]	0.67		0.915	27
1130	2.3[a,b]	0.67		0.915	27
1130	1.2[a,c]	0.75		0.915	27
4900	0.03	0.85	0.053	0.99	24

Notes: (a) Thruster plus PPU
(b) $I_{sp} = 5000\ lb_f\text{-}s/lb_m$
(c) $I_{sp} = 7000\ lb_f\text{-}s/lb_m$

7.2.4.3 Other Systems. The same basic stage as assumed for the LTP vehicle is again used here, except that the tankage factor is only 3 % corresponding to the use of a propellant such as argon or xenon for the thrusters:

$$M_{Tankage} = 1572.5 + 0.03 \cdot M_p \qquad \text{[all masses in kg]}$$

It is assumed that active refrigeration is not required for the propellant.

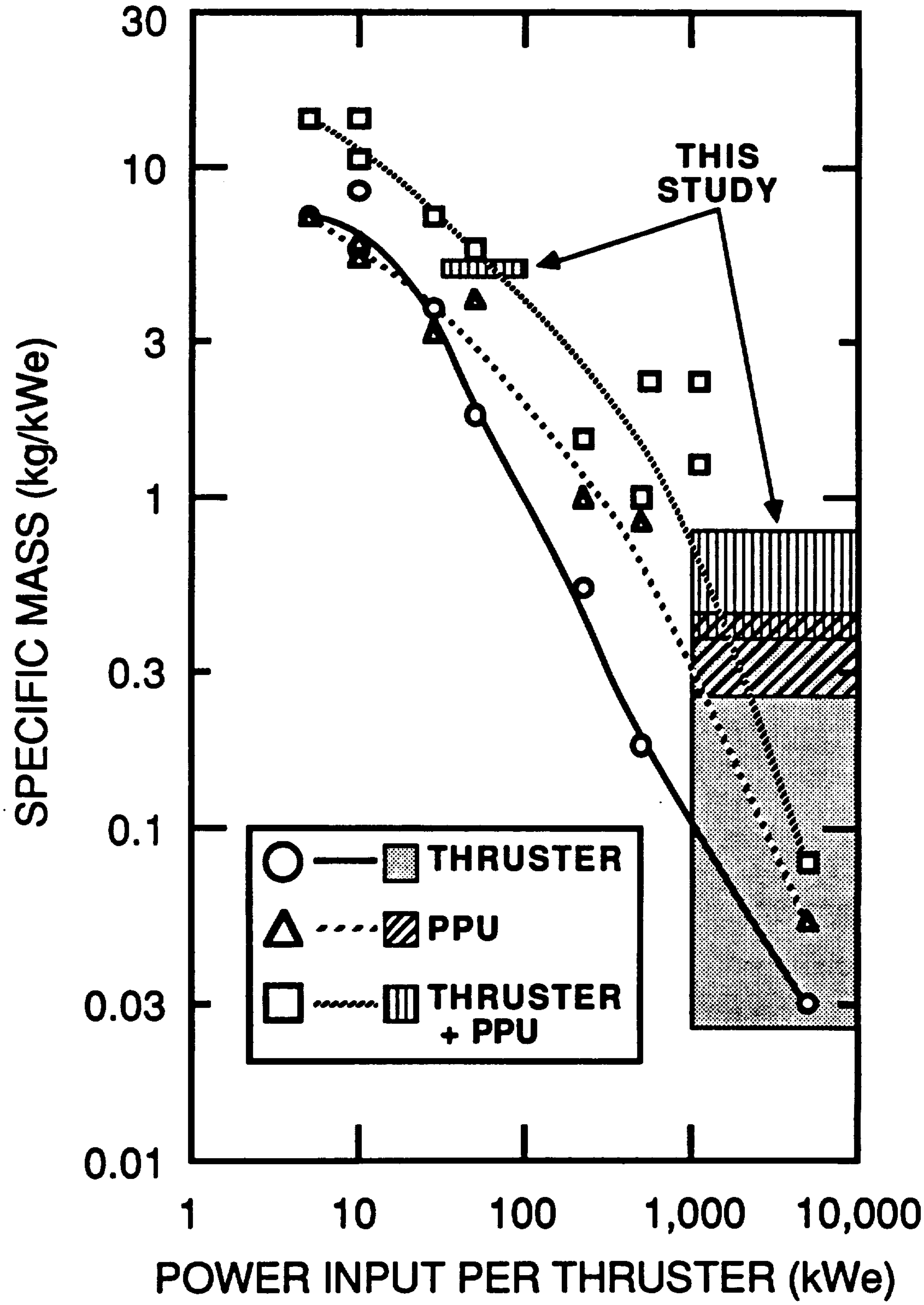


Figure 7-16. Variation of Electric Propulsion Power Processing Unit (PPU) and Ion Thruster Specific Mass With the Power per Thruster

7.2.4.4 Final Vehicle Sizing. Combining the tankage, the laser-powered PV array, and the electric propulsion system (thrusters and PPUs) gives the total LEP OTV vehicle "dry" mass:

$$
\begin{aligned}
M_{Dry} \text{ LEP OTV Vehicle} = {} & 1.5725 + 0.03 \cdot M_p \\
& + (\alpha_{\text{Laser-PV Array}} \, [\text{kg/kW}_e]) \cdot (P_e \, [\text{MW}_e]) \\
& + (\alpha_{\text{PPU\&Thrusters}} \, [\text{kg/kW}_e]) \cdot (P_e \, [\text{MW}_e]) \\
& [\text{all masses in MT}]
\end{aligned}
$$

where M_p is the propellant mass, $\alpha_{\text{Laser-PV Array}}$ is the PV array specific mass as a laser-powered PV array (1.0 kg/kW$_e$ = MT/MW$_e$), $\alpha_{\text{PPU\&Thrusters}}$ is the specific mass of the thrusters and PPU (5.0 kg/kW$_e$ = MT/MW$_e$), and P_e is the "bus" electric power. Finally, a propellant resupply tanker, with a tankage factor of 0.03, is included in the total initial mass in LEO.

To determine the "bus" electric power for the laser-powered PV array, it is necessary to start with the laser beam power leaving the laser transmitter (P_{Beam}) and multiply this by the efficiencies (η) of the various steps in the power transmission and conversion processes:

$$P_e = P_{Beam} \cdot \eta_{Spot} \cdot \eta_{\text{Laser-PV Array}} \cdot \eta_{\text{Miscl. Losses}} = P_{Beam} \cdot 0.9 \cdot 0.15 \cdot 0.823$$

For an LEP OTV, a 1-MW laser beam requires an array with a diameter of 11.9 m; the corresponding diffraction-limited transmitter for GEO distances is 5.8 m. Also, the "bus" electric power is only 0.111 MW$_e$, because the spot size efficiency is 90 % and the conversion efficiency is 12.3 % (15 % • 82.3 %). The same-sized array collects 0.15 MW of sunlight which would produce 0.031 MW$_e$ of "bus" power. Thus, high laser powers are required for an LEP vehicle, primarily due to the low efficiency of converting laser light into electricity.

For use as a laser-augmented SEP vehicle, the array is sized based on its use as an SEP vehicle, but the thrusters and PPUs are sized based on the "bus" power level obtained by using laser light. For example, a 1 MW$_e$ ("bus" power) SEP requires 4.86 MW of sunlight on a 67.8-m diameter array. With a 3.468-m diameter transmitter (the same as that used for the LTP OTV), the laser can transmit power to a distance of about 143,000 km. However, as with the LEP OTV, high laser powers are required. In this case, for a vehicle sized to provide 1 MW$_e$ "bus" power from sunlight, a 32-MW laser beam can be used to provide 3.6 MW$_e$ of "bus" power.

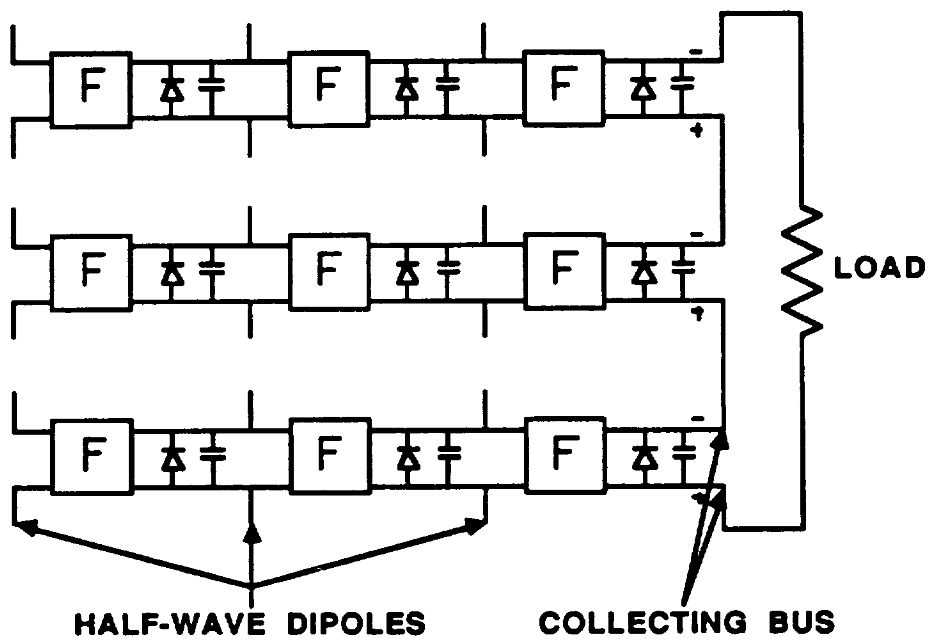
To determine savings on trip time for the laser-augmented SEP, the equivalent Delta-V, and thus propellant mass, required to go from LEO to the maximum laser beam altitude is found. The trip time is then found by dividing the propellant mass by the propellant mass flow rate, which depends on the "jet" power as discussed in Section 2. The "jet" power for the electric propulsion system on the LEP vehicle is a function of the "bus" electric power and the PPU and thruster efficiencies:

$$P_{Jet} = P_e \cdot \eta_{PPU} \cdot \eta_{Thruster} = P_e \cdot 0.92 \cdot 0.75$$

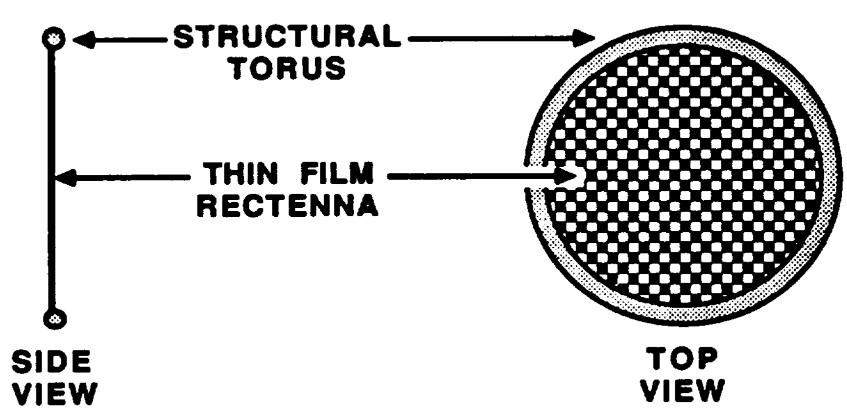
Because the "bus" power (and thus "jet" power and propellant mass flow rate) is 3.6 times greater with the laser in use, the difference in trip time between the two options represents the savings in trip time when the laser is used.

7.2.5 Microwave Electric Propulsion (MEP) Vehicle Sizing

7.2.5.1 <u>Microwave-to-Electric Power Conversion (Rectenna).</u> As with the MTP system, 12.2-cm wavelength microwaves are used. The collector is again 1-km in diameter with a 90-% spot size efficiency. A thin-film etched circuit microwave rectenna is used to convert the microwaves into electricity at an efficiency of 85 %.[1,28] Thin-film rectenna specific masses of 0.25 kg/kW$_e$ ("bus" power) have been demonstrated in the laboratory.[1] The rectenna circuit is mounted on a thin film supported by a toroidal inflatable structure with an areal density of 0.03 kg/m^2. In this case, the inflatable structure is not required to focus the microwave beam, but rather serves only as a support substrate for the rectenna. The rectenna circuit and inflatable structure are illustrated in Fig. 7-17.



RECTENNA CIRCUIT DIAGRAM



RECTENNA INFLATED STRUCTURE CONCEPT

Figure 7-17. Microwave Receiver Rectenna Concept

The rectenna mass thus consists of the sum of the receiver and conversion system (i.e., the inflatable structure and the thin-film rectenna circuit):

$$M_{Rectenna}\,[kg] = pi \cdot (\,1{,}000\text{-m Diameter} / 2\,)^2 \cdot (\,0.03\ kg/m^2\,) + (\,P_e\,[kW_e]\,) \cdot (\,0.25\ kg/kW_e\,)$$

where P_e is the "bus" electric power.

Finally, note that in the LTP, MTP, and MEP vehicles, the optics are of a fixed size and an arbitrary amount of laser or microwave power is allowed to impinge on the optics. In practice, there will be limits to how much power can safely be applied to a vehicle's optics set by thermal loads due to various inefficiencies. For example, if it is arbitrarily assumed that the power density (W/m^2) due to absorbed or waste heat be limited to that of sunlight (1.345 kW/m^2), then the maximum allowable beam power for a 1-km diameter optics MEP vehicle with a 15-% inefficient (i.e., 85-% efficient) optics is over seven gigawatts, far in excess of the beam powers considered in this study. By contrast, for the 20-m diameter LTP vehicle with 10-% inefficient optics (e.g., absorption and reflection losses), the corresponding allowed power is only four megawatts, suggesting that thermal control of the LTP vehicle's optics may be an issue that should be addressed in future studies.

7.2.5.2 Electric Propulsion System. The ion thrusters and PPUs used on the MEP vehicle are the same as those used in the LEP vehicle. No attempt was made to adjust the PPU specific mass or efficiency for differences between the PV array and rectenna output voltages (as was done for the high-powered SEP and NEP vehicles in Sections 3 and 4) because both are relatively low-voltage power sources.

7.2.5.3 Other Systems. The MEP vehicle uses the same tankage, etc. as the LEP vehicle because both employ the same electric propulsion thrusters.

7.2.5.4 Final Vehicle Sizing. As with the LEP vehicle, the "dry" mass of the MEP vehicle is simply the sum of the rectenna optics, electric propulsion system (PPUs and thrusters), and tankage:

$$\begin{aligned} M_{Dry\ MEP\ OTV\ Vehicle} = {} & 1.5725 + 0.03 \cdot M_p + (\,P_e\,[MW_e]\,) \cdot (\,0.25\ kg/kW_e\,) \\ & + pi \cdot (\,1{,}000\text{-m Diameter} / 2\,)^2 \cdot (\,0.03\ kg/m^2\,) / 1000 \\ & + (\,\alpha_{PPU\&Thrusters}\,[kg/kW_e]\,) \cdot (\,P_e\,[MW_e]\,) \end{aligned}$$

[all masses in MT]

where again P_e is the "bus" electric power which is a function of the beam power (P_{Beam}) and the efficiencies (η) of the various steps in the power transmission and conversion processes:

$$P_e = P_{Beam} \cdot \eta_{Spot} \cdot \eta_{Rectenna} = P_{Beam} \cdot 0.9 \cdot 0.85$$

For the specific case of 1-km diameter inflatable optics (23.562 MT), the "dry" mass of the vehicle becomes:

$$M_{Dry}\ \text{MEP OTV Vehicle} = 25.1345 + 0.03 \cdot M_p + (P_e\,[MW_e]) \cdot (0.25\ kg/kW_e) + (\alpha_{PPU\&Thrusters}\,[kg/kW_e]) \cdot (P_e\,[MW_e])$$

[all masses in MT]

Finally, a propellant resupply tanker, with a tankage factor of 0.03, is included in the total initial mass in LEO.

7.2.6 Support Infrastructure

7.2.6.1 Beamed-Energy Transmission System. In determining the total system mass, it was assumed that any space-based beamed-energy transmitter or relay stations would already be available as part of a general cis-lunar beamed-energy transportation infrastructure and thus not be part of the initial mass in LEO (IMLEO) charged to the Mars cargo mission. This assumption also eliminates the need to assess a LEO-to-GEO trip-time penalty for dis-continuous power transmission as, for example, when a vehicle in orbit is out of line-of-sight of a transmitter, because it was assumed that continuous illumination was available from a pre-existing infrastructure. For a laser system, continuous illumination of the vehicle can be achieved by using either ground- or space-based laser transmitter stations in conjunction with a constellation of space-based mirrors; such a system for a 1-micron wavelength has a mass of about 32 MT.[2]

One of the critical issues in assessing the benefits of a beamed-energy transportation system is the "cost" of the infrastructure. In this study, total system mass in LEO has been used as a "cost"-like figure of merit. Thus, one way to reduce the infrastructure "cost" (mass in LEO) is to ground-base the laser or microwave transmitters. This has several cost and operational advantages, because construction, supply of power, and maintenance are all greatly facilitated. However, this does introduce the potential difficulties of beaming through the atmosphere (e.g., scattering, distortion due to thermal blooming, etc.). There is also the problem of keeping the transmitter and vehicle in line of sight to insure continuous illumination. This can be accomplished with a combination of space-based relay mirrors, multiple ground-based transmitters, or by allowing non-continuous illumination of the vehicle, although the latter option would tend to increase the already long LEO-to-GEO trip times of the low-thrust beamed-energy OTVs. By contrast, space-basing the transmitters and relay optics allows continuous operation and eliminates problems associated with transmission through the atmosphere. (In fact, wavelengths can be chosen that are strongly attenuated by the atmosphere in order to protect people on the ground.)[1] Space-basing of the beamed-energy infrastructure does, however, significantly increase the mass in LEO. For example, if a space-based 1-MW (beam), 1-micron wavelength Free Electron Laser (FEL) system is used, the infrastructure mass is about 460 MT.[2] For a microwave system, the space-based system mass is likely to be dominated by the mass of the optics because they can be on the order of several kilometers in size. For example, a single 10-km inflatable optics system has a mass of 2356 MT. As many as six of these optics (one on each of two transmitter stations and one on each of four relay mirrors)[2] might be required for a full space-based microwave system. Substantial savings may be possible using synthetic aperture techniques, although even a single 10-km diameter "Y"-shaped array of fifteen 1-km diameter inflatable optics (the same size as on the microwave-powered OTVs) would weigh 353 MT.

However, these systems can support only one OTV at a time; multiple systems would be required for a fleet of OTVs operating in parallel (up to 20 in the present study). This requirement could be reduced by running the OTVs in series, but then only at the expense of greatly increased trip times. For example, the trip times of the fully loaded

(400 MT payload) OTVs, where only one OTV is required, are on the order of two years for the LEO-to-GEO transfer with a 10-MW beam power. Splitting the payload among multiple OTVs thus results in a trip time savings only when the OTVs are operated in parallel.

7.2.6.2 Payload Support. As with the other advanced concepts discussed previously, the option of splitting the payload among many parallel vehicles is considered. For the beamed-energy OTV missions, a single-stage aerobraked chemical (O_2/H_2) vehicle was attached to each payload unit to perform the trans-Mars injection from GEO altitude and subsequent aerobraking into Mars orbit. This vehicle is simply a re-sized version of the second (aerobraked) stage of the baseline chemical system described in Section 1. Because the trans-Mars injection Delta-V from GEO is less than two-thirds that required when leaving from LEO,[29] only a single-stage, rather than two-stage, vehicle was required. As shown in Fig. 7-18, an adaptor (2.5 % of the supported weight) is added between the cargo payload and the chemical stage; a similar adaptor is added between the chemical stage and the beamed-energy OTV.

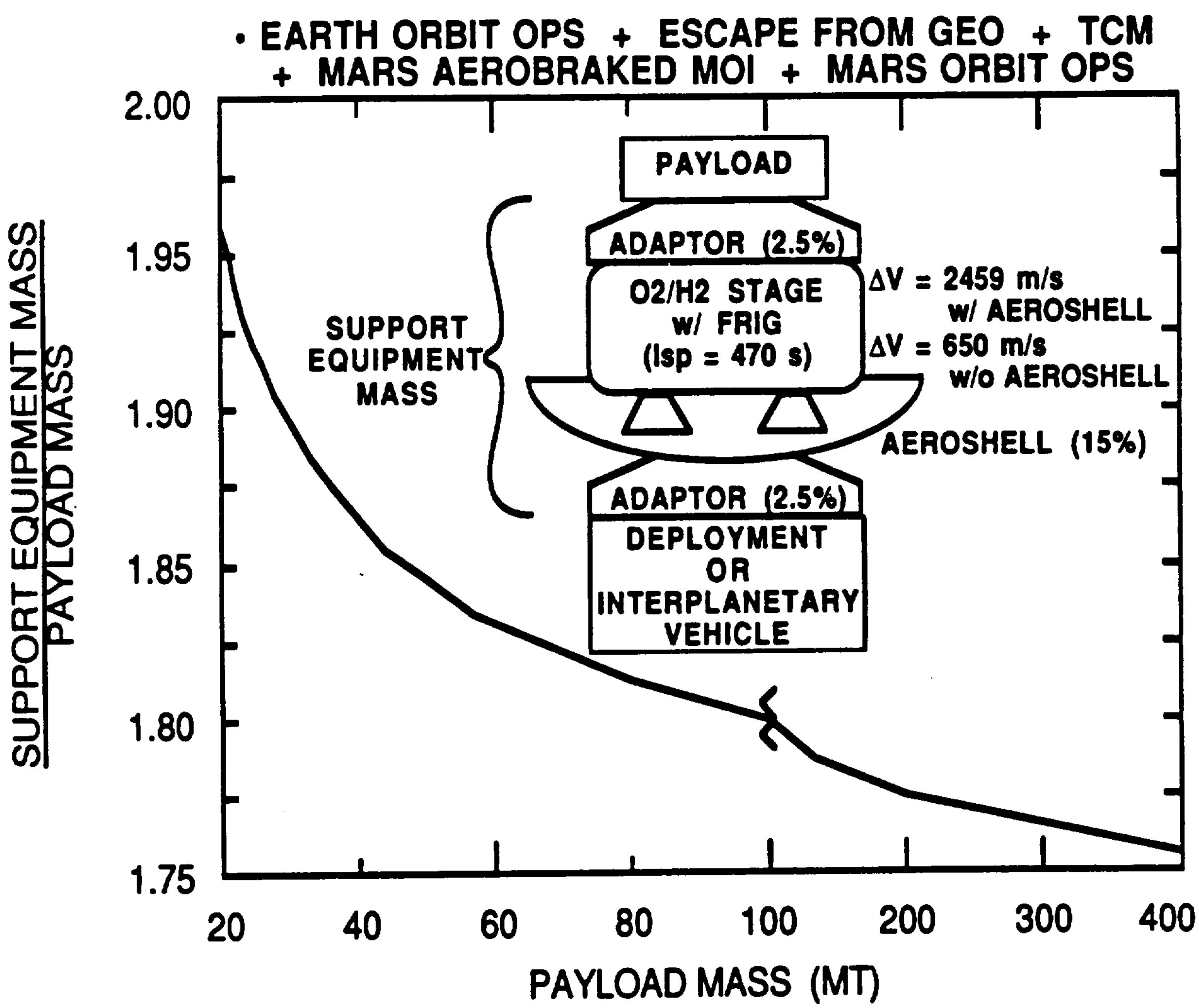


Figure 7-18. Effective Payload Weight Due to Adding an Aerobraked Chemical (O2/H2) Stage for Escape from GEO-Altitude Earth Orbit and Aerobraking Into Mars Orbit (Plus Earth and Mars Orbital Operations and Earth-to-Mars TCM)

For the laser-augmented SEP mission, an Orbital Maneuvering Vehicle (OMV) is added to each cargo payload unit for maneuvering in Earth and Mars orbits, as was done with the high-powered SEP vehicle in Section 3.

7.2.7 Trajectory Analysis

All the low-thrust beamed-energy vehicles are used to deploy the cargo and its attached high-thrust chemical stage to GEO altitude but without the 28.5° plane change to a normal equatorial GEO orbit. This results in an Edelbaum equation[30] (see Section 2) Delta-V of only 4.7 km/s in contrast to the 6.0 km/s required for a low-thrust LEO-to-GEO transfer with plane change. Interestingly, an altitude of about twice GEO minimizes the required Delta-V needed to inject the cargo to the required C_3 of 9.541 km^2/s^2; this is found by differentiating the equation relating Delta-V, C_3, and the departure circular orbit's orbital velocity V_0:

$$\text{Delta-V} = (C_3 + 2 \cdot V_0^2)^{1/2} - V_0$$

For a minimum Delta-V (i.e., d(Delta-V)/d(V_0) = 0), the result is:

$$R_0 = 2 \cdot \mu / C_3$$

where R_0 is the radius of the departure orbit that minimizes the Delta-V requirement, μ is the Earth's gravitational parameter (398,601 km^3/s^2) and $V_0 = (\mu/R_0)^{1/2}$, and C_3 is the excess hyperbolic velocity required for trans-Mars injection. However, this saves only 75 m/s in injection Delta-V over that required when leaving from a GEO altitude (2.259 km/s). Thus, for convenience, a 28.5°-inclination GEO altitude was chosen, because this would minimize the beamed-energy OTV Delta-V and transmission distance, nearly minimize the chemical stage's Delta-V, and permit the use of OTVs that could also be used for routine LEO-to-GEO (0°-inclination) operations.

For the high-thrust chemical stages, the total Delta-V budget consists of 100 m/s for LEO rendezvous and docking, 2259 m/s for trans-Mars injection, and another 100 m/s for Earth-to-Mars trajectory correction maneuvers (TCM) for a total of 2.459 km/s prior to aerobraking at Mars. This trans-Mars injection Delta-V is significantly less than that required for the baseline chemical system leaving from LEO, e.g., 2.259 versus 3.588 km/s, respectively. After aerobraking, a 0.650-km/s Delta-V is allocated for orbit circularization and final rendezvous, as was done for the baseline chemical system in Section 1.

For the high-thrust chemical stage, the trip time from GEO to Mars is assumed to be the same as the 294 days for the baseline chemical vehicle.[29] To this time is added the LEO-to-GEO trip time of the beamed-energy OTV. This beamed-energy OTV trip time is found in the same manner as that described in Section 1, where the trip time is found by dividing the propellant required for the LEO-to-GEO transfer by the propellant mass flow rate. In this case, it is assumed that the beamed-energy OTV is constantly illuminated by the laser or microwave source, so the trip time is not increased due to coasting during shadow periods.

For the laser-augmented SEP case, the propellant and trip time requirements were first found assuming the mission was done as a "pure" SEP mission, as in Section 3. Then, the Edelbaum equation Delta-V was found for the transfer from LEO to the maximum beaming-distance altitude for the outbound (and return) trip. Knowing the initial (and final) vehicle mass in LEO, the propellant consumed during this transfer was found. Next, the trip time for the near-Earth transfer was found for both the conventional and

laser-augmented modes, based on the respective mass flow rates. Finally, the difference between these two trip times was subtracted from the total Earth-to-Mars trip time as a "pure" SEP vehicle to give the trip time with the laser augmentation. The trip time difference between the conventional and laser-augmented modes for the near-Earth portion of the trip was a large fraction of the total (typically about one-third), suggesting that laser augmentation of SEP vehicles can show significant trip time savings for the slow escape (and capture) from Earth.

7.3 RESULTS

Figures 7-19 and 7-20 summarize the results for the beamed-energy propulsion systems. There are modest savings in initial mass in LEO, although the Earth-to-Mars trip times are rather long for near-term laser powers (up to 10 MW beam). Only at very high beam power levels do the trip times begin to decrease significantly. However, if high-powered lasers and microwave transmitters are available as part of a pre-existing beamed-energy infrastructure, their use may be attractive for the Mars cargo mission. Also, in Fig. 7-19, the curves for the low-powered (1 MW beam) cases extend far to the right of the figure towards long trip times (ca. 3000 days) where there are small savings in IMLEO as compared to the high-powered (10 MW beam) cases. However, the savings in IMLEO are small, assuming the laser or microwave transmitter infrastructure is not included, because the masses of the vehicles are generally only weakly dependent on beam power level.

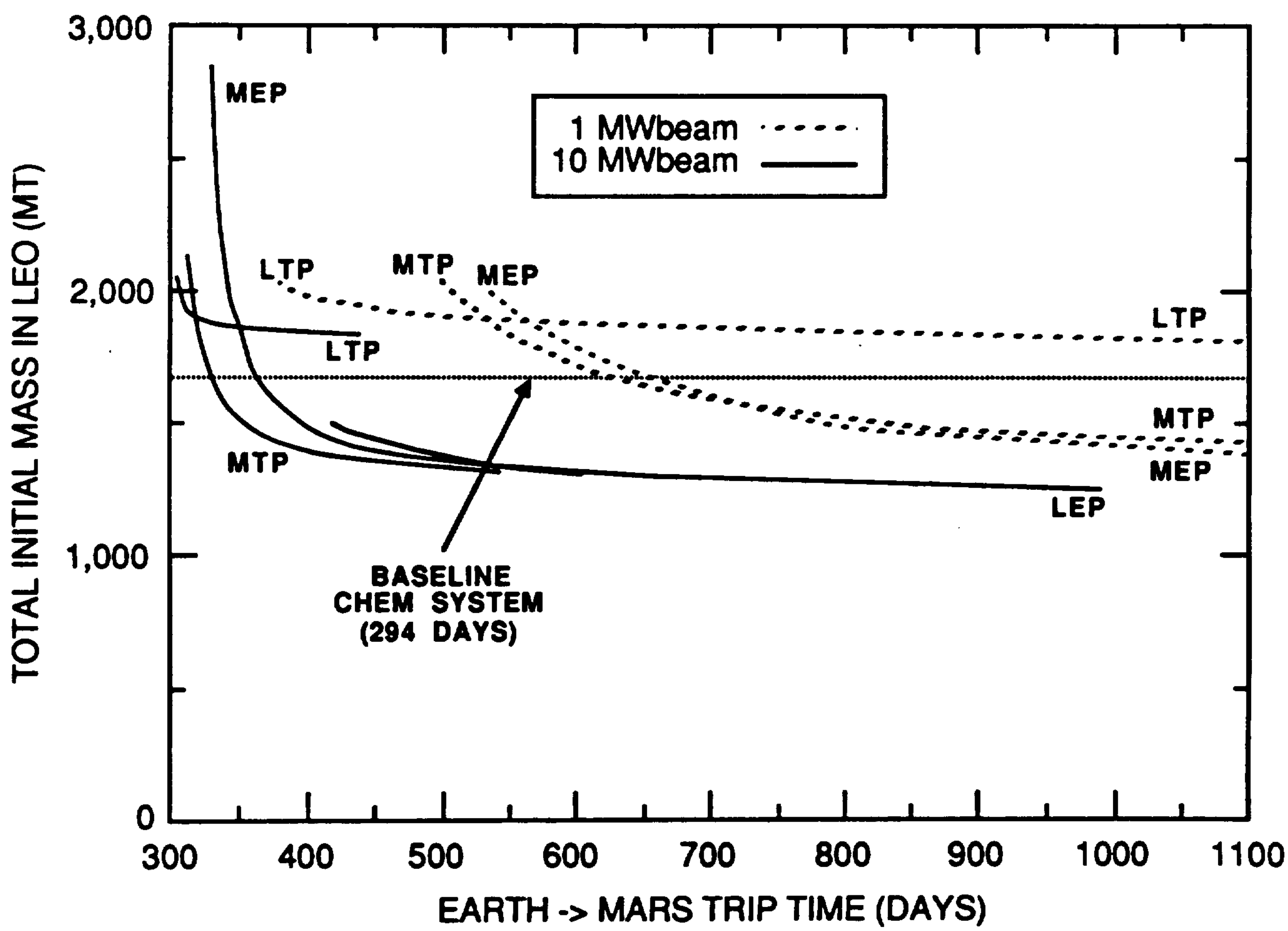


Figure 7-19. Initial Mass in Low Earth Orbit vs. Earth-to-Mars Trip Time for Beamed-Energy Propulsion OTV

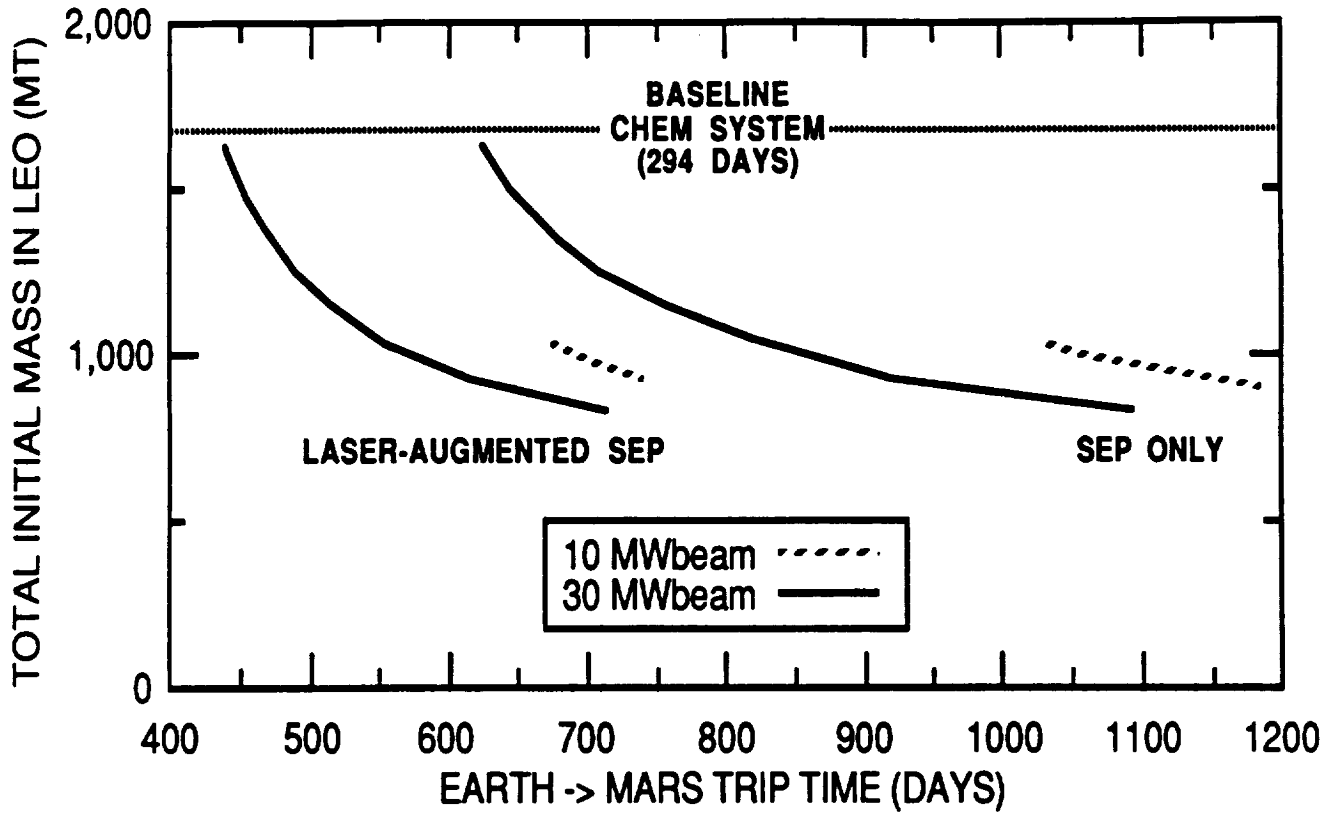


Figure 7-20. Initial Mass in Low Earth Orbit vs. Earth-to-Mars Trip Time for Laser-Augmented Solar Electric Propulsion

Finally, note that the IMLEO values found above do not include any space-based infrastructure, nor is there a trip-time penalty applied for dis-continuous power transmission. Inclusion of the infrastructure mass of even one space-based FEL transmitter (for the LTP or LEP OTVs) or a single space-based microwave transmitter or orbital relay (for the MTP or MEP OTVs) would completely negate any mass advantage of the beamed-energy system.

7.4 CONCLUSIONS

Beamed energy propulsion can provide modest savings in IMLEO and trip time if a pre-existing high-power beamed energy transmission system is available. However, the system can provide beamed power only out to GEO distances for microwave wavelengths and to lunar distances for near-visible wavelengths. Also, very high beamed power levels (>>10 MW beam) are required for significant reductions in trip time due to the overall low efficiency of converting beam power into jet power. Finally, the infrastructure required to provide continuous, high-power beamed energy is extensive and would completely negate any savings in IMLEO if provided to support the Mars cargo mission only. Thus, beamed-energy systems may be attractive for support of general operations in cis-lunar space (e.g., OTV operations, lunar base, orbital factories, etc.), including delivery of cargo to high Earth orbits for injection towards Mars, but they should not be considered if implemented solely on the basis of their use for the Mars cargo mission.

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SECTION 8

SOLAR THERMAL PROPULSION

8.1 INTRODUCTION

In the Solar Thermal Propulsion (STP) concept, shown in Fig. 8-1, sunlight is collected by a large inflatable "mirror" and focused into a thruster where the sunlight is absorbed and used to heat a propellant (e.g., hydrogen) which then expands out through a conventional nozzle. There are several similarities between solar thermal and laser thermal propulsion, as will be described below. The Air Force Astronautics Laboratory is currently funding STP thruster development. A prototype engine, using a rhenium-tube heat exchanger, has achieved specific impulses (I_{sp}) in the 800 lb_f-s/lb_m range. Advanced STP thruster concepts, using particle-bed heat exchangers or particulate absorption directly in the propellant, are projected to achieve I_{sp}s on the order of 1200 lb_f-s/lb_m.[1]

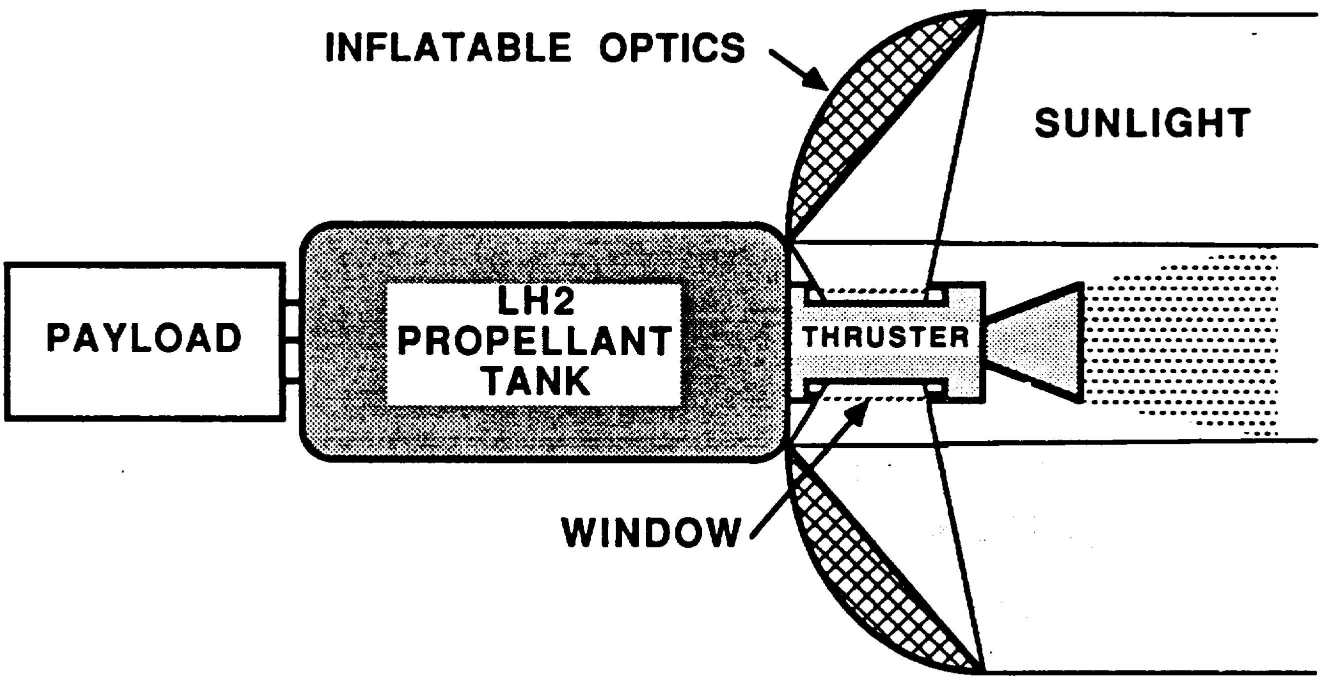

Figure 8-1. Solar Thermal Propulsion Concept

8.2 ASSUMPTIONS

8.2.1 Mission Scenario

Because the STP vehicle is a solar energy (rather than nuclear energy) vehicle, it can operate directly from low Earth orbit (LEO). Thus, no supporting orbit transfer vehicle (OTV) infrastructure is required. Because the vehicle is relatively small and "simple" (at least as compared to electric propulsion concepts), it is assumed that it is expendable; a single-stage vehicle leaves LEO and transfers to Mars/Phobos orbit, but does not return to Earth, as shown in Fig. 8-2.

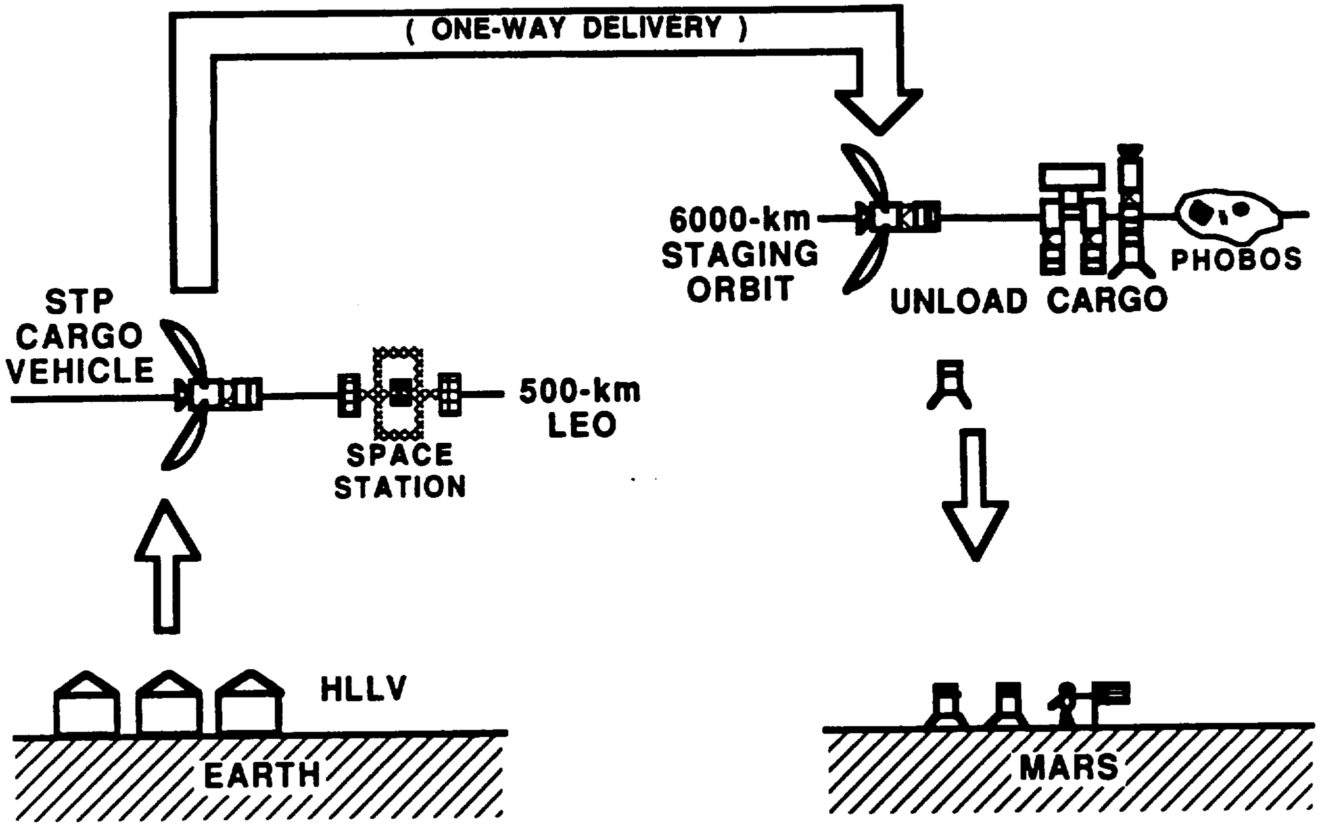

Figure 8-2. Solar Thermal Propulsion Mission Scenario

8.2.2 STP Vehicle Sizing

As mentioned above, the STP concept is very similar to the laser thermal propulsion (LTP) concept discussed in Section 7. Both concepts use inflatable optics to focus near-visible light into a thermal thruster where the light is absorbed and used to heat a propellant. The primary difference, from a vehicle sizing perspective, comes in the size of the inflatable optics and the thruster specific impulse. For example, in the STP vehicle, the size (area) of the inflatable optics varies with the solar power collected, which in turn determines the thruster power. By contrast, in the LTP vehicle, the size and power-handling capacity of the inflatable optics is somewhat arbitrary, being determined more by the laser transmitter than by the vehicle requirements. In terms of potential I_{sp}, a laser source represents a monochromatic point source, so it is possible to employ high-I_{sp} energy coupling mechanisms that make use of specific wavelengths (e.g., molecular resonance absorption) or of the ability to focus the beam to very high intensities (e.g., inverse Bremsstrahlung absorption). By contrast, the sun is a non-monochromatic source with a finite spot (disk) size, so only heat exchanger or particulate absorber mechanisms can be employed. Thus, a STP thruster has an I_{sp} of only 1200 lb_f-s/lb_m instead of the 1500 lb_f-s/lb_m of a LTP thruster.

8.2.2.1 STP Thruster. It is assumed that two high-performance STP thrusters are used with an I_{sp} of 1200 lb_f-s/lb_m and an efficiency (sunlight-to-jet power) of 50 %, which is the same efficiency as that assumed for the LTP thrusters. Two engines are used, each sized for one-half the power available at 1 AU, so that at Earth, both engines can be used at

their full power for Earth escape, while at Mars, one engine can be used at its full power for an all-propulsive Mars orbit insertion (no aerobraking). The thruster scaling equation developed in Section 7 for the laser-thermal thruster is again used, but modified for the case of two thrusters:

$$M_{Thrusters}\,[MT] = 2 \cdot 2203.4 \cdot \{\, P_{Jet}\,[MW] \,/\, (\, 2 \cdot g_c \cdot I_{sp}\,[lb_f\text{-}s/lb_m]\,)\,\}^{1.15}$$

where P_{Jet} is the thruster "jet" power and g_c is 9.8 m/s^2. The jet power depends on the total sunlight power collected and on the efficiencies (η) of the thruster and inflatable optics:

$$P_{Jet} = P_{Solar} \cdot \eta_{Thruster} \cdot \eta_{Optics}$$

8.2.2.2 Inflatable Optics. The inflatable mirrors are of the same kind as those described in Section 7 for the LTP vehicle; they are sized to collect solar power in the range of 1 to 100 MW, corresponding to intercepting 743 to 74,300 m^2, respectively, of sunlight at 1 AU. Assuming two mirrors as shown in Fig. 8-1, the diameters corresponding to each of the intercepted areas are 21.8 m and 218 m, for 1 and 100 MW, respectively. Note however that as in the LTP vehicle, the collectors are at a 45° angle to the incoming light, so the actual area of the collector is $\sqrt{2}$ times the projected area perpendicular to the sun. The STP (and LTP) inflatable collectors have an areal density of 0.03 kg/m^2 and a total efficiency (reflection and transmission) of 90 %. Thus, the jet power is 45 % of the initial solar power that is collected. The mass of the inflatable mirrors is simply:

$$M_{Optics}\,[kg] = 1.414 \cdot (\, P_{Solar}\,[kW_s] \,/\, 1.345\,[kW_s/m^2]\,) \cdot (\, 0.03\ kg/m^2\,)$$

where 1.345 kW$_s$/m^2 is the solar power density at 1 AU. This equation reduces to:

$$M_{Optics}\,[MT] = 1.051 \cdot (\, P_{Solar}\,[MW_s]\,) \cdot (\, 0.03\ kg/m^2\,)$$

Interestingly, the inflatable collectors, although physically large, are light in weight; for example, a 1-MW (solar) collector has a mass of only 32 kg.

8.2.2.3 Propellant Tankage and Refrigeration. As with the LTP vehicle in Section 7, liquid hydrogen (LH_2) propellant is used, and a sorption refrigerator is added to the vehicle for active cooling of the propellant at 20 K.

8.2.2.4 Final Vehicle Sizing. The overall vehicle "dry" mass for the STP vehicle, like the LTV vehicle, is simply the sum of the thruster, optics, propellant tankage, refrigerator, and miscellaneous systems:

$$M_{Dry\ STP\ Vehicle} = 1.6184 + M_{Thrusters} + M_{Optics} + 0.2584 \cdot M_p + 0.1736 \cdot (\, M_p\,)^{2/3}$$

[all masses in MT]

where M_p is the propellant mass. Finally, a propellant resupply tanker, with a tankage factor of 0.2584, is also included in the total initial system mass in LEO.

8.2.3 Trajectory Analysis

The low-thrust STP trajectories were provided by Carl G, Sauer Jr. of JPL.[2] The results of the trajectory code are given in the Appendix; the mission analysis methodology is similar to that described in Section 3 for the SEP system. The primary difference

between the SEP and STP mission analysis is the need to only treat the Earth-to-Mars portion of the trajectory, because the STP vehicle is not returned to Earth.

8.2.4 Support Infrastructure

Because the STP vehicle is relatively small (other than the inflatable collectors), it is assumed that it can directly dock with an orbiting base. Thus, it is not necessary to add a dedicated Orbital Maneuvering Vehicle (OMV) to the payloads, although a 2.5 % structural adaptor is added to each payload. Finally, as with several of the systems described above, the option of splitting payloads among several vehicles is considered.

8.3 RESULTS

As shown in Fig. 8-3, the total initial mass in low Earth orbit (IMLEO) for an STP vehicle system is in excess of 4800 MT, as compared to only 1646 MT for the baseline O_2/H_2 system. There is no trip time advantage, because the chemical system has an Earth-to-Mars trip time of 294 days as compared to the STP system trip times in excess of 300 days.

The poor performance of the STP is due to two reasons. First, the I_{sp} (I200 lb_f-s/lb_m) and thus exhaust velocity (11.8 km/s) of the STP thruster is too low compared to the "effective" low-thrust Delta-V (14 to 18 km/s). This results in a large mass ratio (initial "wet" mass divided by final "dry" mass) of three to five, which in turn implies that the propellant mass is two to four times the final mass ("dry" vehicle plus payload) delivered to Mars orbit. This high propellant mass ratio, by itself, is not severe, except that the vehicle uses liquid hydrogen propellant; the need to store low-density, low-temperature liquid hydrogen results in a STP vehicle with a high dry weight (tankage, etc.) even though the thrusters and inflatable collectors are light in weight.

For example, as shown in Fig. 8-4, the performance of the STP vehicle is fairly sensitive to the assumed tankage factor. The value of 0.2584 used in this analysis is consistent with a value of 0.25 for LH_2 tankage used in previous studies.[3] More recent studies have assumed tankage factors around 0.20.[4] Finally, it may be possible to have a tankage factor as low as 0.15 for hydrogen by using "slush" hydrogen (a mixture of solid and liquid hydrogen) to increase the effective density (i.e., decrease the volume for a given mass) and thereby decrease the required tankage.[3,5] These variations in tankage factor are treated parametrically in Fig. 8-4, where it is seen that there can be significant reductions in IMLEO and trip time for low tankage factors, although the STP vehicle is still not competitive with the IMLEO of the baseline aerobraked chemical (O_2/H_2) system option.

This situation is made worse by the need for a large fleet of vehicles at lower power levels. Operation at higher powers does reduce the total mass and fleet size, although not significantly. For example, with an Earth-to-Mars trip time of about 650 days, 13 vehicles with a total system IMLEO of 5400 MT are required at a 1 MW (sunlight) level; at 3 MW, this drops to 4 vehicles and 4900 MT, respectively. Alternatively, for a fixed total IMLEO, operation at higher power results in modest reductions in trip time; for example, at an IMLEO of abut 6200 MT, a 3-MW, 8-vehicle system has a trip time of about 490 days, while a 1-MW, 20-vehicle system has a trip time of 550 days. Thus, the effect of power level on system performance is modest.

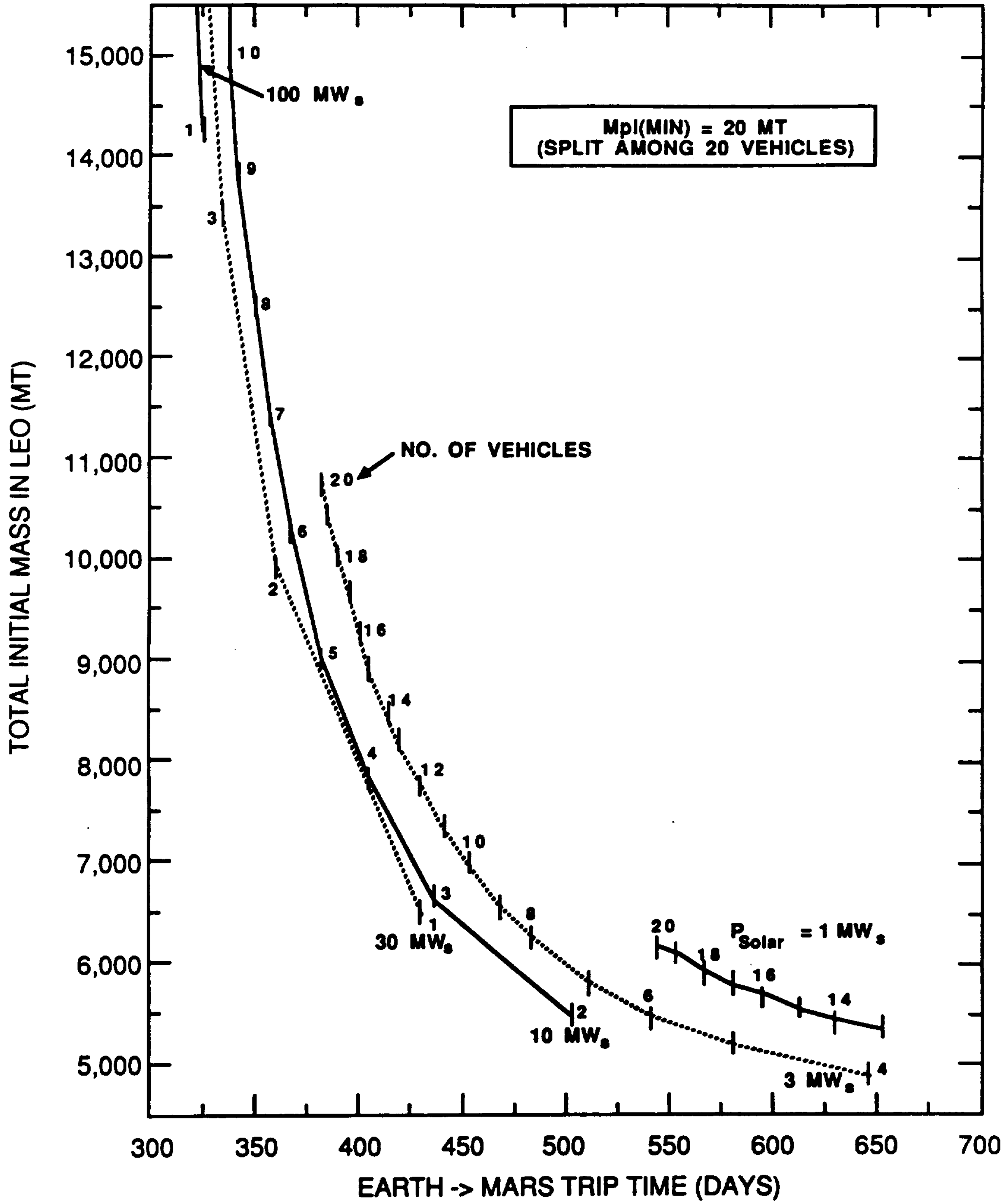

Figure 8-3. Initial Mass in Low Earth Orbit vs. Earth-to-Mars Trip Time for Solar Thermal Propulsion

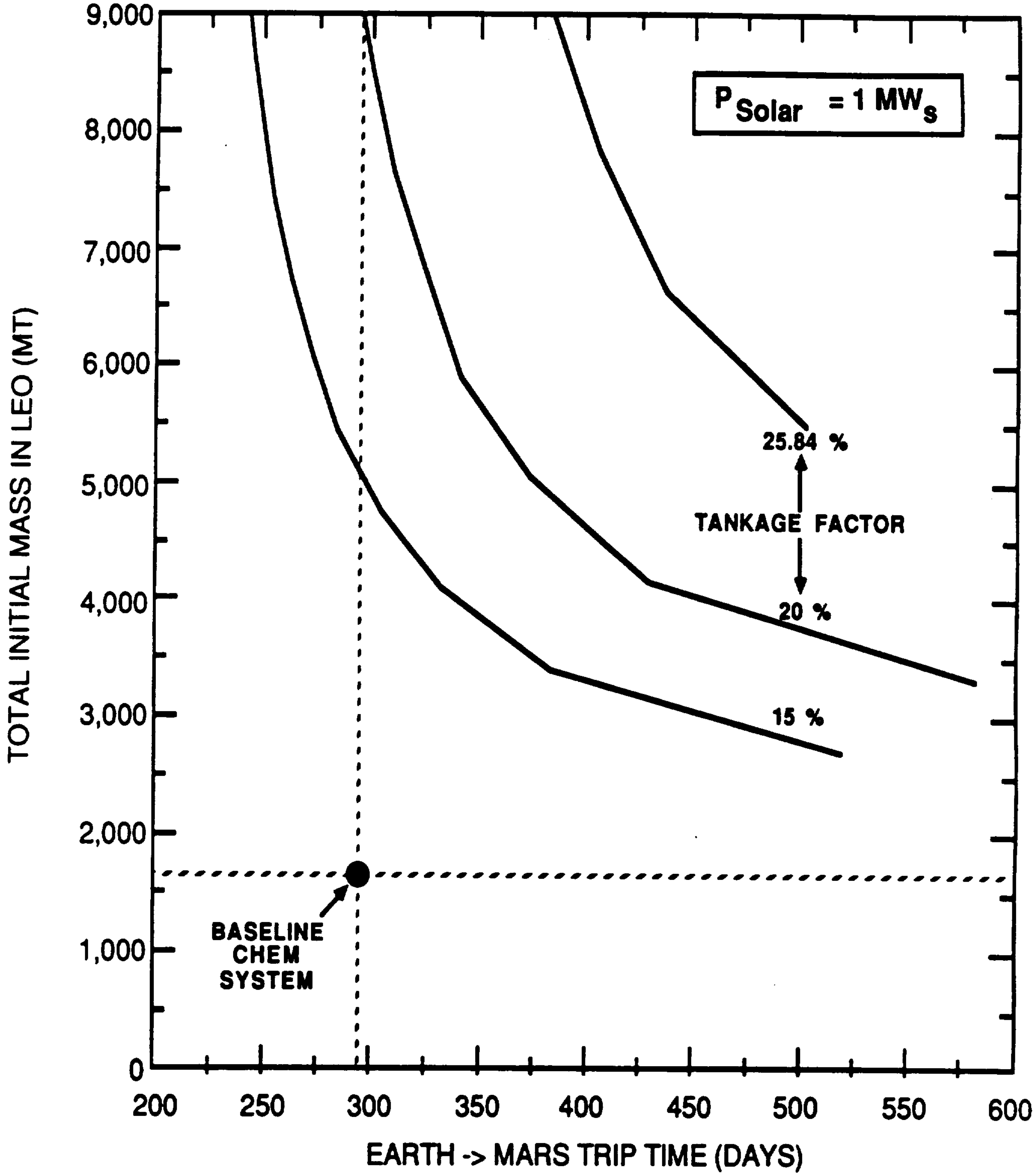

Figure 8-4. Initial Mass in Low Earth Orbit vs. Earth-to-Mars Trip Time for Solar Thermal Propulsion as a Function of the Propellant Tankage Factor

8.4 CONCLUSIONS

The solar thermal propulsion system is not competitive with the baseline chemical O_2/H_2 system. It is at least three times heavier and provides no savings in trip time. This is due primarily to the low I_{sp} of this concept and the need to store liquid hydrogen, resulting in high propellant weights and vehicle "dry" weights.

Note however that a single-stage, non-aerobraked vehicle was used in these analyses. One option to improve performance would be the use of a two-stage vehicle (i.e., one for Earth escape and a second for Mars capture). Thus, the analysis performed above may be in the same category as those analyses performed during the 1920s and 1930s, which "proved" that a single-stage rocket could never reach the Moon. (The analyses were correct, but the assumption that a single-stage rocket was the only way to do the mission was totally wrong.) A second option would be the use of aerobraking into Mars orbit. This could be accomplished by using a low-thrust STP vehicle for Earth escape and a high-thrust O_2/H_2 aerobraked vehicle for Mars orbit insertion. However, these options were not considered in this study because they would have required extensive modification of the low-thrust trajectory codes for their analyses. Finally, it would be possible to use the STP vehicle in an orbit transfer vehicle (OTV) mode similar to that used for the LTP vehicle. It is recommended that OTV-type mission modes be considered and that the low-thrust trajectory codes be augmented to permit evaluation of multiple-stage options for future studies.

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SECTION 9

TETHERS

9.1 INTRODUCTION

Tether concepts for propulsion and power have been investigated within the last decade for a variety of space missions. Two classes of tether systems are electrodynamic tethers, which interact with a planetary magnetic field, and non-conducting tethers which interact with the gravitational field. The present study investigates the benefits of the latter class for propulsive assist in an unmanned, Earth to Phobos cargo mission.

An object placed in orbit about a planetary body remains in orbit because the inward-directed gravitational force is balanced by the inertial or centrifugal force, in response to which the body moves outwards. The tether systems considered here begin operation with the entire system (which includes the payload, propulsive stages, tethers, and tether station) in a circular orbit. The tether is then deployed with the payload and transfer vehicle at one end of the tether and the station at the other. If the payload orbit is to be raised, the tether is deployed "up" or radially outward. Conversely, if the payload orbit is to be lowered, then the tether is deployed "down" or radially inward. After a period of time the tether reaches mechanical equilibrium in a vertical orientation. In addition, the center of mass is located at an altitude slightly lower than the original altitude because of a net tidal force which has done work on the entire system. Once any transient motions have been damped out, the entire system orbits the planet in a circular orbit with uniform angular velocity. For an outbound mission, the payload and transfer vehicle will be above the center of mass and have a velocity which is super-circular, i.e. , faster than the circular orbital velocity at the payload's altitude. The station, on the other hand, will be located below the center of mass and have a velocity which is sub-circular. The payload is then disengaged from the tether and enters a larger elliptical orbit with its perigee located at the release point. The station enters a lower-energy elliptical orbit with its apogee located at the release point. These orbits are depicted in Fig. 9-1. The tether is then reeled back into the station, after which a pair of propulsive burns are required to bring the station back up to its original circular orbit. The payload and transfer vehicle then perform a propulsive burn to reach the required velocity for injection to Mars (C_3=9.541 km^2/s^2).

In this study , the scenario described above corresponds to operations in low Earth orbit (LEO) where a large (500 MT) station is used to assist a payload and transfer vehicle (64.3 MT) to achieve the required earth escape velocity. At Mars, the procedure is reversed with the payload and transfer vehicle being captured by the Deimos tether station and transferred to the Phobos station. An important operational difference is that Deimos and Phobos are used as tether "stations"; the fact that these moons are orders of magnitude more massive than the payload eliminates the necessity to reboost the "station" back to its original orbit.

A two-stage aerobraked chemical (O_2/H_2) vehicle, similar to the baseline chemical vehicle described in Section 1, was used to inject the payload towards Mars after release from the Earth-orbit tether. The second stage continues to Mars where it aerobrakes into an elliptical orbit which allows it to rendezvous with the Deimos tether. This mission scenario is illustrated in Fig. 9-2.

Various facets of the tether system and Mars mission have been investigated in some detail in the past. Paul Penzo of JPL has outlined design requirements as well as operation for a LEO tether transportation system.[1] In addition, he has considered issues

related to operation of a Mars tether transportation system.[2] The change in center of mass due to the net tidal force has been proposed as a means of satellite relocation by Geoffrey Landis of the Lewis Research Center.[3]

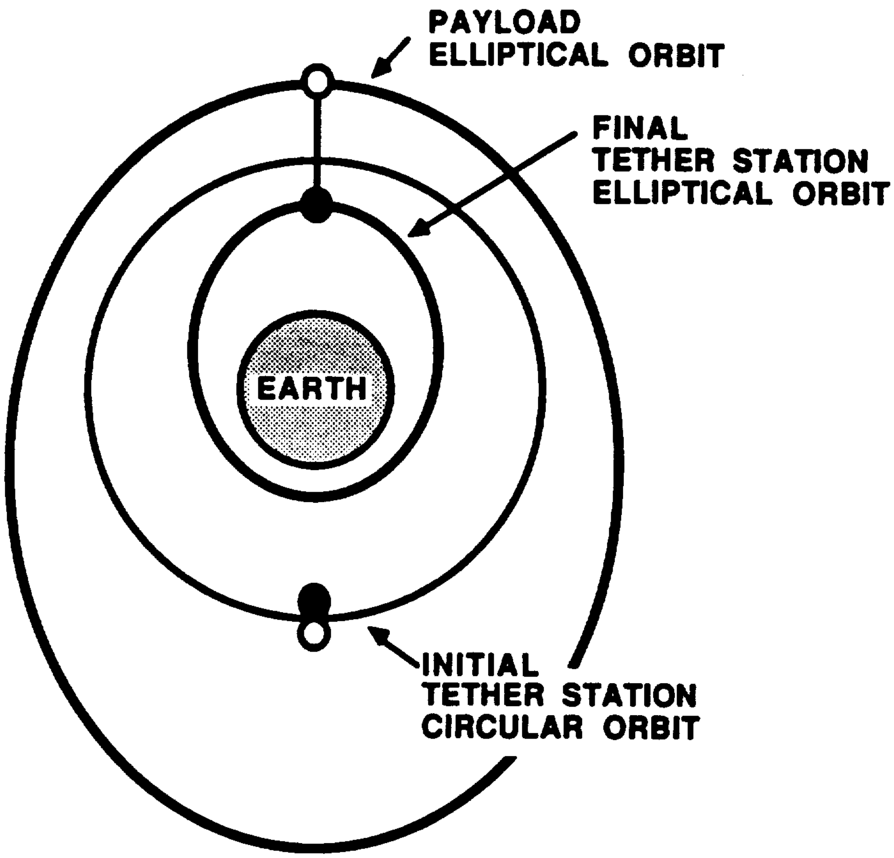

Figure 9-1. Tether System Orbits at Earth

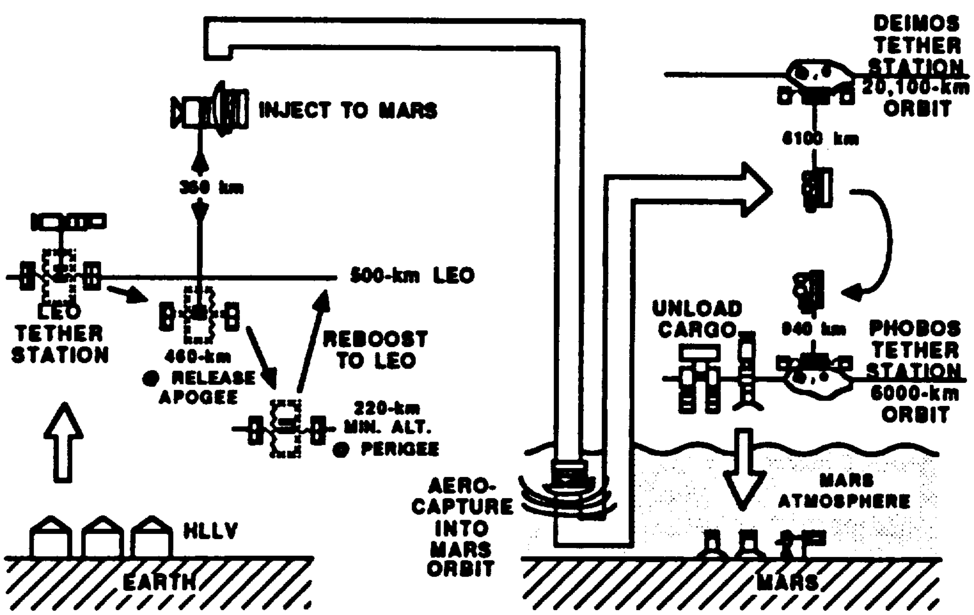

Figure 9-2. Tether Propulsion Mission Scenario

9.2 ASSUMPTIONS

Very little advanced materials technology was assumed in the present study. The tether material was in all cases taken to be Kevlar 29 with a tensile strength of 2700 MN/m^2 and a density of 1540 kg/m^3. All propulsive maneuvers assumed a chemical O_2/H_2 system with a specific impulse of 470 lb_f-s/lb_m. While it may be possible to increase overall system performance by using electric propulsion for orbit raising for the LEO station or for the Earth-to-Mars transit, a chemical system was used to meet the study ground rule of only one "new" technology at a time.

For Earth orbit operations, a piecewise tapered tether was assumed to achieve mass savings with minimal complexity. In all cases, a safety factor of 2 was assumed in determining the allowable stresses. The tethers used in LEO consist of a series of connected segments of different but uniform diameters. For the tethers at Deimos and Phobos, tethers of 3.5-mm constant diameter are assumed.

In the orbital and center of mass calculations only the end masses were considered because in all cases these are the dominant masses. The displacement of the center of mass due to the distribution of mass in the tether itself was considered to be a second order effect and not considered in the analysis. Displacement of the center of mass due to net tidal force was also considered a second order effect from the standpoint of overall system performance and was therefore not considered. However, the mass distribution of the tether was considered in determining the stresses. In general, the long tether segments result in substantial tensile stresses which must be considered in determining the minimum tether diameters and their masses.

9.2.1 Operations in Low Earth Orbit

9.2.1.1 LEO Tether Design. In many cases, one can effect substantial mass savings by using a continuous or piecewise tapered tether. The diameter of a tether is determined by both the maximum tension it must support, as well as the safety factor selected for a given design. In addition to the end masses, the tether must withstand the tension due to its own mass as it orbits the Earth. At any point, the tension will be equal to the integral sum of the gravitational force which is acting radially inwards and the centrifugal force which acts radially outwards. The greatest net force within the tether will occur at the center of mass where these two forces balance. Ideally, one would desire to design a tether in which the cross section at any point is only as large as necessary for the stress at that point. The result would be a continuously tapered tether which, although resulting in the minimum mass possible, would be difficult to manufacture. Analytical expressions for this distribution of tether cross section as a function of location along the length of the tether are derived in detail in the appendix to Reference 1.

Another option is a constant-diameter tether in which the cross section for the entire length is equal to the minimum allowable cross section for a given factor of safety. However, this design is unnecessarily massive. A compromise is the use of a piecewise tapered tether in which a number of segments of uniform, but different, cross section are linked together. Each segment is the minimum diameter necessary to sustain the maximum stress for that span.

In the point design used in this analysis, the tether in low Earth orbit is 350-km long and is divided into four segments. The characteristics of these segments are listed in Table 9-1. In this table, segment number 4 refers to the segment closest to the station. Segment 4 is the segment which sustains the largest stress. The center of mass of the whole system is located within segment number 4 approximately 40 km above the station.

The number of segments is somewhat arbitrary; the larger the number, the closer the approximation to a continuous tether. No specific assumptions concerning fasteners for the segments have been made, but this is not expected to be a major technical issue.

Table 9-1. LEO Tether Characteristics

Segment	Diameter (mm)	Length (km)	Mass (kg)
1	8.604	99.051	8350.3
2	8.619	86.940	7354.4
3	8.824	82.933	7354.4
4	8.925	81.077	7354.4

The total tether cable mass was 30.414 MT with a maximum tension of 84,850 N. While the differences in the segment diameters are not large, the mass savings over a tether of constant (8.925 mm) diameter is 1335 kg.

9.2.1.2 LEO Transportation System Operation. There are a number of factors which characterize the performance of the tether transportation system in low Earth orbit. In a tether-assist launch "cycle", the payload and transfer stages are deployed upward in order to raise the orbit, allowed to stabilize in a radial or vertical orientation, then released. The payload enters a higher-energy orbit while the station enters a lower-energy orbit. At this point, the tether is reeled back into the station and a conventional chemical system is used to boost the station back to its original circular orbit. This station boost assumes a dual-burn Hohmann transfer. The transfer vehicle performs a burn at periapsis in order to reach the final escape velocity.

Table 9-2. Post-Release Earth Orbit Characteristics

Element	Apogee Altitude (km)	Apogee Velocity (km/s)	Perigee Altitude (km)	Perigee Velocity (km/s)
Station	460.1	7.572	227.6	7.838
Payload	3178.8	5.986	810.1	7.959

Parameters such as the payload and station mass as well as the tether length will strongly influence the final orbits to which the station and payload are transferred. One possible danger in operation of a tether transportation system is de-orbiting the station. For this reason, the periapsis of the post-release station orbit is of critical importance. Table 9-2 lists the orbital characteristics of both the payload and the station after release. The point design used in this study assumed a station mass of 500 MT, combined payload and transfer vehicle mass of 64.3 MT and a tether length of 350 km. The initial

circular orbit altitude for the center of mass is 500 km.

For the point design considered, the required velocity change for station reboost was 89 m/s. One can determine an equivalent velocity increment delivered to the payload by calculating the Hohmann transfer velocity increments required to achieve the same final orbit. From the data listed in Table 9-2 this was found to be 673 m/s.

At perigee, the payload and transfer vehicle require an escape velocity of 10.535 km/s. For an excess hyperbolic velocity squared (C_3) of 9.541 km^2/s^2, the required velocity increment is 3.070 km/s, which includes a contingency of 51 m/s for miscellaneous orbital operations.

One issue associated with operation of any tether system is the time and power requirements associated with deploying or retrieving (reeling in) a payload. In general, the required mechanical power is equal to the tension multiplied by the retrieval velocity. When deploying a payload, no power is necessary because the inertial forces acting on the end-mass will act to "unwind" the tether. This has been suggested as a means of power production in conjunction with batteries or fuel cells used for energy storage. However, to retrieve a payload such as a manned vehicle returning to LEO, it is necessary to supply power through a motor to overcome the combined tether and end-mass tension. For simplicity, in this study it was assumed that retrieval velocity is held constant, implying that the power requirement decreases as the tension decreases. The retrieval time therefore represents an upper limit for a given power level. In these calculations, 250 kW of mechanical power was assumed available during retrieval resulting in a time of 33 hours to reel-in a 64.3-MT payload the required 350 km. A solar photovoltaic power system sized for 300 kW_e electrical power was assumed. This implies a motor efficiency of roughly 80 % with additional electrical power used for instrumentation.

9.2.1.3 LEO Infrastructure and Technology Issues. For a tether transportation system to be effective in low Earth orbit, a substantial amount of "dead mass" or ballast is required. The reason for this is that the effect of a payload boost cycle on the station is to lower its orbit. For the point design used in this study, it was found that a 500 MT LEO station was lowered to an orbit having a perigee of 228 km. This is already low enough to be a possible hazard due to atmospheric drag on such a large space structure. Any source of mass could serve as ballast. Rather than a dedicated effort to place such a large amount of material in orbit, a more cost-effective means would involve collection of inactive spacecraft, satellites, and perhaps expended STS-class external propellant tanks.

Table 9-3 gives a mass breakdown for the proposed station facility for low Earth orbit operations. With the exception of the tethers themselves, detailed design of other system components was not considered. The LEO tether station, which was assumed manned, is based on space station generation technology.

As mentioned previously, the propulsion system assumed for the tether station reboost is a chemical O_2/H_2 system as described in Section 1. Scaling equations for this system are the same as the baseline chemical system. After each boost cycle, the station must be refueled with 9.6 MT of propellant in a 1.1 MT (dry) resupply tanker (tankage factor = 0.115) for a total of 10.7 MT per reboost cycle. One possibility is to use a propellant depot as part of the ballast, thereby eliminating the need to refuel after each cycle.

The power system listed in Table 9-3 is that used for tether operations. This system is sized based on a power specific mass of 7.2 kg/kW_e, or twice that assumed for the high-powered SEP vehicles. Other dedicated power systems such as those for communication or science are not listed separately. The mass for these other systems is

assumed to be included in the "structure" mass.

One set of spare tethers is available at any given time and spools are sized by assuming one-half of the tether mass per spool. The tether material assumed was Kevlar 29 and is not considered to be an advanced technology. Lightweight, high strength spools are not considered to present an insurmountable challenge given on-going advances in fiber-wound composites.

One technology area in need of development is that of low weight, high power electrical motors and controllers. No specific design was assumed for this component and the mass listed in the table simply reflects an estimate. An additional issue is that of how to construct such a massive "ballast structure" without incurring too great a penalty in terms of atmospheric drag.

Table 9-3. LEO Tether System Mass Breakdown

LEO Tether System Component	Mass (MT)
Tethers/Spares	60.8
Structure/ Ballast	284.4
Spools (0.5 x 60.8)	30.4
Motors/Controls	10.0
Crew Modules/Life Support	100.0
Power System (300 kW_e x 7.2 kg/kW_e)	2.2
Propulsion System	
Thrusters/Tankage/Feed System	2.6
Propellant (For one reboost)	9.6
Total	500.0

9.2.2 Operations at Phobos and Deimos

9.2.2.1 Phobos and Deimos Transportation System Operation. For the mission scenario considered in this study, the objective was the delivery of cargo to Phobos. A two-tether system was considered for Mars operations and is depicted in Fig. 9-3. Such a tether transportation system at Mars has been considered in previous work by Penzo[2]. It was assumed that the tethers at Deimos and Phobos could be reeled in and transported by rail line or other means from one side of the moon to the other, thus enabling a single tether to operate in an "up" or "down" deployment/retrieval fashion. However, in the cargo Phobos-delivery mission, only the Deimos "down" and Phobos "up" tethers are required. For the manned portion of the mission, the Deimos "up" tether makes it possible to return payloads (e.g., manned vehicles) to Earth; while the Phobos "down" tether permits de-orbiting of landers and retrieval of ascent vehicles.

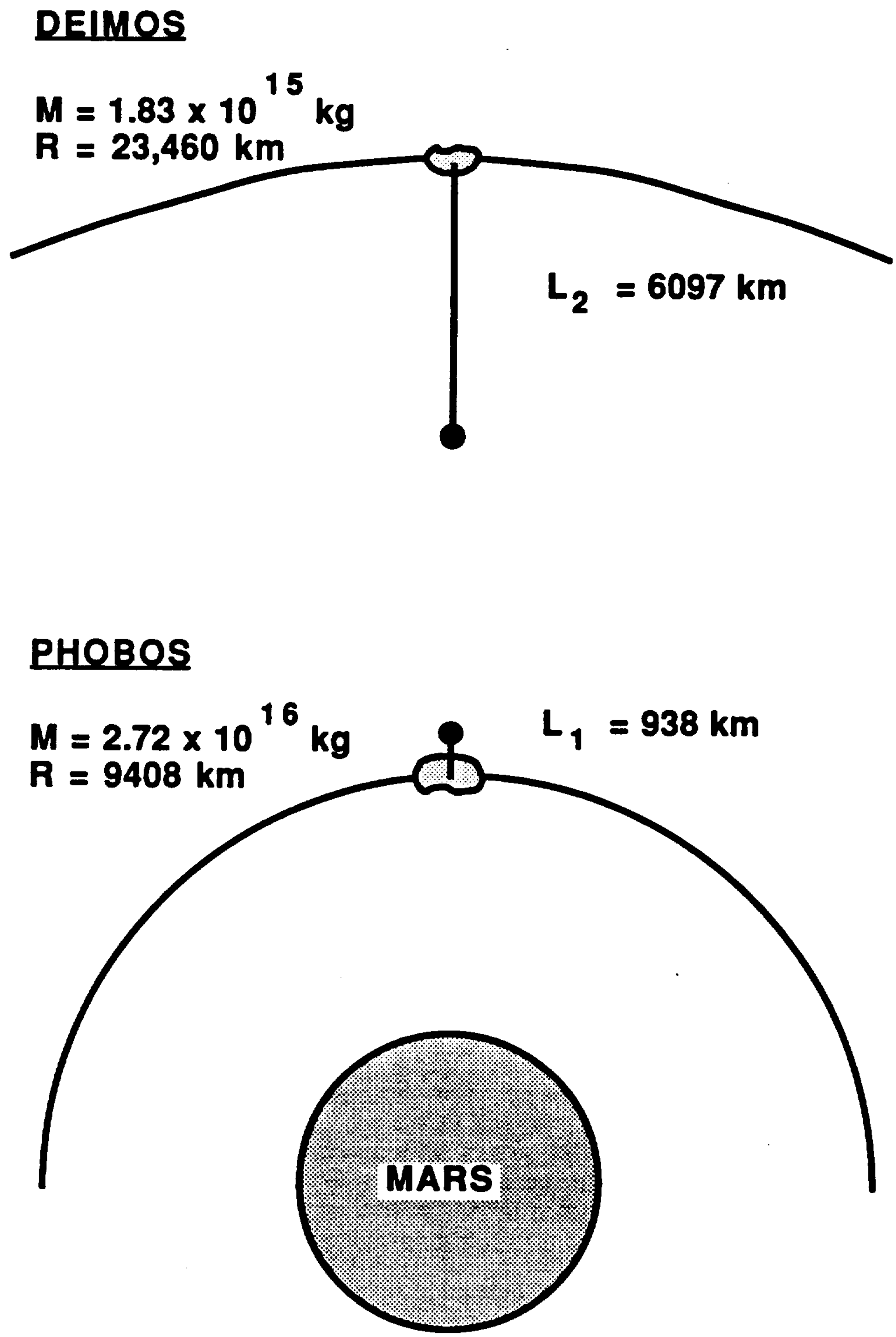

Figure 9-3. Tether Systems at Phobos and Deimos

Assuming that the tether system would be required to support both the cargo and piloted portions of the mission, the tether at Deimos was sized to have a tip velocity equal to the Mars escape velocity of 1.702 km/s at the orbital altitude of 26,174 km which corresponds to the sum of the radius to Deimos and an "up" tether length of 6097 km. It

was found that for Mars orbit capture of the cargo payloads, the use of an aerobraked vehicle rendezvousing with the 6097-km Deimos "down" tether resulted in a lighter vehicle than the options of direct or aerobraked orbit insertion and rendezvous with either the Phobos or Deimos "up" tethers.

Upon entering the Mars system, the spacecraft has a velocity in heliocentric space of 3.162 km/s. It then uses aerocapture at Mars to place itself in an elliptical orbit with an apoapsis altitude corresponding to that of the Deimos "down" tether tip. The vehicle's velocity at apoapsis is 897 m/s; a Delta-V of 103 m/s is required to bring the vehicle in phase with the tether tip which is moving at a velocity of 1.0 km/s. Once captured, the vehicle is raised towards Deimos to a point 2929 km from the moon where it is released. This could be done either by reeling in the tether or by using an elevator-crawler device. At the release point, the vehicle has a velocity of 1.182 km/s. Upon release, the payload enters an orbit with its release point corresponding to an apoapsis of 17,148 km above the Martian surface. The periapsis of this orbit has an altitude of 6966 km and coincides with the radial location of the Phobos upward deployed tether end. The tether tip velocity is 2.346 km/s, which is the same as the orbital velocity at periapsis. It is still likely some maneuvering fuel may be required to complete the docking as well as for contingencies. For this reason, an additional 100 m/s was allocated to Mars operations in determining the propellant loading.

9.2.2.2 Phobos and Deimos Tether Design. The tether capture and release points at Mars were selected based on a specific set of objectives. As mentioned above, the tether at Deimos was first sized to permit the injection of a spacecraft into a Mars-to-Earth transit trajectory when using the tether deployed upwards. When deployed downward, as in the scenario considered here, it enables the capture of an inbound aerobraked spacecraft with a minimum phasing/rendezvous Delta-V. After capture, the spacecraft is raised to a release point. The location of this release point, as well as the Phobos "up" tether length, were determined by requiring that the spacecraft, upon release from the Deimos "down" tether, enter an elliptical orbit whose velocity at apoapsis matches that of the release point, and whose velocity at periapsis matches that of the tip of the upward-deployed Phobos tether. The orbital velocities at apoapsis and periapsis were fixed by the orbital velocities of Deimos and Phobos (i.e., the tether attach points), respectively, so that it was necessary to determine what elliptical orbit (and thus what apoapsis and periapsis altitudes) would be required to satisfy the velocity constraints.

The tethers used at Mars were all assumed to be of constant 3.5-mm diameter. Because of the lower tensions involved, the additional complexity of a piecewise continuous tether, while minimal, was not felt to be warranted. A safety factor of 2 was assumed in the calculation of allowable stress as was done for the LEO analysis. Table 9-4 summarizes the characteristics for the Mars tether system. In this table, "stress ratio" refers to the ratio of maximum stress in a tether to the allowable stress based on a safety factor of 2. These calculations assume a tether end-mass of 20 MT.

Table 9-4. Mars Tether System Characteristics

Tether	Length (km)	Mass (MT)	Tip Velocity (km/s)	Stress Ratio	Tension (N)
Deimos	6097.0	85.06	1.000	0.3753	4874.9
Phobos	937.7	13.08	2.346	0.2738	3525.8

Also, in the calculations of tension for the Phobos upward deployed tether, the gravitational field associated with Phobos has been accounted for. Because the center of mass for the Phobos tether system is located within the moon, the tether and payload are subject to the outward centrifugal force which keeps the tether in tension. The net effect of Phobos' gravitational field is to reduce this tension because it acts radially inward on each mass element. The result is a reduction in the tension by about 2 N.

For Mars operations, it was assumed that the payloads were moved along the tethers by an elevator/crawler type of device. This allows shorter retrieval times because only the forces acting on the payload and elevator/crawler (and not the whole tether) need be overcome. Again, assuming 250 kW of available mechanical power, the constant-speed retrieval times were found to be 2.75 hours at Phobos and 5.95 hours at Deimos. The time at Deimos represents the time needed to move the payload 3168 km from the capture point (tip) to the release point.

9.2.2.3 <u>Phobos and Deimos Infrastructure and Technology Issues.</u> Table 9-5 lists a mass breakdown for the Deimos tether facility. For the most part, the technologies such as spools, motors, and controls are assumed to be the same as those used in low Earth orbit. One difference however is that the Phobos-Deimos stations are assumed to be automated. This requires an advanced, but near-term level of development in areas such as telerobotics.

Table 9-5. Deimos Tether Station Mass Breakdown

Deimos Tether System Component	Mass (MT)
Station Structure/Anchor	20.0
Tethers/Spares (2 x 85.06)	170.2
Spools (0.5 x 170.12)	85.1
Motors/Controls	10.0
Automation/Communication	5.0
Power System (300 kW_e x 16.7 kg/kW_e)	5.0
Earth-Mars Communication	5.0
Rail Link	10.0
Total	310.3

In addition, communication links are required between the tether stations and the planet surface as well as among the stations themselves. As at Earth, the power system is again assumed to produce 300 kW_e electric power with a power system specific mass of 16.7 kg/kW_e (at Mars) and a rail link, or other means of transporting payloads from the "top side" to the "bottom side" of the moon.

Table 9-6 lists a similar mass breakdown for the tether facility at Phobos. Included in the mass list is the rail link for the downward deployed tether option for delivering payloads to the surface of Mars. The Phobos station structure/anchor is less than that of the Deimos station because the Phobos tether system is much lighter.

Table 9-6. Phobos Tether Station Mass Breakdown

Phobos Tether System Component	Mass (MT)
Station Structure/Anchor	10.0
Tethers/Spares (2 x 13.08)	26.2
Spools (0.5 x 26.16)	13.1
Motors/Controls	10.0
Automation/Communication	5.0
Power System (300 kW_e x 16.7 kg/kW_e)	5.0
Rail Link	10.0
Total	79.3

By far the largest challenge in the entire tether-assisted propulsion scenario is the establishment of stations at Deimos and Phobos. This requires a large investment in time and resources; first to get the raw materials there, then to assemble and anchor large structures with their associated machinery to the surface of the moons. Such a complex system would require regular maintenance to remain effective and therefore implies that a reasonable presence of personnel and equipment are already available at Mars.

9.2.3 Aerobraked Chemical (O_2/H_2) System

A two-stage aerobraked chemical (O_2/H_2) system was used in conjunction with the tether systems. This vehicle is simply a re-sized version of the baseline chemical system described in Section 1. The Delta-V required for trans-Mars injection by the first stage is reduced from 3.688 km/s for an all-propulsive system to 3.070 km/s with a tether system. As with the baseline system, a 100-m/s Delta-V is allocated for midcourse maneuvers by the second stage prior to aerocapture. The aerocapture maneuver places the vehicle in an orbit with a 17,360-km radius apoapsis for rendezvous with the tip of the Deimos tether. At apoapsis, the vehicle's orbital velocity is only 0.897 km/s whereas the tether tip is moving at a velocity of 1.0 km/s; thus, the vehicle must provide a Delta-V of 0.103 km/s for rendezvous. Finally, an additional contingency of 100 m/s in Delta-V capability is included for final rendezvous and docking with the tethers at Deimos and Phobos.

Because the mass that can be injected by the LEO tether station is limited, the two-stage vehicle was sized for the minimum-sized 20-MT payload element, resulting in a vehicle (including adaptors and aeroshell) weighing 44.3 MT. Twenty of these vehicles are required to deliver the full 400 MT of payload to Mars. A mass breakdown for this vehicle is given in Table 9-7.

Table 9-7. Transit Vehicle Mass Summary

Element	Mass (MT)	Delta-V (km/s)
Payload	20.0	
Stage 2		
Adaptor (2.5 %)	0.5	
M_{Dry}	1.3	
M_p (Post-Aerocapture)	1.0	0.203
Aeroshell (15 %)	3.4	
M_p (Pre-Aerocapture)	0.6	0.100
Total	6.8	
Stage 1		
Adaptor (2.5 %)	0.7	
M_{Dry}	5.5	
M_p (Earth Escape)	31.3	3.070
Total	37.5	
Total	64.3	

9.3 RESULTS

Figure 9-4 shows the breakdown of total initial mass in LEO for the tether-assisted propulsion system. It is evident that the required infrastructure mass for this option is substantial. The mass required to deliver the Phobos and Deimos stations to their respective orbits (236.7 MT and 923.7 MT, respectively, using vehicles similar to the baseline chemical system), plus the two tether stations is over fifteen hundred metric tons.The total infrastructure required, including the LEO tether station, is 2050 MT. This is, however, a one-time investment which can continue to operate for many applications beyond the initial cargo mission.

Unfortunately, the operating "costs" in mass for the tether system are found to be only 147 MT less than the corresponding baseline chemical system (1499 and 1646 MT, respectively). The reason for this is that the cargo must be broken into twenty smaller 20-MT packages, each of which requires 44.3 MT of propellant and transfer vehicle mass for the Earth-to-Mars transit. This penalty could be reduced if it were possible to increase the payload delivered with each cycle. However, for a given LEO tether length and station mass, increasing the payload mass lowers the altitude the station will attain at perigee after the payload has been released. For the given scenario, a 500-MT station and 350-km tether are used to boost a 64.3-MT Earth-to-Mars transit vehicle and payload. After release, the station enters an orbit with a perigee of 228 km. For this set of conditions, any further increase in payload mass would probably cause the LEO tether station to de-orbit.

Finally, the transit time for the tether assisted propulsion option is essentially that of the baseline chemical system (294 days) because the time required for the tether assist portion of the mission is almost negligible in comparison. The frequency with which payloads can be sent from LEO is determined by the deployment time for the 350-km tether with attached payload.

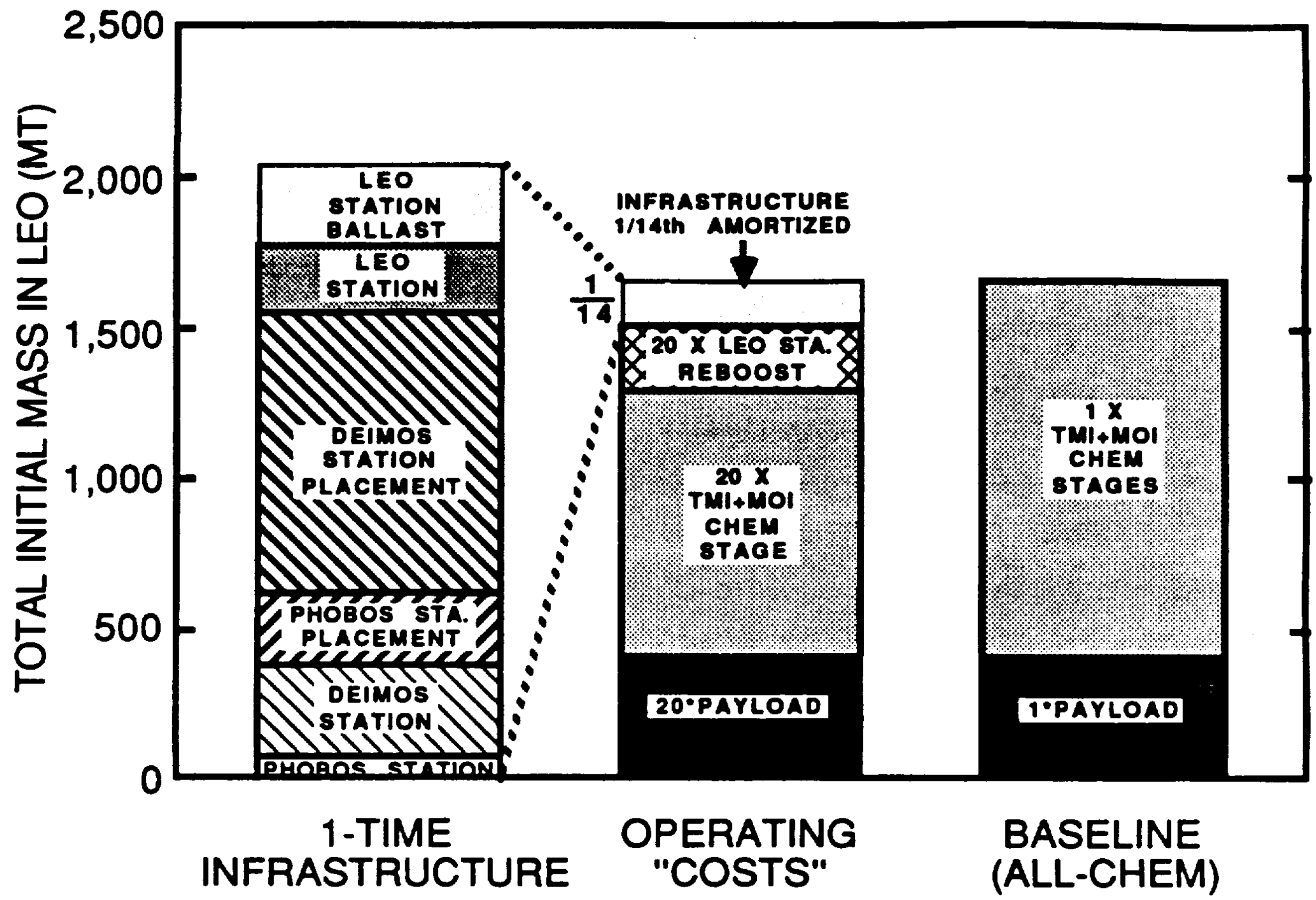

Figure 9-4. Initial Mass in Low Earth Orbit for Tether Propulsion (Earth-to-Mars trip time = 294 days)

9.4 CONCLUSIONS

In the context of the Mars cargo mission considered here, the tether-assisted propulsion system option is not sufficiently better than the baseline chemical system to warrant its use. The operations costs involved with the twenty separate vehicles is only 9 % less than that of the baseline system with a single O_2/H_2 vehicle transporting the full 400-MT payload to Mars. In order to just break even with the baseline chemical system, the tether system infrastructure would have to be "amortized" over 14 cargo delivery cycles. It would be possible to decrease the operational "costs" by increasing the payload mass, which would in turn would require a heavier LEO station to avoid its re-entering the atmosphere.

Interestingly, the LEO tether station only requires 215.6 MT of systems, the remaining 284.4 MT is "ballast". The station mass could be increased inexpensively by adding extra mass such as an orbital propellant depot. An alternative to increasing the mass would be to use the same station mass, but start at a higher orbit.

Either of the alternatives noted above would affect the quantity of propellant required for station re-boost. It is not clear without further study at which point raising the station orbit or increasing the mass is no longer beneficial.

Finally, it should be noted that these conclusions are based on the use of a tether system for the cargo mission. The use of tethers for the piloted portion of the mission should be investigated, because a downward-deployed tether on Phobos can be used to de-orbit landers and "pick-up" ascent vehicles. Similarly, the tether station on Deimos can be used to provide escape velocity from Mars for the return to Earth, thereby greatly reducing the Delta-V required for the trans-Earth injection and the return home.

9.7 REFERENCES

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SECTION 10

CONCLUSIONS

10.1 CONCEPTS WITH MAJOR BENEFITS

If the primary figure-of-merit for the Mars cargo mission is initial system mass in low Earth orbit (IMLEO), then solar sails provide the greatest mass savings over the baseline aerobraked chemical O_2/H_2 system. However, solar sails suffer from having very long trip times. A good performance compromise between a low IMLEO and short trip time can be obtained using 100-MW_e class NEP systems; they can even be both lighter and faster overall than the baseline chemical system. The trade-off between IMLEO and trip time is illustrated in Fig. 10-1 for the various concepts considered in this study.

Such systems may be particularly suited to the piloted portion of the mission, where a premium is placed on trip time. A 100-MW_e SEP system is a close competitor to the NEP system, providing almost as good a performance but without the technological, operational, or "political" constraints of space nuclear power. It is recommended that these three systems continue to be evaluated for both Mars and lunar missions, with the high-powered NEP and SEP evaluated for piloted as well as cargo missions.

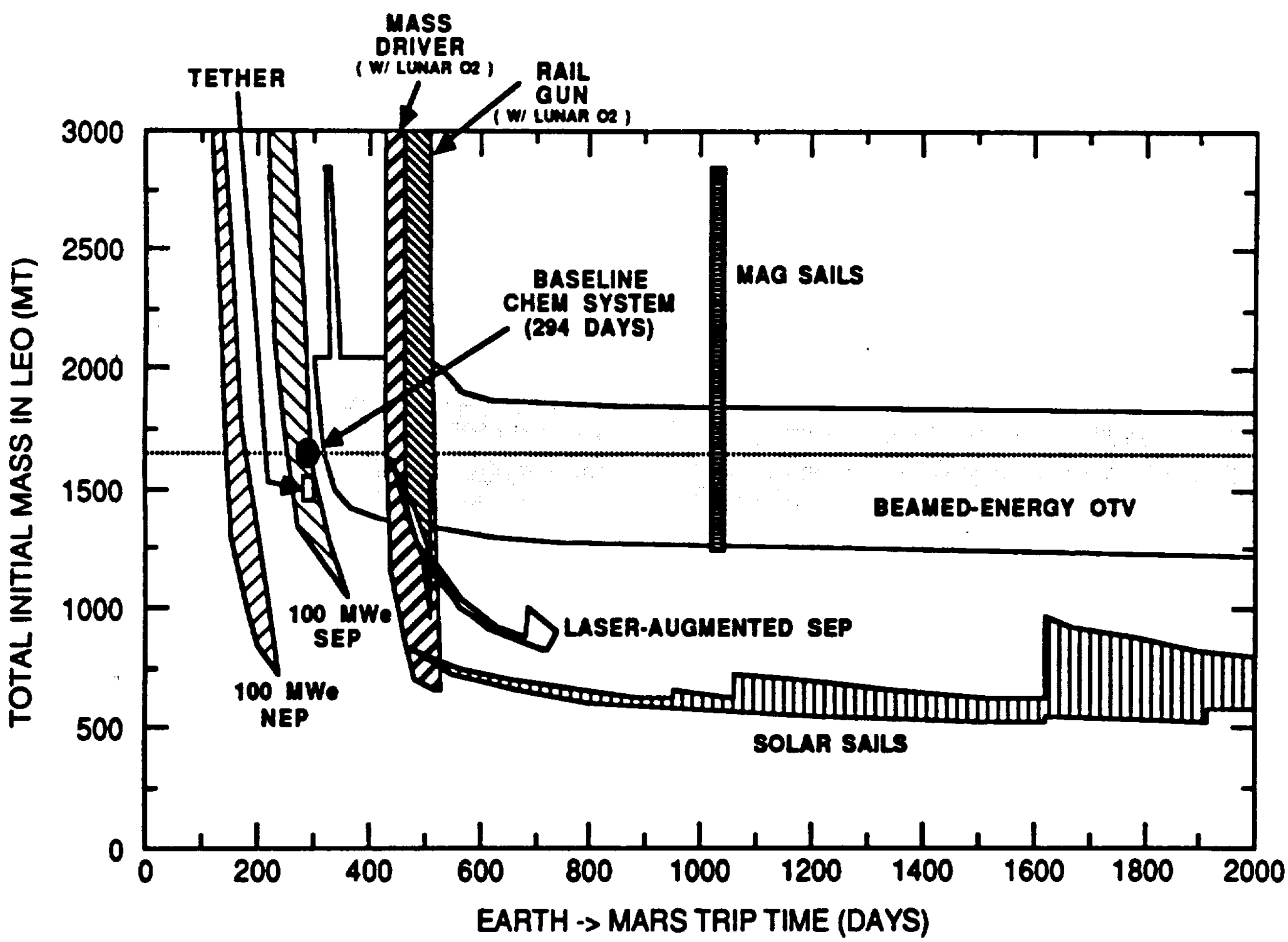

Figure 10-1. Initial Mass in Low Earth Orbit versus Earth-to-Mars Trip Time for the Advanced Propulsion Concepts Considered in this Study

10.2 CONCEPTS WITH MODERATE BENEFITS

Magsails, mass drivers, beamed-energy, and tether concepts were found to have moderate benefits in mass or trip time, but their performance is contingent on several factors which could reduce their effectiveness. For example, the magnetic sail (magsail) concept, like the solar sail, has infinite specific impulse. However, it can only operate far from a planet, thus imposing a large OTV infrastructure overhead. Because the magsail concept does show a potential for significant performance, but is in an early stage of development, it is recommended that additional studies be performed to evaluate its technology requirements and mission benefits more fully.

Mass drivers have a low I_{sp} for the Mars cargo mission, but they do have a high efficiency (electric-to-jet power). They also can make use of any material as propellant. Thus, if copious amounts of "free" lunar O_2 propellant were available, a mass driver operating at modest power levels (10 MW_e or less) could show a mass savings over the baseline system, and do so for trip times on the order of 500 days (70% greater than the baseline system). However, this performance is contingent on the availability of "free" lunar oxygen propellant; without this "free" propellant, the mass driver is not competitive. Also, the performance of the mass driver is very sensitive to the assumed thruster specific mass and to the propellant tankage factor. Thus, it is recommended that the infrastructure "cost" (lunar O_2-production plant, Moon-to-LEO delivery system) of supplying lunar O_2 to LEO be evaluated in detail in future studies.

Beamed-energy concepts were found to provide some benefits in mass when used as OTVs to deploy the payload (with a chemical stage) at GEO altitudes. A laser-augmented SEP vehicle used for the round trip to Mars also provides significant trip time savings because the laser provides a rapid Earth escape/capture. However, all the beamed-energy concepts suffer from the limited range over which power can be beamed (e.g., microwaves to GEO or near-visible light to the Moon). Even the laser-augmented SEP system, which reverts to a normal solar powered SEP far from the Earth, requires very high-powered lasers (10-MW beam or more) to provide any significant trip time savings. Also, the space-based infrastructure (laser or microwave power stations, or orbital relay mirrors) required to support beamed-energy transmission would need to be "amortized" over many users. Thus, beamed-energy concepts may be attractive for use in OTVs. However, these OTVs and their supporting infrastructure would require a broad mission base in which to show an overall benefit. Therefore, beamed-energy concepts should be considered in future advanced OTV studies, but these concepts would provide only modest benefits if applied solely to the Mars cargo mission.

Tether systems show only a small advantage in IMLEO over the baseline system. This is due primarily to the need to split the 400 MT payload into twenty 20-MT segments, each with its own chemical O_2/H_2 stage. Also, there is a significant set-up mass investment which must be "amortized" over many missions. For example, the LEO, Deimos, and Phobos tether station infrastructure must be "amortized" over 14 Mars cargo missions just to break even with the IMLEO of the baseline system. Note, however, that tethers may have greater benefits for the piloted portion of the mission. For example, tethers can be used to lower (de-orbit) landers and raise ascent vehicles. Also, a tether station on Deimos can "sling" a vehicle returning to Earth with Mars' escape velocity, thereby greatly reducing the trans-Earth injection propulsion requirements. These features do not apply to the cargo mission and were not considered in this study; however, they should be evaluated for the piloted portion of the mission in future studies.

10.3 CONCEPTS WITH NO BENEFITS

Finally, two concepts were found to have very poor performance for the Mars cargo

mission. These were solar thermal propulsion and rail guns.

Solar thermal propulsion suffers from having too low an I_{sp} (1200 lb_f-s/lb_m) for this mission. Rail guns suffer from both a low I_{sp} and a low efficiency (electric-to-jet power). They require high powers (50 MW_e) for optimum performance and can only show a mass savings over the baseline chemical system if copious amounts of "free" lunar oxygen are available as propellant in LEO.

10.4 RECOMMENDATIONS

Based on the results of this study, solar sails, 100-MW_e class NEP systems, and 100-MW_e class SEP systems should be considered in detail for application to the Mars cargo mission. Further, 100-MW_e class NEP and SEP systems should be evaluated in detail for the piloted portion of future Mars missions because they have the potential for significant savings in both IMLEO and trip time as compared to the baseline chemical systems. Similarly, tethers should be evaluated for the piloted portion of the Mars mission because they may provide major savings in mass for the Mars-to-Earth portion of the trip. Magsails, mass drivers, and beamed-energy concepts should also be considered for the Mars cargo mission, although their performance will depend on a number of factors (e.g., "amortization" of a space-based laser for laser propulsion vehicles).

Finally, it should be noted that the conclusions reached in this study are highly mission-scenario dependent. Thus, a concept that has no benefit for the Mars cargo mission scenario assumed in this study may show significant benefits for the piloted mission. Similarly, concepts that are not attractive for Mars missions may provide major benefits when used for cis-lunar missions (e.g., LEO-to-GEO OTVs or lunar base missions). Also, different thrusting or trajectory strategies may have a significant impact on performance. For example, a continuous low-thrust spiral planetary escape or capture, as used in this study, usually results in a higher effective velocity requirement for the mission (due to gravity losses) than a multiple-impulse medium-thrust trajectory. Furthurmore, in this study, the concepts were used in a "pure" Mars cargo mission mode with a minimum of mixing of modes. For example, only the beamed-energy concepts were used in a LEO-to-GEO OTV mode due to the limitations in transmission distances. Future studies should consider the option of "mixed" mission modes of operation; such as the use of an advanced concept for a LEO-to-GEO OTV-type transfer followed by trans-Mars injection by a second system. This may be a particularly attractive approach, because a number of previous studies[1,2] have shown that systems with I_{sp}s of 1000 to 1500 lb_f-s/lb_m (e.g., mass drivers, rail guns, solar thermal propulsion, laser/microwave thermal propulsion) can provide major savings in IMLEO as compared to chemical systems, and savings in trip time as compared to high-I_{sp} electric propulsion systems at comparable power levels. Finally, the same advanced propulsion concepts considered in this study for the Mars cargo mission should also be evaluated for the lunar base cargo mission, again with IMLEO and trip time as the primary figures of merit.

10.5 REFERENCES

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2.2.3

A COMPENDIUM OF PROPELLANT OPERATIONS IN THE 1988-89 OFFICE OF EXPLORATION CASE STUDIES

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A COMPENDIUM OF PROPELLANT OPERATIONS IN THE 1988-89 OFFICE OF EXPLORATION CASE STUDIES

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This compendium of propellant storage and transfer situations was compiled to help understand the scope and diversity of propellant operations imbedded in the 1988 and '89 Office of Exploration (OEXP) case studies. There was no intent to critique nor evaluate the individual propellant activities. The primary intent was simply to catalogue them as they were postulated in the official studies. Cases scrutinized include those for 1988 as reported in NASA Technical Memorandum 4075, and for 1989 as defined in the "final" Study Requirements Document, Document No. Z-2.1-002, March 3, 1989.

This catalogue, or compendium, is offered mainly as a labor saving device to readers who wish to analyze any of the cases from some propellant perspective of their own. However, noteworthy situations and issues are discussed, as are conclusions related to the scope of the cases or which appear to be fairly universal. Last minute iterations of the 1989 studies may vary a few of the details, but should have no bearing on the relevance of the compendium or the conclusions.

The table summarizes each unique propellant activity and issue, and their distinguishing characteristics: location, operating mode, gravity, environment and materials being transferred. For simplicity, activities were lumped for several different materials if other variables were alike. The reader should not lose sight of the fact that transfer of distinct propellants can imply far different technologies. For example, in activity #10 the transfer of argon is far simpler than the transfer of cryogenic liquid hydrogen to the same spacecraft. The last column in the table is the list of applicable case studies for each propellant activity. If a given case directly references the propellant related requirement or issue, the case is listed without parentheses. If the case implies the necessity of the requirement or the issue, then that case is listed in parentheses. This is not a shortcoming of case definition, but the result of deliberate minimization of duplicate trade-offs.

A few other points are helpful in reading the table. Where multiple options are listed, option "a" is the baseline specified in the main case description. The other options were treated under section 4, "Special Reports and Studies," of TM 4075. "Radiation" is a specified environment only when nuclear reactors are present and the need for additional radiation protection should be evaluated. In reality, natural radiation is an ever present concern and often far dominates the possible radiation from reactors. For the "standoff" refueling at Phobos or Deimos (#19), the receiving vehicle is postulated to be hovering off the surface. The dust environment is applicable to the storage site, but the vehicle itself is hoped to be dust free. Gravity is described as negative, since the propellant must be pumped up. Where hovering is optional (#20 & #21), gravity is defined as $\pm$. Similar strategy could be used at an asteroid.

The exploration case studies have spanned most possible combinations of factors which characterize propellant operations. Operations could be needed at all orbital locations: low Earth orbit, low lunar orbit, Mars orbit, at libration points, and possibly even in heliocentric orbit. Similarly, many surface types and gravities would be encountered: lunar, Martian and small body. These bodies all have very dusty environments, in sharp contrast to the vacuum clean orbital environments. Debris and micrometeorite risk is high in some locations. Gravities include zero-g, milli-g, one/third-g, and surprisingly the unique negative milli-g. Though human tended operations are widely postulated, both short and long range telerobotic operations could be needed, with strong technological implications. A variety of propellants have been envisioned for transfer: from the difficult cryogenic hydrogen, elusive helium, or very hazardous oxygen-fuel gels, to simple and safe water or tanks of inert gases.

Some elements of propellant operations appear in virtually all instances. Long duration propellant storage emerges as universal in these cases, a technology challenge for cryogenics in particular. A closely related issue is the more favorable "dry" launch of cryogenic storage tanks. If launched "wet," the tanks must be structurally stronger, and consequently insulate more poorly. The desirability of transferring fluids, as opposed to transferring tanks only, is apparent since there can usually be substantial advantages to topping off tanks, even if the tanks themselves had been transferred. The final universal need is to routinely service "wet" systems, a safety concern. The alternative of not servicing wet systems would be to squander large quantities of precious propellants.

As can be seen, diverse propellant operations are needed for any robust lunar or Mars program. Selecting a subset of cases does little to narrow the multiplicity of options. The implications seem quite clear. To prepare for any lunar or Mars initiative, America needs a broad technology development program, and an in-space testing program, for propellant storage and transfer. Major decisions regarding the implementation of any exploration scenario would depend upon the outcome of such development and testing. Delaying such programs would either delay the key initiative dates, or force premature selection of mission design.

Acronyms

ECV	Electric cargo vehicle
HMO	High Mars orbit
h-t	Human-tended
IMLEO	Initial mass to low Earth orbit
i-p	Interplanetary
ISPP	In-situ propellant production
LAV	Lunar ascent vehicle
LCSV	Lunar crew sortie vehicle
LEO	Low Earth orbit
LH_2	Liquid hydrogen
Lib pt	Libration point
LLO	Low lunar orbit
LLOX	Lunar liquid oxygen
LMO	Low Mars orbit
LO	Low orbit
LOV	Lunar operations vehicle
LOX	Liquid oxygen
MCSV	Mars crew sortie vehicle
mm	Micrometeorite
MO	Mars orbit
MPV	Mars piloted vehicle
MTV	Mars transfer vehicle
NEP	Nuclear electric vehicle
NTR	Nuclear thermal rocket
Ph-D	Phobos-Deimos
PTF	Propellant tank farm
SSF	Space Station Freedom
TEIS	Trans Earth injection stage
t-r	Telerobotic
0-g	Zero-gravity

A Summary of Propellant Transfer and Storage Requirements/Issues from the OEXP Case Studies

A. Willoughby, June 1989

#	Short Title	Location	Oper. Mode	Gravity	Environment	Material to be Transferred	Applicable Cases
1	No node option	LEO	h-t	0-g	vacuum	cryogens, fluids	88-1, 89-3
2a	Long storage	i-p + MO	-	0-g	vacuum	LH_2, LOX	88-1, 88-2, 89-3
2b	Long storage	i-p + Ph-D	-	0-g/milli-g	vacuum, dust, + warm body	LH_2, LOX	
3	IMLEO sensitivity	LEO-MO-LEO	-	0-g	vacuum	mostly LH_2	88-1, 88-2, 89-3
4a	MO risks	LMO	-	0-g	micrometeorite	LH_2, LOX	88-1, 88-2, 89-3
4b	MO risks	HMO	-	0-g	vacuum	LH_2, LOX	
4c	MO risks	Ph-D	-	milli-g	dust	LH_2, LOX	
5a	MO transfer options	LMO	h-t	0-g	micrometeorite	stages	88-1, 88-2, 89-3
5b	MO transfer options	Mars vicin.	h-t	0-g/milli-g	location dependent	cryogens	
5c	MO transfer options	Mars vicin.	h-t	0-g/milli-g	location dependent	tanks	
6	MO fluids transfer	LMO	h-t	0-g	micrometeorite	cryogens	88-2, 88-4
7	Tank topping	MO	h-t	0-g	-	cryogens	(88-1),(88-2),(89-3)
8	LEO fuel system options	LEO	h-t	0-g	vacuum	cryogens, fluids	88-2, 89-2, (89-4), (89-5)
9	LEO fuel transfer	LEO	h-t	0-g	vacuum	cryogens	(88-1), 88-2, 88-3, (88-4),89-1, 89-2, 89-3, (89-4), 89-5
10	Hot NEP refueling	LEO	h-t	0-g	vacuum, + radiation	argon, LH_2	88-4, (89-4), (89-5)
11	NEP telerobotic delivery	LLO	t/r	0-g	vacuum	LH_2	88-4, (89-4)

A Summary of Propellant Transfer and Storage Requirements/Issues
from the OEXP Case Studies (Continued)

A. Willoughby, June 1989

#	Short Title	Location	Oper. Mode	Gravity	Environment	Material to be Transferred	Applicable Cases
12	Lunar LLOX transfer	Moon	h-t	.16g	dust	LLOX	88-4, 89-1, 89-4
13	LO T/R transfer	LLO	t/r	0-g	vacuum	LLOX	88-4, 89-1, (89-4)
14	Ph-D ISPP	Ph-D	h-t	milli-g	dust, mm	water, LH_2, LOX	88-4, (89-5)
15	Ph-D telerobotic ISPP	Ph-D	t/r	milli-g	dust, mm	water, LH_2, LOX	88-4, 89-2, (89-5)
16	Ph-D transfer	Ph-D	h-t	milli-g	dust, mm	LH_2, LOX	88-4, 89-2
17	LO transfer	LLO	h-t	0-g	vacuum	LH_2, LOX	89-1, (89-4)
18	Lib. point transfer	Lib pt TBD	h-t	0-g	vacuum	LH_2, LLOX	89-1
19	Ph-D standoff transfer	Ph-D	h-t	neg. milli-g	dust, mm	LH_2, LOX	89-2, (89-5)
20	NEP telerobotic tankerfill	Ph-D	t/r	$\pm$ milli-g	dust, mm	LH_2, LOX, argon	89-2, (89-5)
21	Ph-D NTR refuel	Ph-D	h-t	$\pm$ milli-g	dust, mm, + radiation	water, LH_2, CH4	89-2, (89-5)
22	LEO NTR refuel	LEO	h-t	0-g	vacuum, + radiation	LH_2, water, CH4	89-5, (89-2)
23	"Wet" systems service	any	h-t	any	vacuum or dust	LH_2, other fuels, LOX, O/F gels	88-3, (88-all), (89-all)
24	"Dry launch" cryogenic storage tanks	to LEO, to MO	-	up to 1g	launch stress	LH_2, other cryogens	(88-all), (89-all)

The descriptions below elaborate on the short titles in the summary table. They are sorted by specific case studies and the numbers correlate with those in the table. Repeated #'s identify propellant operations, or propellant issues, that are in essence alike.

FY88 Case 1, (88-1) Expedition to Phobos

(1) No node nor depot in LEO is an option, in fact, the desired option.

(2) Storage time for return trip cryogenic propellants is very lengthy, at least 28 months.

(3) Sensitivity is extreme, conservative tankage factors and high boiloff rates double the IMLEO.

(4) Low Mars orbit (the baseline) is a risky, yet arbitrary, choice for storage of the Earth return propellants. Compared to HMO, LMO has a worse thermal environment, a worse meteorite environment leading to possible propellant loss [catastrophic for the "single tank" configuration], and worse orbital drifts making rendezvous very risky if beacon goes out.

(5) "Docking propellant transfer", though baseline, is not the only option. Other options are elaborated in section 5.2.4. of TM 4075.

FY88 Case 2, (88-2) Expeditions to Mars

(2) Long storage times are needed, similar to above.

(3) Sensitivity to tankage factors and boiloff, creating up to 50 percent increase in IMLEO.

(4) LMO risk factors exist as elaborated above.

(5) There are options to propellant transfer other than docking with the trans Earth injection stage (TEIS).

(6) Mission #3 requires a fluid transfer in LMO since the TEIS is delivered dry.

(7) Topping off the tanks would enhance all three missions.

(8) A complete range of LEO options with respect to SSF, coorbiting PTF and coorbiting platform, is discussed in TM 4075, section 2.2.5.1.

(9) Fueling of MTV with LH_2 and LOX in LEO, with possible intermediate transfers to/from storage tanks, is needed.

FY88, Case 3, (88-3) Lunar Observatories

(9) Straightforward fueling and refueling of LTV in LEO, with LH2 and LOX is required. Reusable LAV's serviced in LEO are an option.

FY88, Case 4, (88-4) Lunar Outpost to Early Mars Evolution

(10) Electric cargo vehicle (ECV) must be refueled with argon in LEO. ECV has a hot radioactive reactor! Humans are in vicinity and may participate.

(11) LH_2 must be transferred from ECV (with hot reactor) to lunar operations vehicles (LOV's) in LLO. Baseline operations are automated or telerobotic. Human-tended operations impose significant penalty and possible risks.

(12) Transfer of LLOX to lunar vehicle or to storage is performed while on lunar surface. These are human-tended operations in dusty lunar-g environment.

(13) LLOX must be transferred in LLO onto Mars piloted vehicle (MPV) (without its crew) while attached to an ECV with hot reactor. Baseline is automated or telerobotic. Human-tended operations impose significant penalty and possible risks.

(14) Phobos propellants are produced and stored in a human-tended mode in a milli-g, high dust environment.

(15) Automatic/telerobotic vs. man-tended operations is an unresolved question for Ph-D propellant operations in case 88-4. The automation and telerobotic options are covered by this item.

(16) Ph-D propellants are transferred to the MPV on Ph-D in a human-tended mode.

(6) Terrestrial propellants are transferred in LMO to Mars crew sortie vehicle (MCSV), in a human-tended mode.

FY89 Case 1, (89-1) Lunar Evolution

(12) LLOX is transferred to lunar crew sortie vehicles (LCSV's) while on lunar surface, through the experimental phase of the case study. Human-tended operations in dusty lunar-g environment.

(13) Propellants (LH_2/LOX) are transferred from cargo STV to unmanned vehicles in LLO. Baseline operations are automated or telerobotic. Human-tended operations impose significant penalty.

(17) Propellants (LH_2/LOX) are transferred from STV to manned vehicles in LLO. Human-tended operations are baselined.

(9) Propellants (LH_2/LOX) are transferred to STV and other vehicles in LEO. Human-tended operations are possible.

(18) Propellants are transferred to vehicles or depot at Earth-Moon libration point. Human-tended operations are baselined in operational phase (2014) of the case study.

FY89 Case 2, (89-2) Mars Evolution

(8) A range of LEO options exist to fuel all-up Mars vehicle. Human-tended operations are baselined.

(15) Water, LH_2 and LOX are produced and stored on Ph-D. Operations occur in milli-g, high dust environment, robotically controlled from Earth with long communications delays. There is no direct human presence postulated.

(19) Propellant is transferred to Mars crew sortie vehicle (MCSV) in vicinity of Phobos with possible use of tethers to stand-off Phobos. Human-tended operations.

(16) Ph-D propellant is used to fuel Mars piloted vehicle (MPV). Human-tended operations in milli-g, dusty environment.

(20) Filling of NEP-propelled freighter/tanker at Phobos or vicinity on sixth flight (2014) using telerobotic operations. Differs from (15) in that tanker may stand off Phobos. Differs from (19) by use of telerobotics from Earth, with attendant long communication delays.

(21) Possible human-tended refueling of nuclear thermal rocket (NTR) at Phobos. Fuel could be hydrogen, water or methane. Reactor will be in cool down phase, time not specified.

FY89 Case 3, (89-3) Mars Expedition

(1) No node and no depot in LEO is an option. Propellant transfer in LEO is a requirement, whether or not a node or depot exists.

(2) Total storage time for return trip propellants is lengthy, 27-33 months.

(3) Sensitivity of IMLEO to tankage factors and boiloff rate is probably very high.

(4) LMO of 500 km circular orbit is a very risky, yet arbitrary, choice for storage of the Earth return propellants (and hardware) for reasons enumerated in case 88-1.

(5) There are options for propellant transfer at Mars other than docking with the trans-Earth injection stage (TEIS).

(7) Topping off the tanks would be a mission asset.

FY89 Case 4, (89-4) Lunar Oasis

Propellant transfer requirements not yet defined. They are expected to include a subset of transfer requirements from FY88 Case 4, exclusive of LLOX export. Therefore, only operation (12) is essential. Operations (8), (9), (10), (11), (13), and (17) are possible.

FY89, Case 5, (89-5) Asteroid

Propellant transfer requirements not yet defined. They are expected to be a subset of transfer requirements from the FY88 expeditionary cases. Only operations (8) and (9), LEO refueling and LEO options trade, seem essential. Operations and issues related to the follow-on propellant production at asteroids would be similar to those at Phobos or Deimos. These are (14), (15), and (19). Nuclear electric and nuclear thermal vehicles are particularly appropriate for asteroidal missions. This may involve propellant transfer operations (10), (20) or (21), or possibly a new case:

(22) NTR will be refueled at LEO, with a reactor in cool down condition. Humans are in vicinity even if operation were not human tended.

Generic Issues, All case studies:

(23) Servicing a "wet" propulsion systems, pertinent to all reusable vehicles and repair of expendable vehicles. The residual propellants present a potential hazard to service personnel and facilities. Venting to vacuum, one possible solution, could be a costly waste.

(24) Cryogenic tanks in a launch environment. If tanks are launched "wet," the weight and launch stresses require additional structure and compromise thermal insulation. Subsequent boiloff losses often would favor separate cryogenic storage tanks which have been launched dry.

2.2.4

FUEL SYSTEMS ARCHITECTURE ASSESSMENT

by

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Fuel Systems Architecture Assessment

The objective of this multi-year study is to define cryogenic fluid management architecture options in terms of launching, storing, and transferring throughout the Earth-Moon-Mars system in support of the node and transfer vehicle concept definitions .

Background

Overall Approach. Study of cryogenic propellant management spans all of the case studies and, as such, is being performed as a broad trade study. The study was performed by defining a trade space to develop fuel system architecture options based on case study requirements. The options were evaluated based on defined evaluation criteria to narrow the options for more detailed assessment.

Trade Space. The defined trade space, as shown in Figure 1, involves:
- the node location
- the extent of manned intervention required for on-orbit operations
- Earth-to-Orbit transportation capabilities (these capabilities include the mass, diameter, length, and volume delivery limitations to LEO, delivery location, the orbital adjustment capabilities of the delivered payload, cryogenic propellant payload delivery limitations, etc.)
- mission requirements that must be accommodated
- the configuration or transfer characteristics (represents the fluid transfer, acquisition method, or initial fluid state that would have a major impact to systems required)

Focused Approach. For FY 1989, it became clear after developing the trade space matrix and evaluation criteria required for evaluation that assessment of the entire trade space would be extremely time consuming and top-level, given time and man-power constraints. Consequently, a focused approach was utilized to provide depth on a few representative concepts that could provide insight for future trade space reduction or emphasis. It was decided to concentrate on defining free flying concepts for Low Earth Orbit (keeping propellant away from manned activities) utilizing the largest ETO vehicle option (Shuttle-Z) and demanding propellant requirements from Mars Evolution case study with additional requirements for unmanned missions (GEO, planetary, etc.). The techniques for transfer involved a zero gravity and a thrust-settling-assisted transfer. These two techniques will drive differences in system configuration, allowing their impacts on propellant management to be compared.

Mars Evolution Focus

Key Assumptions.

1. The propellant quantities were derived from Mars Evolution case study's mission 1 (600t) based on transportation integration agent cycle 2 data.

2. Additional propellant quantity for unmanned GEO and planetary STV transportation (90t) was based on the Civil Needs Data Base. This quantity was included as a delta increase to the case study requirements.

3. The ETO capability assumed for delivery of concept elements and propellant was a "Shuttle-Z" providing 140t lift capability with a 12.5m diameter by 25m long payload envelope launched 4 times per year.

4. Mass estimates were for single-string systems; redundant system masses were not estimated. No margin for contingency was included.

5. The maximum transfer time was constrained to one 8 hour shift.

Trades. Various trades were performed to determine the most efficient and least complex concepts for propellant management. Some of the key trades involved assessments of the storage tank configurations, structural configuration, pressure system for propellant transfer operations, and debris protection.

The structural configuration options consisted of the 5 meter truss to be used for Freedom, a large box truss that would be tank configuration specific, and a tetrahedron structure. The tetrahedron structure offered the lowest mass due to its inherent structural strength characteristics and provided an opportunity to minimize assembly EVA.

The storage tank configuration options explored consisted of a single, very large cylindrical tank, common spherical tanks, and Space Shuttle sized cylindrical tank sets previously defined by General Dynamics during the Long Term Cryogenic Storage Facility study performed for MSFC. Common spherical tanks were chosen based on low boiloff, structural configuration compatibility, lowest complexity, on-orbit assembly operations, and ground handling implications.

Propellant pressurization can be accomplished by various methods, such as helium pressurization, but due to the large quantities of propellant for the Mars transfer vehicle that must be transferred, a method must be developed that is efficient and reliable without introducing unwarranted complications and safety hazards. All options addressed utilized the propellant gases to minimize logistic requirements and high pressure storage requirements. The options consisted of a real time (produced when needed) gas generation using heating elements to force gas production, a stored boiloff system using conditioned stored boiloff, and a turbo pump that utilized a gas generator to drive the pump. Based on determination of the energy consumption, systems required, weight, complexity, safety, and overall efficiency, the real time pressure generation system was chosen.

Orbital debris protection is becoming an ever increasing design consideration for Low Earth Orbit spacecraft. The use of a dual bumper shield was investigated for both the entire space structure and individual tank and line protection. There was not a major mass difference between the two methods, but the common shield required assembly EVA or complicated deployment techniques. The individual tank and line protection was chosen for inclusion in the concept.

Concept Description. The selected concept to support the Mars Evolution Case Study is shown, along with its characteristics, in Figure 2. Propellant transfer operations could be accomplished with zero-g transfer methods or via thrust induced propellant settling in this depot configuration. The depot concept consists of four regular tetrahedron substructures (1 oxygen, 3 hydrogen) that include the spherical propellant storage tank, high pressure gaseous propellant storage tank, pressure generation system, and a service substructure. The service substructure contains a photovoltaic array (7 KW), batteries, fuel cells, GN&C, refueling nozzles, docking provisions consisting of two attach points at the refueling nozzles, and a strongback to accept a third attach point to permit varying size propellant delivery and mission vehicles. The concept utilizes gaseous hydrogen/oxygen attitude control and reboost engines (also used for the thrust induced gravity) to take advantage of the cryogenic propellant boiloff and eliminate the need for a separate propellant, such as hydrazine (Figure 3).

Each tetrahedron substructure would be launched with the storage tank integrated with its thermal (120 layers of insulation) and debris shield protection, plumbing, and pressurization tankage. The tetrahedron structure will provide the support required for launch based on NSTS launch load conditions. The hydrogen storage tanks are launched full. The oxygen storage tank is launched empty. LOX supply tankers are launched as an integral element of each assembly launch manifest. The total depot system, including propellant, is launched in six "Shuttle-Z" equivalent flights (Figure 4). The substructures will be attached with the use of corner guides and a power winch to lower the amount of EVA required. The structure will require temporary struts to stabilize the structure during the assembly phase.

The depot has three operational phases: depot resupply, STV fueling, and storage. An assessment of the operational procedures for each phase was performed. The resupply of the depot and STV fueling operations are very similar, with the depot providing all services (e.g. pressure for tanker to transfer propellant to depot). An initial evaluation of the transfer operation indicated that due to the propellant fluid motion along with large center-of-gravity shifts, active control was needed throughout the procedure. Thrusting through the reboost engine to provide this control was also sufficient for propellant settling (10^{-4} g's) and provided a part of the reboost energy required. During the storage phase the use of a thermodynamic vent system (TVS) is required to maintain tank pressure without expelling liquid in the process. A zero or low gravity liquid acquisition device will be required for the TVS to provide given initial conditions to the system and avoid complex control systems. Settling of the propellant by thrusting to develop a defined liquid/gas interface for pressure control by gas expulsion will consume enormous amounts of propellant making this method impractical.

Definition of the subsystems and the operational procedures for the resupply of the depot and an STV allowed the determination of the propellant utilization for these operations. Table 1 provides a summary of the propellant losses during the various operational phases of the depot for the first Mars mission after Depot assembly. The largest loss is due to operational losses associated with attitude control and reboost of the facility. Alternative configurations and methods of providing gravity-gradient stabilization (to minimize attitude control propellant expense) will be investigated as part of future studies. The transfer-in-and-out losses include chilldown of the lines, power and pressure generation for transfer, and center of gravity control thrust.

TABLE 1 - PROPELLANT UTILIZATION SUMMARY

	LH2 (Kg)	LO2 (Kg)	TOTAL	
LAUNCHED	118,000	629,400	747,400	
ASCENT LOSSES	2,230	3,750	5,980	(0.8%)
BOIL OFF*	6,000	4,040	10,040	(1.3%)
TRANSFER IN LOSSES**	1,120	5,140	6,260	(0.8%)
OPERATION LOSSES***	8,870	41,400	50,270	(6.7%)
TRANSFER OUT LOSS (TOTAL)	1,480	12,100	13,580	(1.8%)
TOTAL DELIVERED (After 18 Months)	98,300	562,970	661,270	

* Hydrogen Boiloff = 0.28% /month; Oxygen Boiloff = 0.04% /month

** Initial Fill Only (Includes Chilldown, Power Generation, and CG Control) Subsequent Depot refuelings would increase transfer in losses (H2 by 1936 Kg; O2 by 5,390 Kg)

*** Attitude Control (Based on 1 engine @ 2% Duty Cycle) and Reboost

Technologies. Definition of the depot and the subsystems required provided insight into the required technologies. The technologies identified were:
- cryogenic fluid management
- autonomous rendezvous and docking
- automation and robotics (A&R) for assembly and maintenance
- zero net positive suction head cryogenic fluid pumping

The cryogenic fluid management technologies encompass a large number of new technologies that are essential for successful depot operations. These technologies can be divided into four major areas: storage, supply, transfer, and fluid handling.

Storage. Long term storage of cryogenic fuels will be essential for minimizing propellant losses between missions. Confident system design criteria must be established for development of future cryogenic storage thermal systems to provide low conductive structural supports, high efficient multi-layer thermal insulation protection, minimum system weight, understanding of system integration (thermal/structural), high reliability, and repeatable fabrication techniques. Controllability of the cryogenic storage system operating conditions must be developed along with minimizing temperature stratification of the fluid and avoid unpredictable pressure surges. The effects of launch environment (vibration, acceleration, pressure differential), space environment (debris, micrometeorites, atomic oxygen), degradation due to pre-launch purge systems, and ground handling of the thermodynamic protection system must be understood.

Fluid Handling. The capability to handle large quantities of cryogenic fuels safely in a low gravity environment without causing major dynamic control problems to the overall system is essential. The ability to predict fluid motion to allow development of an acceptable attitude control system is critical for a depot. Understanding of the gravitational environment effects, the need for and impact of baffling, flow induced sloshing, and impact of center of gravity shifts resulting from liquid transfer operations are required.

Supply. The capability to supply cryogenic fuels safely and efficiently in a zero or low gravity environment is one of the most difficult, but extremely essential, capabilities required for performing transfer operations in space. Methods will need to be developed for providing the required pressure differential and liquid subcooling while utilizing minimum power and propellant losses. Understanding of the pressurant temperature effects; pump/compressor system complexity, reliability, and efficiency (minimize heat addition to transferred fluid); pump cavitation criteria; and acquisition/pump interactions must be established. The method utilized for acquisition of the fluid must be efficient, provide minimal thermal disturbances, meet outflow demands, and minimize residual propellants.

Transfer. The capability to transfer cryogenic fuels safely and efficiently in a low gravity environment will be essential for resupplying a fuel depot and filling mission vehicle fuel tanks. The technology will enable single phase liquid transfer with minimal propellant losses during transfer line and tank chilldown, determine the proper liquid injection technique and sequence into the receiver tank, understand the effects of low gravity or acceleration environment on transfer operations, prevent inadvertent venting of propellant during the fill process, provide accurate mass flow measurements, and most importantly provide predictable tank filling capability.

Observations. Upon completion of the evaluation of this preliminary depot concept for LEO, some observations can be determined:

a. Boiloff losses for storage of propellants for the demanding Mars missions are not the only source of concern. The propellant losses due to operations for stability and reboost consume a considerable amount along with the transfer in (resupply) and out (mission vehicle fueling) losses for the depot.

b. Boiloff utilization for attitude control, pressurant, and reboost provides an effective and efficient method for providing essential operational capability. All the boiloff can be used in a productive manner. The need for reliquefaction of boiloff is unnecessary and would only add complications to the overall system.

c. Power requirements can be much higher during the transfer operations, especially if the transfer operation must be accomplished in a relatively short time span.

d. Thrusting may be required during transfer operations to maintain control of the vehicle/depot configuration. The thrusting during transfer can be adequate for propellant settling at the sump, but a low gravity liquid acquisition device will still be required for storage operations.

Mars Expedition

In terms of fuel systems in support of the Mars Expedition case study a top-level assessment of the major mission drivers' and constraints' impacts on the developed infrastructure to support the missions was performed.

Mission Drivers. The major drivers on the fuel system architecture for the Mars Expedition missions are as follows:

a. All space vehicles are expendable.
b. The ETO transportation must support propellant exchange/transfer.
c. No orbital nodes are required.
d. Design for 1995 technology.

Case Study Implications. An assessment of the mission drivers and the defined STV configurations developed based on the use of the "Shuttle-Z" ETO concept lead to the conclusion that the Trans-Mars Injection (TMI) stage, also the Shuttle-Z's third stage, should be launched with a refueling tanker. The refueling tanker would resupply the TMI during third stage ascent to take advantage of the acceleration-induced gravity and minimize tanker storage time. This will provide the minimum ETO launches with minimum transfer losses, but at the cost of increased boiloff (which could be effectively utilized as stated previously) and dry mass.

The propellant tanks for Mars operations require on-orbit fueling due to manifest and design constraints. The preferred method developed was to add the transfer capability (pumps, pressurization, liquid acquisition in zero gravity, and refueling couplings) to a TMI stage to perform this operation, as well as replacing the boiloff that has occurred in the TMI stages since their launch.

The technologies required for TMI stage propellant storage, transfer, and management will not be available by 1995 per the current Pathfinder schedule.. Acceleration of the flight experiment program must occur, otherwise alternative methods must be utilized or higher risk incurred.

Lunar Evolution

In terms of fuel systems in support of the Lunar Evolution case study a top-level assessment of the impacts of the major mission drivers and constraints on the developed infrastructure to support the missions was performed.

Mission Drivers. The major drivers on the fuel system architecture for the Lunar Evolution missions are as follows:

a. All space vehicles are reusable. LOX/LH2 propulsion for all transfer and ascent/descent vehicles.
b. The ETO transportation must support propellant exchange/transfer.
c. No orbital nodes other than Space Station Freedom.
d. Space Station Freedom will not be used to store main-stage propellants.
e. Lunar LOX is one-half that needed for roundtrip of cargo or crew.

Case Study Implications. An assessment to determine the preferred LEO propellant refueling option based on the mission drivers and the defined STV configurations looked at three options: tanker resupply for each mission, temporary depot based on Shuttle-Z third stage, or a permanent free-flying depot structure. The permanent depot satisfied more of the evaluation criteria, but violated the case study groundrules. The temporary depot provides similar advantages (low boiloff, minimum ETO launches, ETO schedule independence) with the addition of commonality for a Mars mission, but requires the development of an HLLV.

Lunar oxygen produced for the ascent/descent vehicles is being considered to lower the dependence on Earth launched propellants. But, hydrogen propellant must still be provided for the propulsion systems due to the scarcity of hydrogen on the lunar surface. An assessment of two options was made to determine the most effective method for hydrogen transportation to provide the required propellant. The two options were defined for hydrogen transportation are: 1) transport spherical tanks as cargo to Lunar orbit during the cargo missions (manifest of cryogenic propellant with the crew provides limited capability and is an unneeded crew hazard), 2) increase the hydrogen tank volume to manifest the propellant cargo with the mission propellant. The first option required a hydrogen manifest of 14 metric tons on the cargo mission in tank sizes ranging from a single 7.3 meter diameter tank to four 4.6 meter diameter tanks. The second option required a manifest of 7 metric tons per mission (cargo and piloted) within the vehicles tanks with only one meter increase in diameter. The preferred method is an increase in transfer vehicle tankage with propellant transfer performed in Lunar orbit.

Summary

Past efforts have provided insight into various aspects for defining the fuel system architecture that may be required for future manned exploration missions. In particular:

- propellant losses are not limited to boiloff and must be factored into future designs to minimize all losses
- operational losses (attitude control and reboost) can have a significant impact on propellant losses for on-orbit storage of large quantities of propellant
- boiloff loses can and should be used to the depot's advantage, eliminating the need for reliquefaction systems
- required transfer times can have a significant effect on surge power requirements
- expansion of the transfer vehicle's tankage of propellant may be the most efficient method for propellant logistics to the Moon
- fluid management technology will be required for the depot and transportation vehicles
- low gravity propellant settling may be the most attractive method for propellant transfer

Future Trades

- Gravity-Gradient vs. Minimum Assembly
- Tank Exchange vs. Transfer
- Space Station Freedom vs. Free-flyer

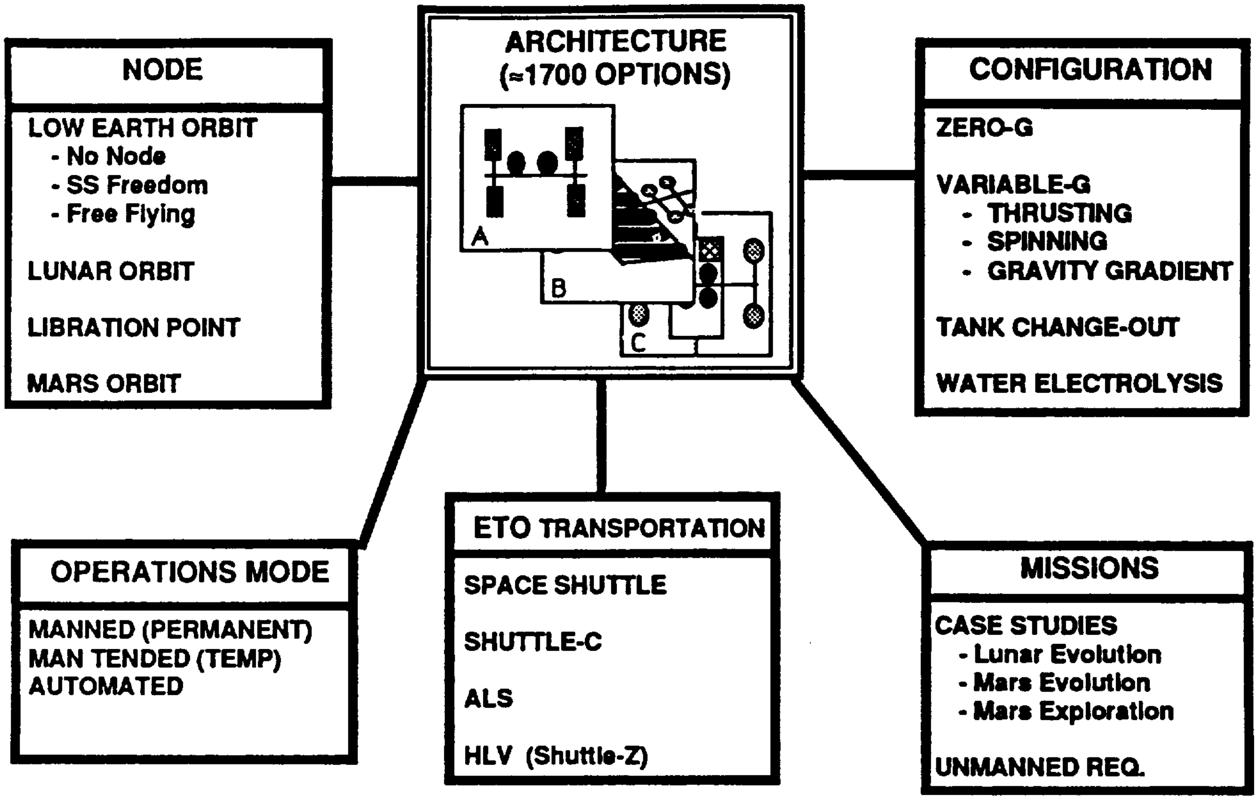


Figure 1 - Defined Trade Space

Dry Mass	44,500 kg
Propellant Type	LOX/LH_2
Mixture Ratio	6:1
Storage Tanks	
Hydrogen	
Number	3
Size (spherical)	10.5 m Diameter
Propellant Capacity	117,300 kg
Oxygen	
Number	1
Size (spherical)	10.5 m Diameter
Propellant Capacity	625,000 kg
Total Mass	792,000 kg

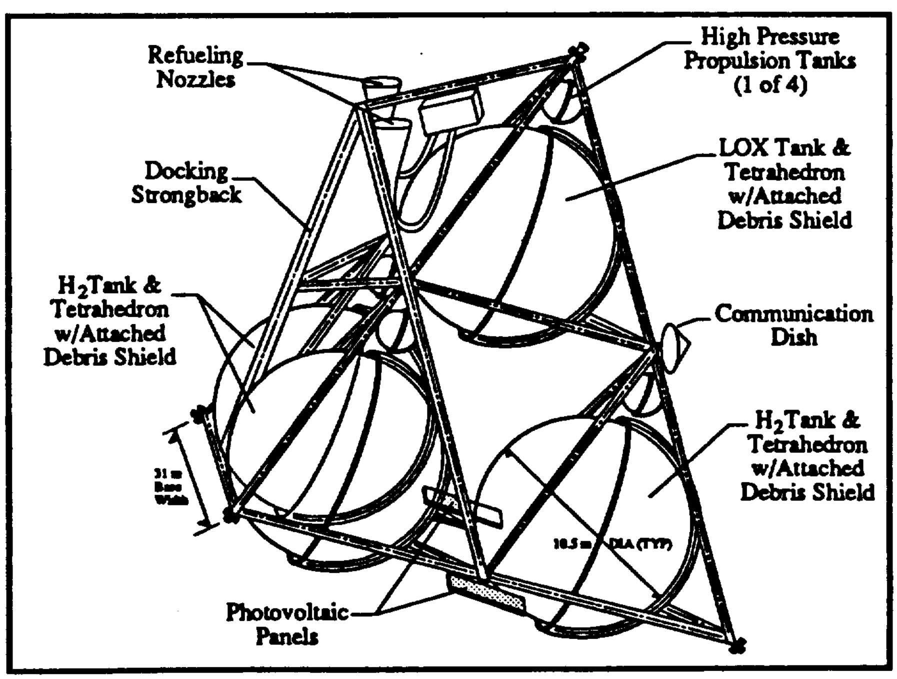


Figure 2 - Preffered Configuration for Reduced Matrix

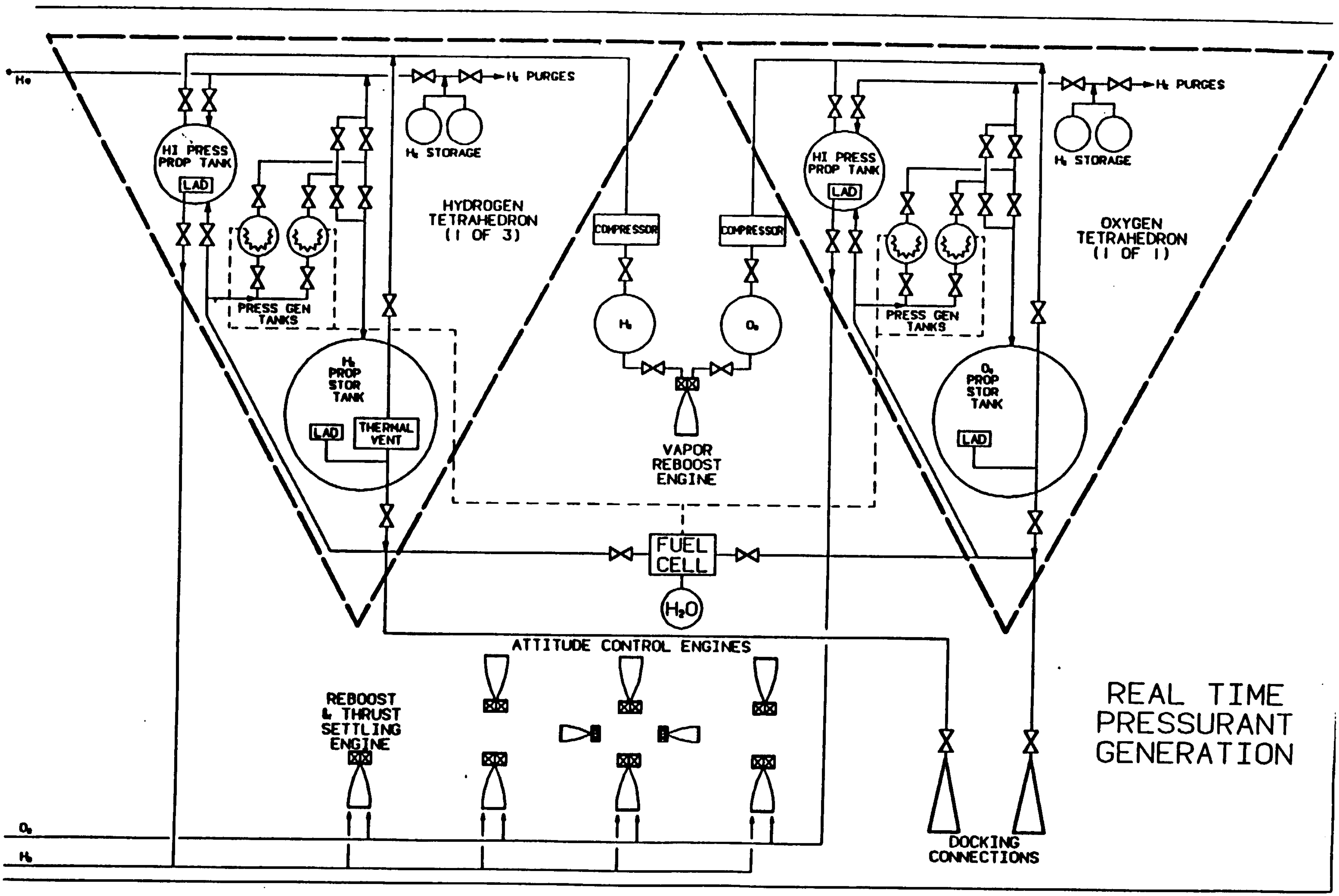


Figure 3 - Flow Diagram

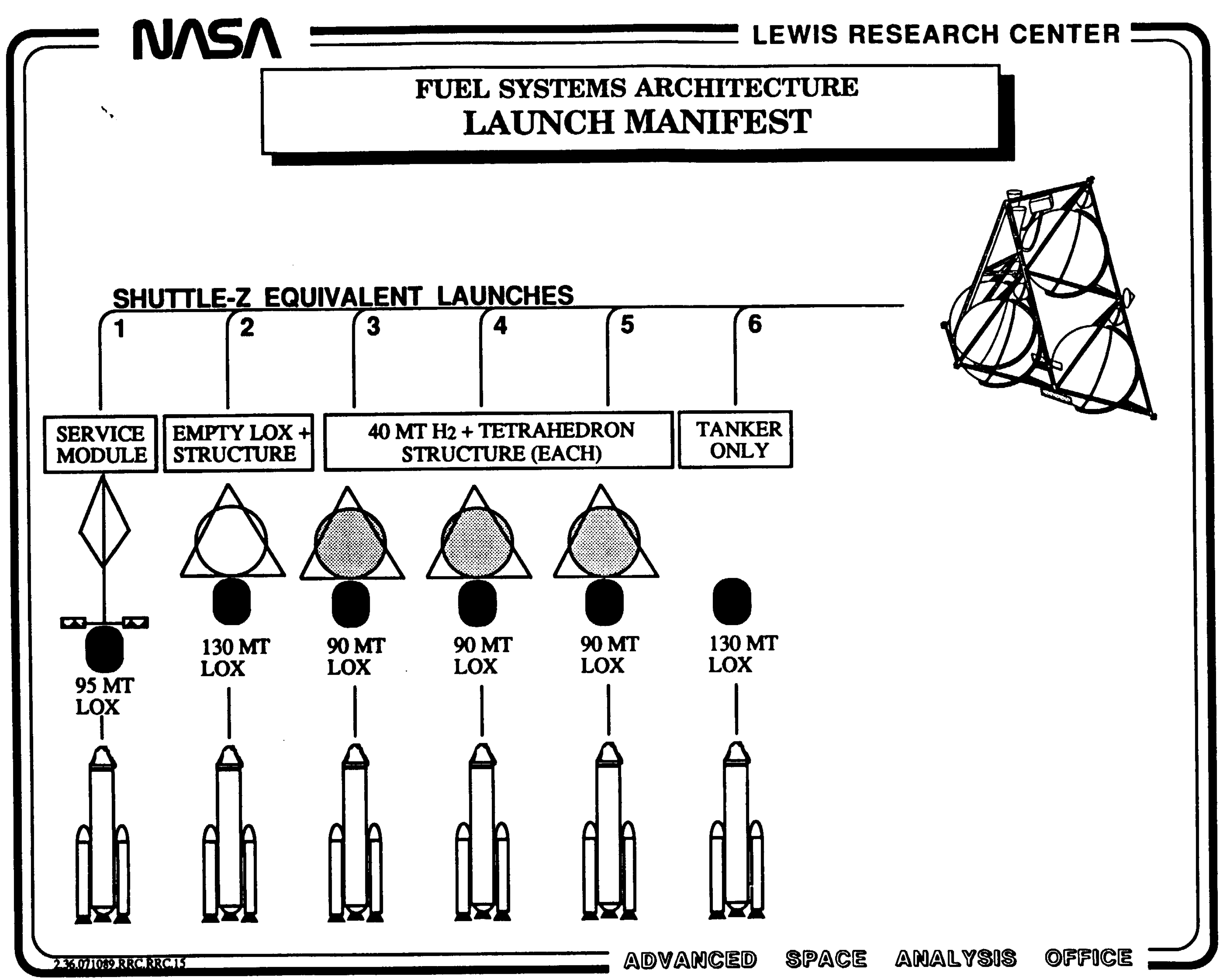


Figure 4 - Launch Manifest

2.2.5

NUCLEAR ELECTRIC PROPULSION SYSTEM ASSESSSMENTS

LEE MASON

ADVANCED SPACE ANALYSIS OFFICE
LEWIS RESEARCH CENTER

OEXP WORKING GROUP #4
ST. LOUIS, MISSOURI
JULY 13, 1989

Background

NASA's Office of Exploration (OEXP) has been involved in examining potential long range scenarios for human exploration of the solar system. Their approach has been to develop a series of case studies which establish overall characteristics of possible exploration missions. A mission which has received considerable attention is an evolutionary exploration of Mars.

The overall objective of this study centered around the selection of a reference mission for the NEP cargo vehicle in the Mars evolutionary case study. This was accomplished through the assessment of power and propulsion system alternatives, and the analysis of mission scenario options.

In addition to the study team listed, members of the LeRC Low Thrust Propulsion Branch and MSFC Advanced Programs Office participated in the design process. MSFC members were asked to review all vehicle concepts and comment on system integration issues.

The mass of the power plant on nuclear electric vehicles is a critical driver in assessing vehicle performance. For that reason, a great deal of emphasis was placed on the mass results for the various power system options. The majority of the power and propulsion mass and performance estimates were generated through the use of LeRC computer models. However, additional computer models were supplied by Oak Ridge National Laboratory (ALKASYS) and Sandia National Laboratory (RSMASS). GE made use of their own optimization and neutronics codes in identifying SP-100 scale-up characteristics.

The mission analysis work was performed in house using the low thrust optimization codes, VARITOP and CHEBYTOP.

BACKGROUND

- STUDY TEAM
  - LEE MASON, KURT HACK, JIM GILLAND; LeRC, ASAO
  - HARVEY BLOOMFIELD; LeRC, POWER SYSTEMS INTEGRATION OFFICE
  - GENERAL ELECTRIC; SAN JOSE

- OBJECTIVE
  1.) PROVIDE BASELINE POWER AND PROPULSION SYSTEM CONCEPT AND MISSION SCENARIO FOR NEP CARGO VEHICLE IN MARS EVOLUTION CASE STUDY (SUB IA)
  2.) DEVELOP MMW POWER SYSTEM DATABASE (MASS AND PERFORMANCE)
  3.) ASSESS ELECTRIC THRUSTER OPTIONS (MASS, PERFORMANCE, PROPELLANT)
  4.) DETERMINE POTENTIAL FOR SP-100 REACTOR SCALE-UP
  5.) INTEGRATE SYSTEMS INTO OVERALL NEP CARGO VEHICLE DESIGN
  6.) DETERMINE SCENARIO OPTIONS AND MISSION PERFORMANCE LEVERAGE

- APPROACH
  - IN-HOUSE/CONTRACTOR
  - POWER AND PROPULSION SYSTEMS ANALYSIS
  - REACTOR OPTIMIZATION AND NEUTRONICS MODELING (SP-100 SCALE-UP)
  - LOW-THRUST TRAJECTORY OPTIMIZATION MODELS

Why E.P.?

An evolutionary human exploration of Mars will require the delivery of heavy cargo for timely and efficient expansion. The high specific impulse possible with electric propulsion can significantly reduce propellant requirements for Mars missions allowing a greater percentage of the vehicle mass to be payload. One might describe this benefit in terms of payload mass fraction---the fraction of the total vehicle mass which can be allotted for payload. The high payload mass fraction possible with NEP makes it particularly favorable for cargo missions.

The low propellant requirements associated with electric propulsion could also make vehicle reuse attractive because of the minimal propellant resupply needed for additional trips. This same characteristic also results in mass savings to LEO for cargo missions, and a reduced dependence on in-situ propellant production. Similarly, variation of vehicle mass with launch date opportunity is minimal allowing for flexible mission scheduling.

Additional studies are necessary to identify the design requirements and logistics required for using NEP power systems at the vehicle's destination. This ability appears to be feasible in reference to the use of the power system at the Martian moons.

WHY E.P.?

- HIGH Isp (4000-8000 seconds) REDUCES PROPELLANT REQUIREMENTS
- HIGH PAYLOAD FRACTION MAKES NEP IDEAL FOR CARGO DELIVERY
  - 50-60% FOR NEP VERSUS 10-15% FOR CHEMICAL
  - POTENTIAL FOR REUSABILITY
- FACTOR OF 2 SAVINGS IN IMEO OVER CHEMICAL CARGO DELIVERY
- POSSIBLE USE OF POWER SOURCE AT DESTINATION
- POWER SYSTEMS ON NEP VEHICLES SYNERGISTIC WITH HIGH POWER SURFACE APPLICATIONS
- USE OF NEP REDUCES DEPENDENCE ON IN-SITU RESOURCE UTILIZATION (PROPELLANT MANUFACTURING)
- NEP VEHICLE PERFORMANCE IS NOT AS SENSITIVE TO LAUNCH DATE AS CHEMICAL

NEP Mission Application

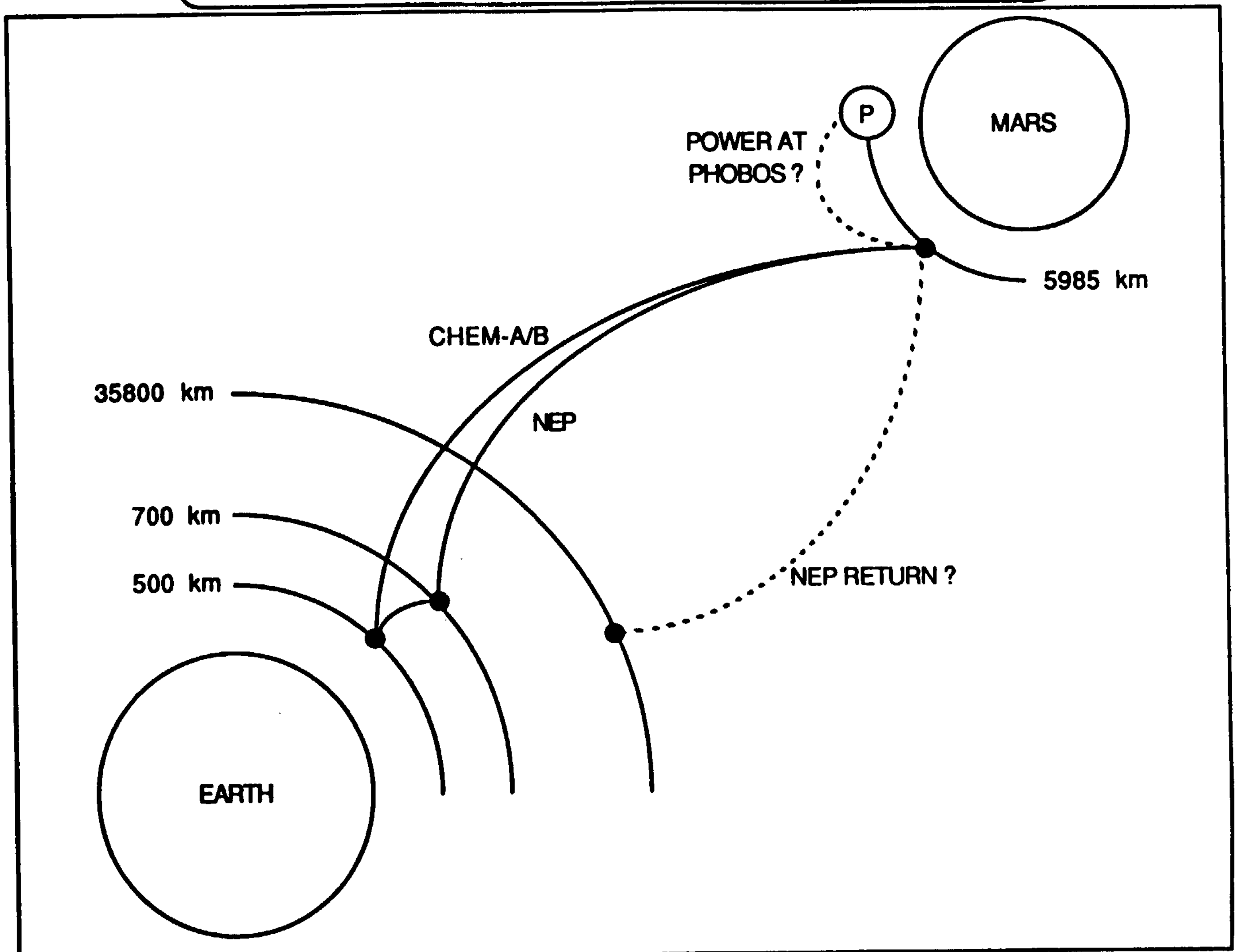
The most attractive characteristic of NEP is its versatility. While typical chemical/aerobrake vehicles would be expended following their mission, a NEP vehicle offers the possibility of subsequent cargo missions, power at one of the Martian moons, or orbit transfers at Mars.

A typical NEP low thrust trajectory includes a planetary spiral escape, heliocentric transfer, and planetary spiral capture. In general, the spiral periods within the planetary gravity wells are the major portion of the overall transit time. Low thrust trip times will usually exceed typical high thrust trip times because of these long spiral periods. However, EP systems require much less propellant than chemical vehicles for these orbital transfers.

This study concentrates on one way and round trip NEP missions from Earth nuclear safe orbit to Phobos orbit with payload masses of hundreds of metric tonnes. Emphasis is placed on the technology options for NEP power and propulsion systems and the trajectory optimization analysis. The reference mission timeframe is 2010 to 2020.

Future work will examine the potential for NEP to perform human transfers to Mars as well as cargo transport in cislunar space.

NEP MISSION APPLICATION



Propulsion System Comparison

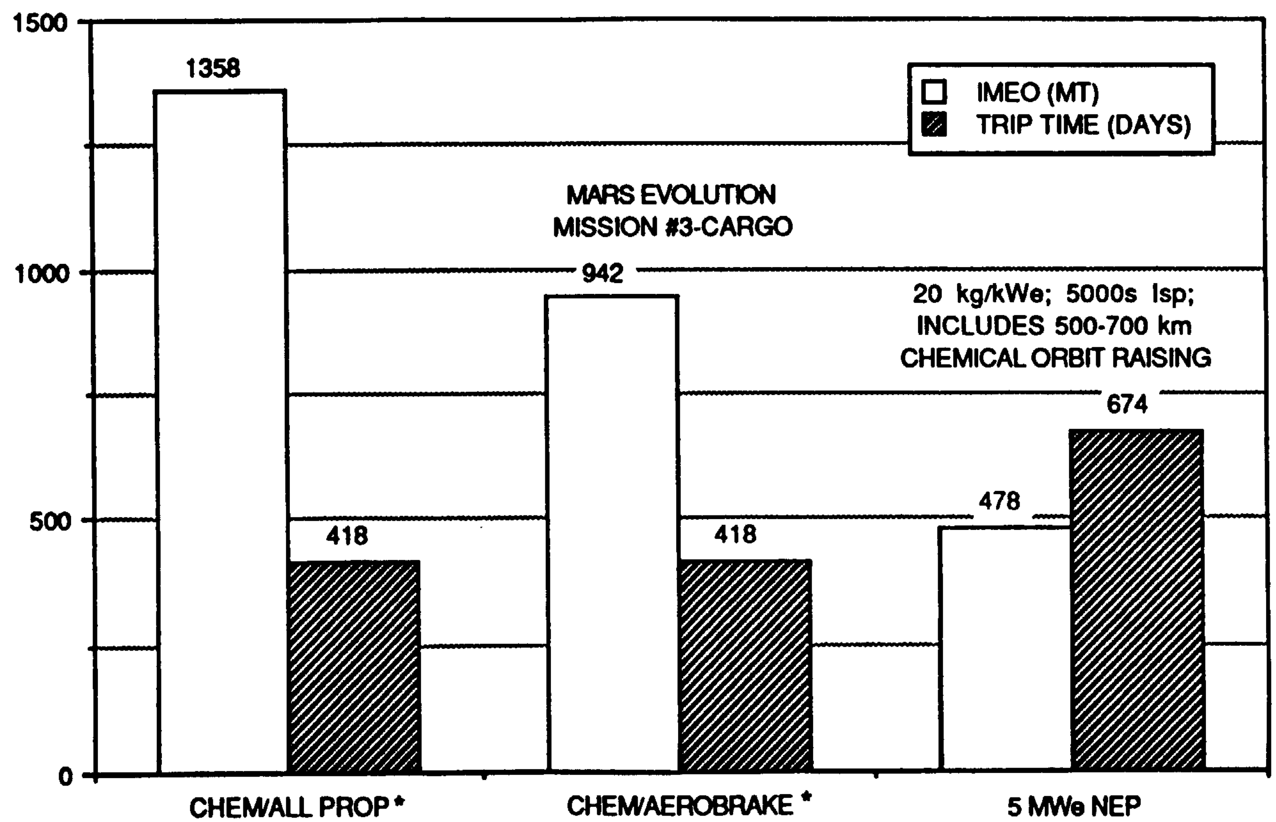
A NEP system was compared to two chemical systems (chemical/all propulsive and chemical/aerobrake) delivering the same payload. The basis for the comparison was mass in low Earth orbit and vehicle transfer time. The transfer occurs between a 500 km Earth orbit and a 6000 km orbit at Mars. The Martian orbit was selected for its equivalence to the altitude of Phobos. The 222 MT payload was consistent with the payload considered for the initial cargo vehicle in OEXP's Mars evolutionary case study.

The chemical systems' performance is based on a projected specific impulse of 470 seconds. The aerobrake mass was assumed to be 15% of the braked vehicle mass. The two chemical propulsion system missions were optimized for minimum total ΔV, thereby minimizing the initial mass in earth orbit (IMEO). The NEP vehicle employs an arbitrary 5 MWe, 20 kg/kWe power source and 5000 second ion thrusters. Included in the mass estimate for the NEP option is a chemical stage to raise the vehicle from the initial 500 km orbit to a 700 km orbit presumed to be nuclear safe. It is at nuclear safe orbit (NSO) where the reactor would be started.

To complete the mission, the NEP vehicle requires one-half the IMEO as compared to the chemical/aerobrake vehicle and 35% that of the all-propulsive chemical vehicle. The low thrust trip is longer than the high thrust chemical vehicle trip by 256 days. However, the delivery of cargo does not necessitate fast trip times. The major concern is that the manifest reach its destination when it is needed.

PROPULSION SYSTEM COMPARISON

NEP versus CHEMICAL: MASS TO ORBIT (IMEO) AND TRIP TIME
222 MT DELIVERED FROM 500 km LEO TO 6000 km PHOBOS



* Isp = 470 s

Propulsion System Concepts

Two forms of electric propulsion were considered for this high power application: argon ion thrusters and hydrogen magnetoplasmadynamic (MPD) thrusters. The NASA OAST Pathfinder and base R&T programs have identified a research program for developing both concepts to MWe power levels. At present, ion thrusters have been developed to flight-ready status at low power levels ($<$10 kWe). MPD thrusters have been operated at higher power levels (100's - 1000's kWe) but require further research to improve performance and lifetime.

The argon ion engine designs are based on state-of-the-art ring cusp thruster technology scaled to higher power levels using theoretical and empirical scaling relationships. The high power engines necessary for this application require advances in the manufacture of large, 5-10 m^2 high voltage accelerating grids at 1 mm separations for use in the demanding thermal and electrical operating environment. In addition, ion thruster performance must be assessed at high power levels.

MPD thruster design is based on the current understanding of self induced field, MWe power level thruster performance obtained from the literature. Performance is projected from current values using empirical and theoretical relationships. Thruster sizing is determined by radiative cooling limits for the thruster anode.

Ion thrusters were selected as the reference system for this application based on their combination of projected high efficiency, long life, and relative maturity. The long trip times for Mars missions and relatively short thruster lifetimes make it necessary to include additional ion thrusters for redundancy over the course of the mission.

PROPULSION SYSTEM CONCEPTS

- PROPULSION SYSTEM CONSISTS OF THRUSTER ELECTRODES, MAGNETS, STRUCTURE, GIMBALS, AND THERMAL REJECTION
- SELF RADIATING ARGON ION THRUSTERS CHOSEN FOR REFERENCE EP CONFIGURATION BECAUSE OF DURABILITY AND HIGH EFFICIENCY
- HYDROGEN MPD THRUSTERS WITH PYROLITIC GRAPHITE RADIATOR ALSO CONSIDERED BECAUSE OF POTENTIAL FOR COMPACTNESS AT HIGH POWER
- THRUSTER PERFORMANCE PARAMETERS:

THRUSTER PERFORMANCE PARAMETERS:	ION	MPD
LIFETIME (HRS)	10,000	5,000
TOTAL VOLTAGE (VOLTS-DC)	2000	175
SPECIFIC IMPULSE (SECONDS)	5000	5000
THRUSTER EFFICIENCY (kWj/kWe)	0.67	0.60
INPUT POWER PER THRUSTER (MWe)	1.25	2.50
THRUSTER AREA (m2)	10	0.6
NUMBER OF THRUSTERS (MARS ONE WAY)	8	8
SPECIFIC MASS PER THRUSTER (kg/kWe)	0.57	0.05
TANKAGE FRACTION	0.1	0.1

Power System Groundrules

Four power conversion options were investigated: closed Brayton, free-piston Stirling, potassium Rankine, and in-core thermionic. The dynamic systems were integrated with an SP-100 derivative reactor, defined as a reactor which uses technology being developed under the DOE, DOD, and NASA SP-100 program but is scaled in thermal power level beyond the 2.5 MWt reference value. The static, thermionic system would utilize a reactor in which the power conversion was integrated directly with the reactor fuel. This would require a new reactor development program.

The reactor shielding is designed for protection of the cargo against radiation damage. A conical shadow shield concept was selected for this application. The dose limits are consistent with those being planned for SP-100. Separation distance and shield half angle are such that it is safe for humans to perform short term cargo change out and vehicle refurbishment following the mission.

Peak cycle temperature and radiator specific mass were varied for the power conversion options as a function of mission timeframe. Power conditioning specific mass was varied as a function of power level.

POWER SYSTEM GROUNDRULES

- DYNAMIC CONVERSION SYSTEMS INTEGRATED WITH SP-100 DERIVATIVE REACTOR (FAST SPECTRUM WITH UN FUEL PINS AND LITHIUM COOLANT)

- THERMIONIC SYSTEM INTEGRATES POWER CONVERSION WITH REACTOR FUEL

- INSTRUMENT RATED SHIELDING
  - 5×10^5 RAD AND 1×10^{13} NVT DOSE LIMITS AT PAYLOAD
  - 100 m SEPARATION DISTANCE; 20 m PAYLOAD DIAMETER
  - 18° HALF ANGLE FOR NEUTRON SCATTERING AND RADIATOR PACKAGING

- POWER CONVERSION OPTIONS INCLUDE CLOSED BRAYTON, FREE-PISTON STIRLING, POTASSIUM RANKINE, AND IN-CORE THERMIONIC
  - PEAK TEMPERATURES RANGE FROM 1100-1500 K FOR DYNAMIC SYSTEMS AND 1800-2200 K FOR THERMIONIC
  - RADIATOR SPECIFIC MASSES RANGE FROM 10-5 kg/m2
  - POWER CONDITIONING MASSES RANGE FROM 4 kg/kWe AT 2 MWe TO 1.6 kg/kWe AT 10 MWe PLUS MASS OF POWER CONDITIONING RADIATOR

- TECHNOLOGY ENHANCEMENTS (TEMPERATURE, RADIATOR SPECIFIC MASS) ARE FUNCTIONS OF MISSION TIMEFRAME - REFERENCE TIMEFRAME IS 2010-2020

MMW Power System Database

The four conversion systems were evaluated at power levels of 2, 5, and 10 MWe, lifetimes of 3 and 10 years, and mission timeframes of near term, mid-term, and far term. The different timeframes incorporate technology enhancements with advancing time. Near term technology consists of an 1100K peak temperature limit and an 8 kg/m^2 radiator specific mass. Mid-term technology corresponds to a 1300K peak temperature limit (1800K for thermionic) and a 6 kg/m^2 radiator. Far term technology includes a 1500K peak temperature limit (2200K for thermionic) and a 5 kg/m^2 radiator. Mid-term technology was assumed to be appropriate for the reference timeframe, 2010-2020.

Included in the mass estimates is a high voltage, direct current power conditioning system suitable for the ion thrusters.

The results indicate that mission timeframe could have a significant effect on power system specific mass. Higher power level and shorter power system lifetime has a lesser effect on specific mass. The lower specific mass associated with shorter power system lifetime is due mainly to a decrease in the amount of reactor fuel and radiator armoring. Power system specific mass also decreases with increasing power due to an economy of scale.

MMW POWER SYSTEM DATABASE

POWER, kWe	LIFE, YEARS	TIME-FRAME	SYSTEM SPECIFIC MASS (kg/kWe)			
			BRAYTON	STIRLING	K-RANKINE	THERMIONIC
2000	3	NT	26.4	25.5	17.8	-
		MT	16.2	20.9	11.3	10.1
		FT	12.4	18.8	9.5	7.4
2000	10	NT	29.3	28.1	18.9	-
		MT	18.2	22.9	12.6	12.5
		FT	14.1	20.5	12.4	8.8
5000	3	NT	24.1	24.2	17.2	-
		MT	14.3	19.6	10.2	8.7
		FT	10.8	17.5	8.3	6.2
5000	10	NT	26.8	26.6	18.1	-
		MT	16.3	21.5	11.6	10.9
		FT	12.5	19.1	11.1	7.4
10000	3	NT	22.6	22.6	-	-
		MT	12.8	18.0	8.9	7.1
		FT	9.4	16.0	6.9	4.7
10000	10	NT	25.2	25.0	-	-
		MT	14.7	19.9	10.2	9.2
		FT	10.9	17.5	9.8	5.8

NT = NEAR TERM = 1100K PEAK TEMPERATURE; 8 kg/m2 RADIATOR
MT = MID TERM = 1300K (DYNAMIC SYSTEMS) OR 1800K (THERMIONIC) PEAK TEMPERATURE; 6 kg/m2 RADIATOR
FT = FAR TERM = 1500K (DYNAMIC SYSTEMS) OR 2200K (THERMIONIC) PEAK TEMPERATURE; 5 kg/m2 RADIATOR

Nuclear Power System Options - 5 MWe, 10 year Life

A reference case was selected to compare the various power system options using the following assumptions: 5 MWe power level, 10 year life, and mid-term technology. The comparison shows the in-core thermionic system to be the lowest mass option followed in order by the potassium Rankine, closed Brayton, and free-piston Stirling. The in-core thermionic system was also shown to have the smallest radiator due to its high heat rejection temperature. Radiator area could have a significant impact in terms of vehicle assembly.

For the dynamic systems, an important consideration may be the number of individual engines necessary to produce the power level. Large numbers of engines may introduce vehicle integration and heat transport problems. Brayton and Rankine systems could meet the 5 MWe power requirement with two engines while the Stirling system would require 10 engines scaled considerably from present power levels. An aspect which must be studied further is the number of engines (and spares) needed to achieve a specified reliability.

The selection of a power system for this application must also take into account relative system maturity. A rating system of 1-10 was established to rank the systems. Higher numbers indicate systems which could most easily be developed for a 5 MWe, 10 year life NEP mission. These numbers should not be confused with the NASA technology readiness levels.

NUCLEAR POWER SYSTEM OPTIONS

5 MWe - 10 YEAR LIFE

SYSTEM	EFFICIENCY (%)	MASS (kg/kWe)[1]	RAD. AREA (m2)	RELATIVE MATURITY[2]
• CLOSED BRAYTON 1300K TURBINE INLET; 2 ENGINES; OPTIMIZED COMPRESSOR INLET TEMP	19.5	16.3	5526	10
• FREE PISTON STIRLING 1300K HEATER HEAD; 10 ENGINES; OPTIMIZED COOLER HEAD TEMP	23.1	21.5	2973	7
• POTASSIUM RANKINE 1300K TURBINE INLET; 2 ENGINES; OPTIMIZED CONDENSER TEMP	18.4	11.6	1625	5
• IN-CORE THERMIONIC 1800K EMITTER; 1050K COLLECTOR	9.7	10.9	1389	3

1 kg/kWe INCLUDE MASS OF 2000 V DC POWER CONDITIONING SYSTEM.

2 SCALE 1-10, HIGHER NUMBERS INDICATE SYSTEMS WHICH COULD MOST EASILY BE DEVELOPED FOR A 5 MWe, 10 YEAR NEP APPLICATION BASED ON DEVELOPMENT TO DATE; NOT TO BE CONFUSED WITH NASA TECHNOLOGY READINESS LEVELS.

Nuclear Power System Options - Maturity Rationale

The ranking of systems by relative maturity shows the closed Brayton to be the most mature. This conclusion was based on the successful testing of Brayton units during the 1970s, its selection for solar dynamic power on Space Station Freedom, and the projected ease with which it can be scaled to higher power levels.

The DOE has also selected Brayton cycle conversion for development in the TECS (Turbine Energy Conversion Systems) program. That program is aimed at the development of a dynamic conversion system for integration with various heat sources including solar, isotope, and nuclear reactor. DOE has invested significant resources in preparing a design for a Brayton cycle dynamic isotope power system (DIPS) at the 6 kWe power level.

The remaining conversion systems have major development issues associated with their suitability for the 5 MWe, 10 year life NEP application.

NUCLEAR POWER SYSTEM OPTIONS

MATURITY RATIONALE

BRAYTON:

1140K, 15 kWe SPACE PROTOTYPE SYSTEM SUCCESSFULLY TESTED IN EARLY 1970'S; 25 kWe SYSTEM BEING DEVELOPED FOR SPACE STATION; EASILY SCALED TO HIGHER POWER LEVELS.

STIRLING:

1050K, 25 kWe SPACE ENGINE BEING DEVELOPED UNDER NASA CSTI FUNDING; ENGINE PROGRAM HAS GOALS OF 1300K AND 25 kWe PER ENGINE. TECHNOLOGY DEVELOPMENT REQUIRED TO ACHIEVE HIGH UNIT POWER LEVELS.

RANKINE:

SUBSTANTIAL COMPONENT TESTING IN LATE 1960'S; PROLONGED TECHNOLOGY HIATUS UNTIL RECENT DOE STUDIES; PHASE I DESIGN STUDIES IN DOE MMW PROGRAM. TECHNOLOGY DEVELOPMENT TO UNDERSTAND MICROGRAVITY TWO-PHASE FLOW CONDITIONS.

THERMIONIC:

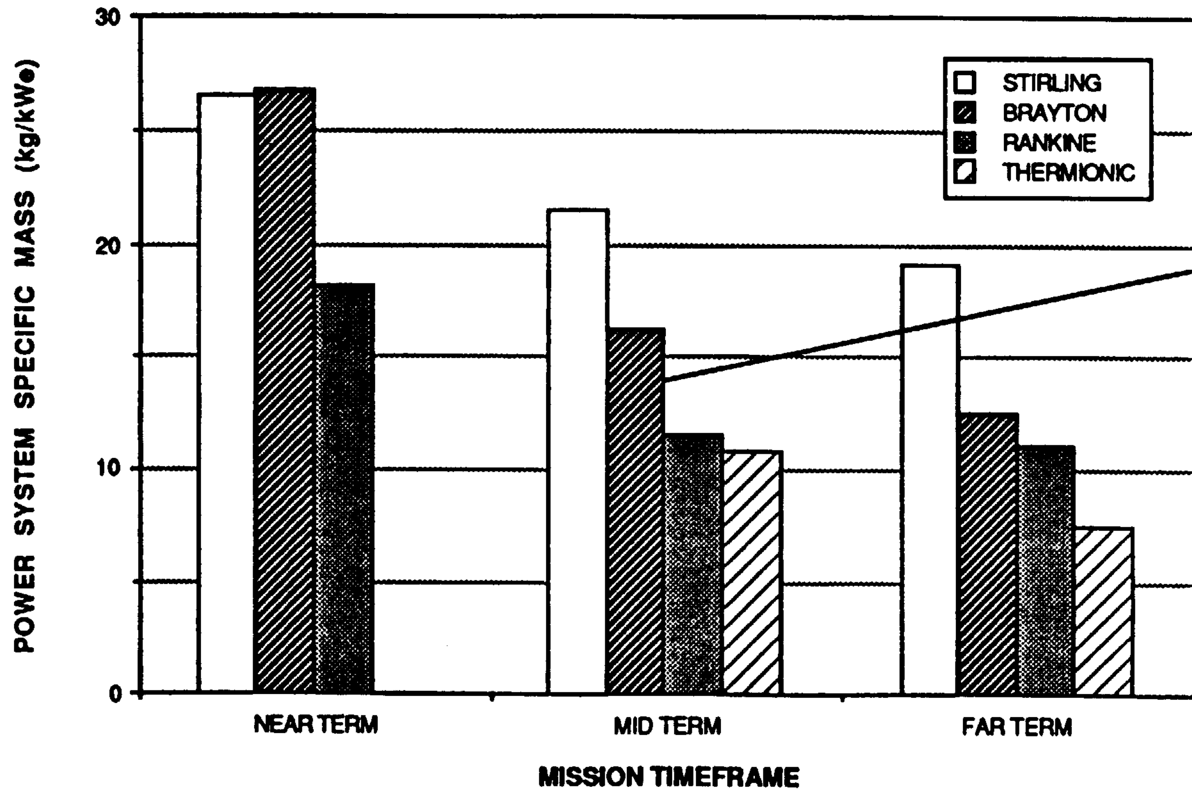
THERMIONIC FUEL ELEMENT (TFE) BEING DEVELOPED IN DOE PROGRAM. TECHNOLOGY DEVELOPMENT REQUIRED TO EXTEND LIFE OF EMITTERS AND INSULATORS.

Power System Sensitivity

The most dramatic effect on power system specific mass is brought on by mission timeframe. This chart shows the impact of mission timeframe on specific mass for the 5 MWe, 10 year lifetime designs. From near term technology to mid-term technology, as much as a 40% mass savings can be attained. The mass savings in going from mid-term to far term technology range from 25% for Brayton to less than 5% for the Rankine system.

A complete mass breakdown for the mid-term Brayton is presented. The main radiator is 36% of the total mass while the reactor and shield is about 25%. The Brayton engines and alternators are only 14% of the total power system mass. Brayton conversion was selected for this application for its combination of high relative maturity, moderate specific mass and ease of power scale-up.

POWER SYSTEM SENSITIVITY
TO MISSION TIMEFRAME



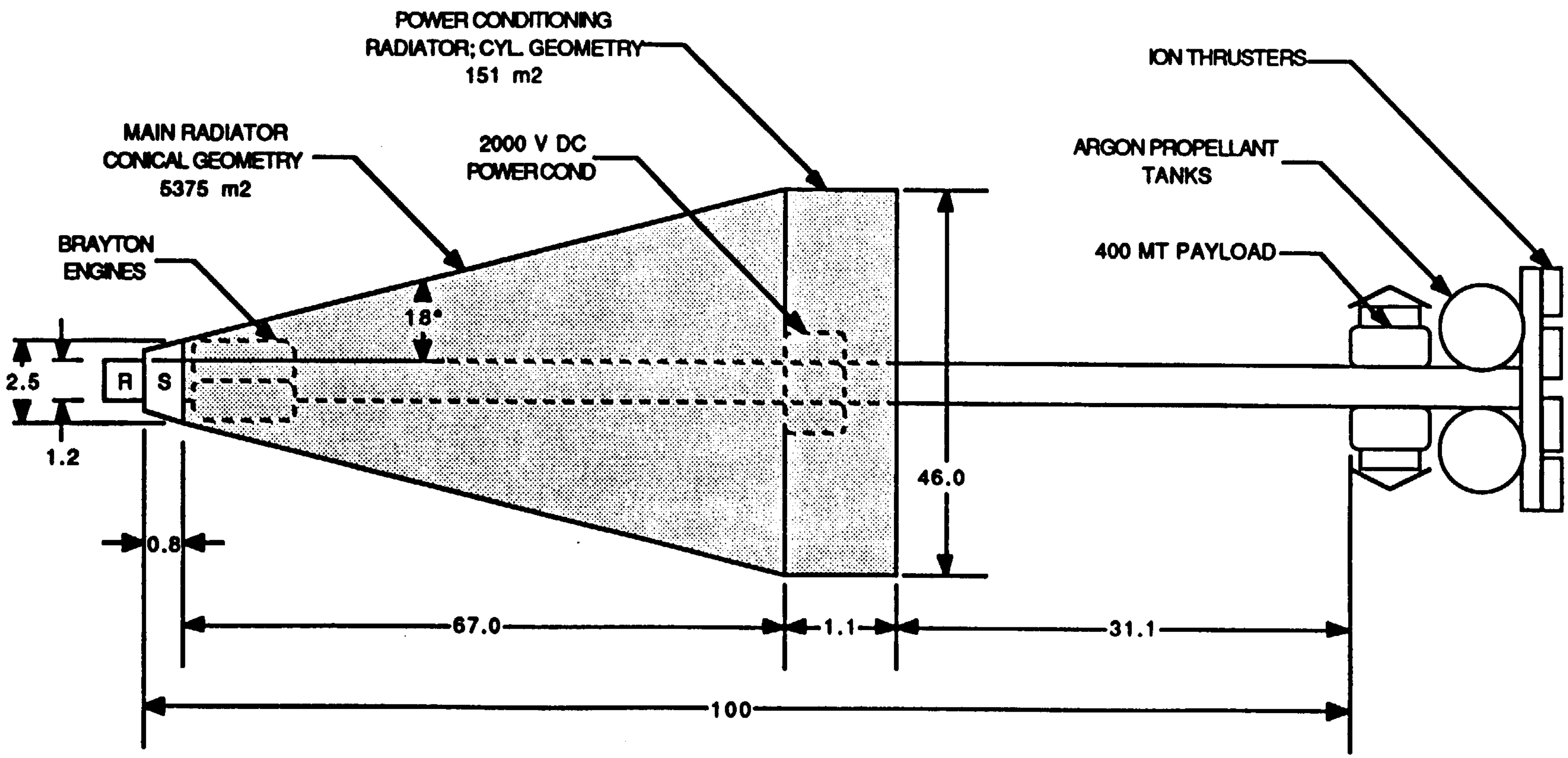
REFERENCE SYSTEM MASS (TONNES)	
REACTOR, SHIELD	20.0
BRAYTON ENGINES	11.7
MAIN RADIATOR	29.7
POWER COND.	16.0
STRUCTURE, MISC.	4.2
TOTAL	81.6

NEP Vehicle Configuration

A 5 MWe vehicle concept is shown which utilizes the reference power and propulsion system designs. The Brayton radiator is the dominant subsystem in terms of both mass and size. However, there is enough space on the 100m boom and within the 18° half angle to easily package the radiator subsystem. The radiator is shaped in a truncated cone geometry and displaces 2/3 of the 100m boom length. This configuration does not necessarily represent an optimum design from the standpoint of boom length versus shield half angle.

The dual Brayton engines and alternators are located directly behind the reactor's shadow shield to minimize heat transport losses. The power conditioning components and waste heat rejection are located at the end of the main radiator to limit radiation effects. The placement of the payload, argon propellant tanks, and ion thrusters at the opposite end of the boom from the reactor eases the risk of radiation exposure during short duration human-tended vehicle refurbishment and cargo change-out.

NEP VEHICLE CONFIGURATION



DIMENSIONS IN METERS
NOT DRAWN TO SCALE

Power System Trade Study Conclusions

The parallel effort by GE to assess SP-100 reactor technology at high thermal power levels revealed that with modifications to the reactor control strategy and fuel burn-up, SP-100 could be scaled to the 50 MWt level. The reference power system option producing 5 MWe requires a reactor thermal power level of about 25 MWt.

Closed Brayton cycle conversion was selected despite its higher specific mass and larger radiator than the in-core thermionic and potassium Rankine systems. Relative maturity and the ease with which it can be scaled in power level were the major reasons. While the radiator is large, it can be packaged on the 100m boom within the established half angle. The other conversion system options have development issues that must be overcome before the may be considered for a NEP mission application in this timeframe.

POWER SYSTEM

TRADE STUDY CONCLUSIONS

- SP-100 REACTOR TECHNOLOGY CAN BE APPLIED TO NEP WITH MINOR MODIFICATIONS (REACTOR CONTROL STRATEGIES)
- CLOSED BRAYTON POWER CONVERSION SELECTED FOR COMBINATION OF LOW MASS AND HIGH MATURITY
    - CBC RADIATOR AREA IS SUBSTANTIAL, BUT CAN BE PACKAGED WITHIN SHADOW SHIELD ANGLE ON 100 m BOOM
- BY INCORPORATING HIGH TEMPERATURE MATERIALS OR ADVANCED RADIATOR CONCEPTS, SYSTEM PERFORMANCE COULD BE ENHANCED
- OTHER SYSTEMS, i.e. POTASSIUM RANKINE AND IN-CORE THERMIONIC OFFER IMPROVED PERFORMANCE BUT REQUIRE SIGNIFICANT DEVELOPMENT
- FOR ALL SYSTEMS CONSIDERED, SPECIFIC MASS CAN BE REDUCED BY SHORTER POWER SYSTEM LIFETIMES OR HIGHER POWER LEVELS

NEP Cargo Vehicle Performance

Typically, a low thrust trajectory will become more mass efficient for longer trip times. The variation of initial mass in earth orbit with trip time is usually examined. The power system specific mass also has an impact on the IMEO. These trades were analyzed for the delivery of a 400 MT payload to Mars.

The trajectory assumed a starting point at a 408 km orbit and a final Mars areosynchronous circular orbit. The reactor power system is providing 5 MWe to the thruster subsystem operating at an efficiency of 67.5% and a specific impulse of 5000 seconds. The 11.5 MT mass of the thruster subsystem did not include the additional thrusters needed for redundancy over the Mars mission.

A minimum trip time exists for each specific mass---the all-propulsive case, where no coasts occur for the duration of the trip. The addition of coast periods to the trajectory decreases IMEO but increases trip time. The all-propulsive limit is shown, as well as curves of constant trip time.

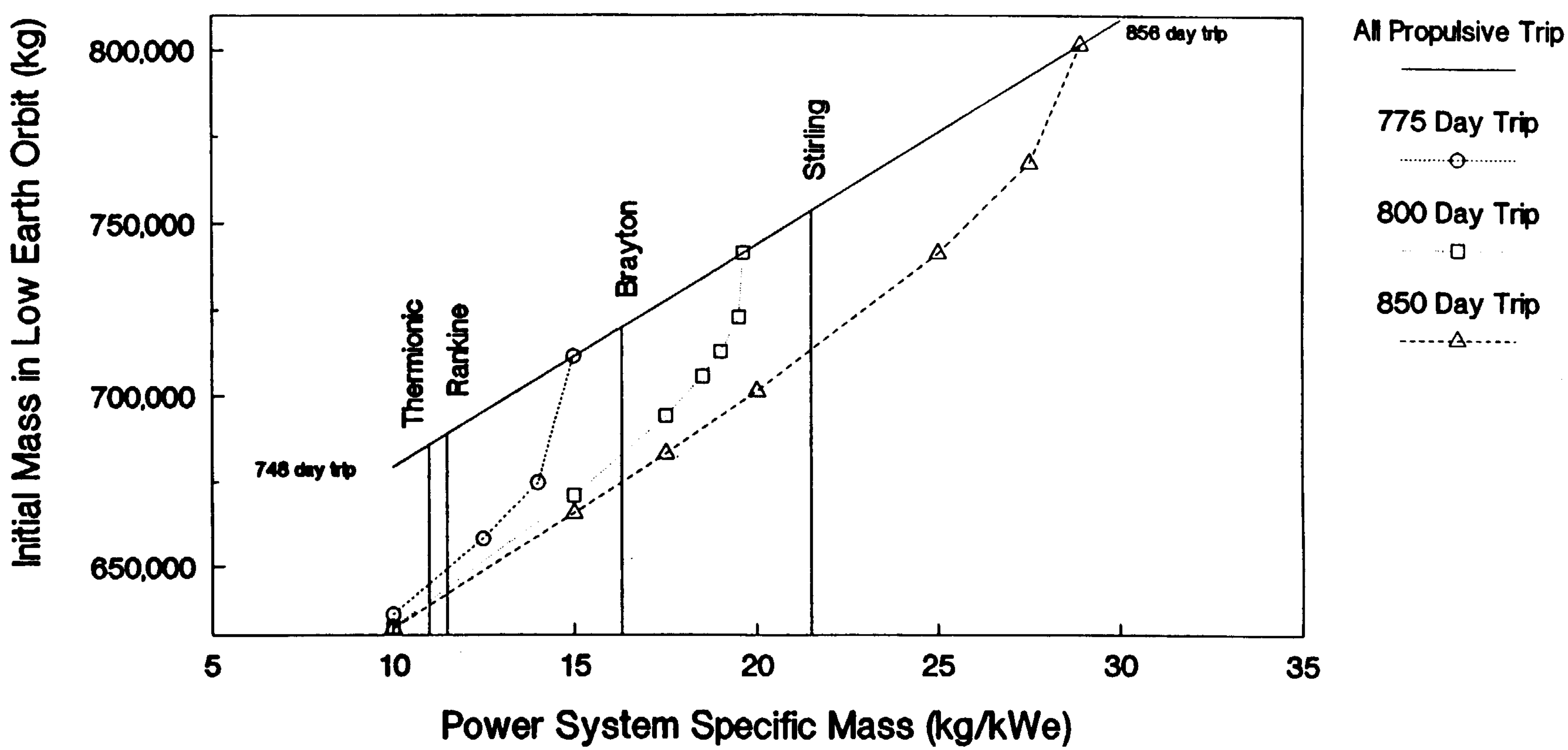
The four power conversion systems are indicated on the chart. These represent the specific masses for the 5 MWe power level, 10 year steady-state lifetime, mid-term technology power system options reference case.

Note that by selecting a lighter conversion system, one not only reduces the limiting trip time but also reduces the mass required in low earth orbit.

NEP Cargo Vehicle Performance

Power Level – 5 MWe

All Low Thrust Trajectory



Propulsion System Specific Mass = 2.29 kg/kWe
Orbits: Earth–408 km, Mars–Areosynchronous Circ.
Isp = 5000 sec

Delivered Payload = 400,000 kg
Earth–Mars One Way Trip
ASAO – May 1989

Mission Scenario Options

A baseline round trip NEP mission was established which utilized the reference 5 MWe, 10 year life power system. That vehicle delivers 400 MT from nuclear safe orbit (NSO) to Phobos orbit and returns 100 MT to Earth geosynchronous orbit. A 2013 mission opportunity was assumed. The actual Earth departure date was optimized for minimum IMEO.

Several scenario options were investigated and compared with the baseline NEP case. These included a power level sensitivity, a one way expendable trip, and an all nuclear cargo delivery option. The power level sensitivity case gives mission results for 2 MWe and 10 MWe vehicles delivering and returning the same payload as the 5 MWe vehicle for a similar mission opportunity. The one way expendable vehicle utilizes a 3 year life power system producing 5 MWe and delivers the same 400 MT to Phobos orbit for the 2013 opportunity. The all nuclear cargo option examines the possibility of delivering the entire Mars evolution manifest as proposed by the OEXP 1989 Scenario Requirements Document (SRD) in 1 and 1/2 round trips beginning in 2007. This cargo delivery scenario differs from the reference scenario proposed in the Mars evolutionary case study which is performed by a combination of four piloted vehicles, one chemical/aerobrake cargo vehicle and one NEP cargo vehicle.

MISSION SCENARIO OPTIONS

- BASELINE ROUND TRIP
  - 5 MWe; 16.3 kg/kWe; 10 YEAR LIFE; MID TERM TECHNOLOGY
  - NSO TO PHOBOS; 400 MT DELIVERED (~2013 OPPORTUNITY)
  - PHOBOS TO GEO; 100 MT RETURNED
- POWER LEVEL SENSITIVITY
  - 2 MWe; 18.2 kg/kWe; 10 YEAR LIFE; MID TERM TECHNOLOGY
  - 10 MWe; 14.7 kg/kWe; 10 YEAR LIFE; MID TERM TECHNOLOGY
  - NSO TO PHOBOS; 400 MT DELIVERED (~2013 OPPORTUNITY)
  - PHOBOS TO GEO; 100 MT RETURNED
- ONE WAY EXPENDABLE TRIP
  - 5 MWe; 14.3 kg/kWe; 3 YEAR LIFE; MID TERM TECHNOLOGY
  - NSO TO PHOBOS; 400 MT DELIVERED (~2013 OPPORTUNITY)
- ALL NUCLEAR CARGO DELIVERY (DELIVERS COMPLETE MARS EVOLUTION MANIFEST FROM SRD IN 1 AND 1/2 ROUND TRIPS)
  - 5 MWe; 18.6 kg/kWe; 10 YEAR LIFE; NEAR TERM RADIATOR
  - NSO TO PHOBOS; 408 MT DELIVERED (~2007 OPPORTUNITY)
  - PHOBOS TO GEO; 100 MT RETURNED
  - GEO TO PHOBOS; 231 MT DELIVERED (~2011 OPPORTUNITY)

Trip Time Comparison

The baseline vehicle completes the mission in 4.3 years which includes a 231 day stay in Phobos orbit. In comparing transit time (total trip time less Phobos stay time), the 2 MWe vehicle requires 2 and 1/2 additional years while the 10 MWe completes the transfer in 11 months less than the baseline. In all cases in which the vehicle returns to Earth orbit (GEO), the Phobos orbit stay time was calculated to insure an optimal round trip.

The one way NEP trip is similar to the mission envisioned for the chemical/aerobrake cargo vehicle in the Mars evolutionary case study. That vehicle is similarly expended following its cargo delivery. The expendable NEP cargo vehicle completes its 400 MT delivery in 2 and 1/4 years. Such a scenario would also eliminate any risk associated with returning a reactor to Earth orbit.

The all nuclear cargo mission delivers the entire Mars evolution mass manifest within 6.7 years. This includes a 5 month stay at Phobos following the initial leg and a one year stay in Earth geosynchronous orbit following the Earth return leg. The one year stay in GEO should be sufficient for reactor cool down and enable human tended cargo change-out and vehicle refurbishment.

TRIP TIME COMPARISON

- DAYS -

EVENT	BASELINE	2 MWe	10 MWe	ONE-WAY	1 1/2 R.T.
NSO DEPART	AUG-12	NOV-10	FEB-13	NOV-12	MAR-06
OUTBOUND	937	1810	659	823	1019
MARS STAY	231	755	340	-	150
RETURN	387	438	324	-	436
EARTH STAY	-	-	-	-	344
OUTBOUND	-	-	-	-	493
TOTAL TRIP	1555	3003	1323	823	2442
THRUSTING	1059	2078	679	696	1394
TRANSIT	***1323***	***2248***	***984***	***823***	***1948***

Mass Comparison

The baseline 5 MWe NEP vehicle requires an initial mass in earth orbit of 835 MT to complete its 400 MT cargo delivery to Mars and 100 MT cargo return to GEO. The propellant is the most massive component of the vehicle requiring over 30% of the total vehicle mass. Included in the initial mass to Earth orbit for all the round trip cases is the propellant required for the return trip. The 10 MWe spacecraft requires 196 MT more total mass than the baseline, while the 2 MWe case could reduce the IMEO by 127 MT.

The one way vehicle has an initial mass of 702 MT of which 57% is the payload to be delivered.

The mission utilizing 1 and 1/2 round trips would take 1756 MT to perform the scenario. Of this mass, 584 MT is for the chemical tug stages to boost the vehicle to NSO and to lift the resupply mass from 500 km to the GEO staging altitude.

MASS COMPARISON

- METRIC TONNES -

COMPONENT	BASELINE	2 MWe	10 MWe	ONE-WAY	1 1/2 R.T.
PROPULSION	46	23	92	23	46/23
POWER	82	36	147	71	93
PROPELLANT	256	206	329	169	262/75
TANKAGE	26	21	33	17	26/8
IMEO	810	686	1001	680	835/430
CHEM STAGE	25	22	30	22	26/558
P.L. DELIVERED	400	400	400	400	408/231
P.L. RETURNED	100	100	100	-	100
TOTAL TO LEO	*835*	*708*	*1031*	*702*	*1756*

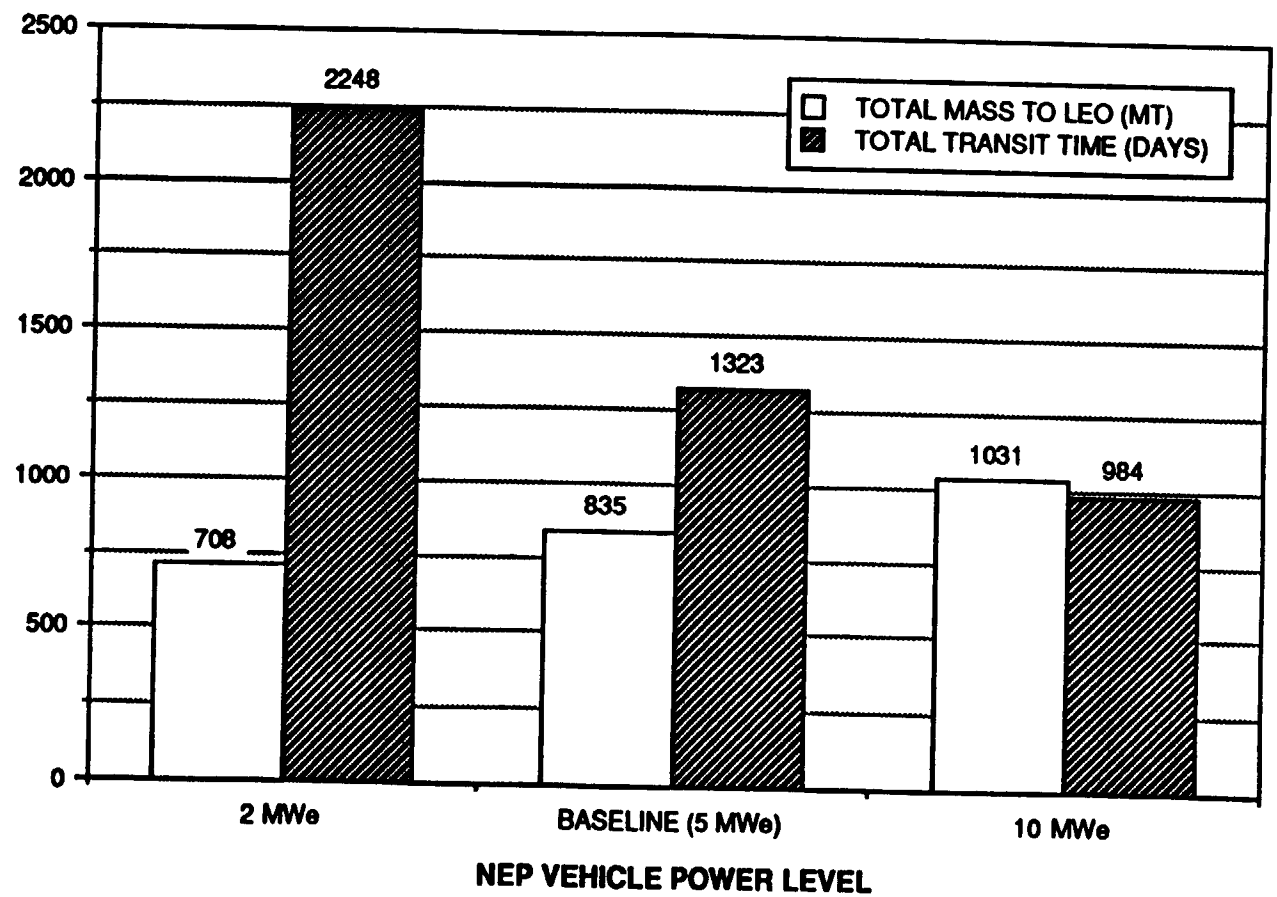
Round Trip Scenario Comparison

The effect of power level on mission performance can be significant. In the range considered for this application, IMEO increases and transit time decreases with increasing power level. While power system specific mass is decreasing with increasing power level, the mass to orbit increases. This effect holds true for large payloads where the power and propulsion system masses are small in relation to the overall vehicle mass.

For the 2 MWe case, trip time increases by 925 days over the baseline, 5 MWe case. By increasing the baseline power to 10 MWe, the transit time decreases by 339 days. These cases show a trend of decreasing time savings with increasing power level. It is presumed that this trend will continue and the transit time savings in going to higher power levels will become negligible. Among the power level options, the 5 MWe reference case appears to offer the best compromise of both mass and transit time for this 400 MT cargo mission.

ROUND TRIP SCENARIO COMPARISON

POWER LEVEL SENSITIVITY: TOTAL MASS TO ORBIT AND TRIP TIME
400 MT CARGO FROM LEO TO PHOBOS



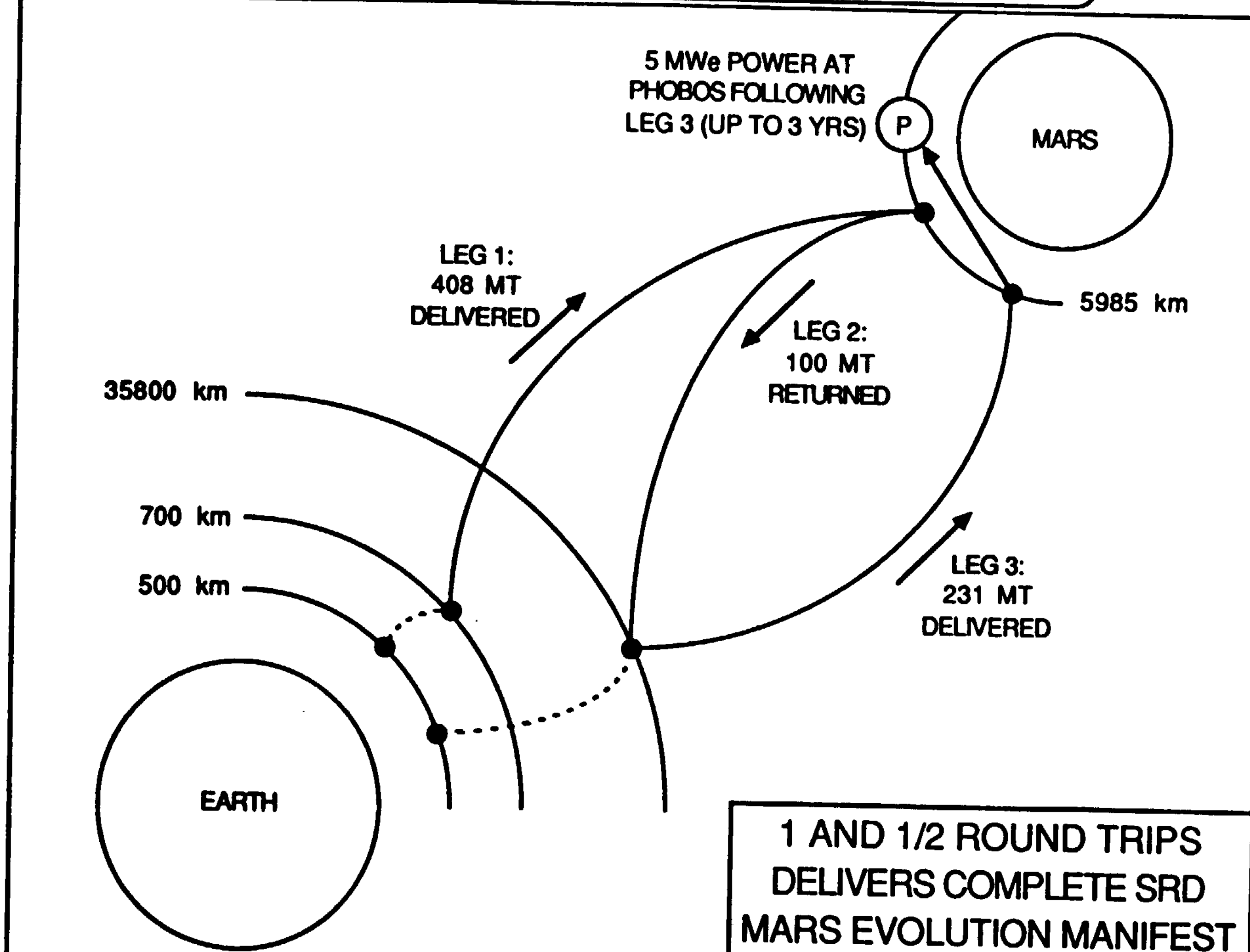
All Nuclear Cargo Delivery

The all nuclear cargo case would deliver the entire Mars evolution mass manifest in 1 and 1/2 round trips using the 5 MWe vehicle. In this case, 408 MT is delivered to Phobos by 2008 to establish the Mars gateway and initial outpost elements and initiate the human tended phase. One hundred metric tonnes of payload is returned from Phobos gateway to GEO and the original vehicle is refurbished with new propellant, tanks, electric thrusters, and payload. The original thrusters and tanks are expended. The final leg of the mission delivers 231 MT of cargo back to Phobos by 2012 to establish the Mars operational phase. Included in the delivered payloads is a sufficient number of Mars cargo landers to transport the material to the surface.

This entire mission would take 6.7 years to complete and would require 1756 MT to be delivered to LEO. Of the total mass to LEO, 584 MT or 33% is for chemical stages to boost the initial vehicle to NSO and to refurbish the vehicle after it has returned to GEO. Further studies are required to determine if GEO is the optimum staging altitude.

This 1 and 1/2 round trip scenario offers significant advantages over the current cargo delivery scenario which is performed by a combination of four piloted vehicles, one chemical/aerobrake cargo vehicle, and one NEP cargo vehicle. First, a single, common vehicle can be used for transporting the entire Mars evolution cargo manifest. Vehicle reusability provides an efficient use of invested mass to orbit, and eliminates the need to develop multiple technologies for cargo transportation systems. The elimination of cargo on the piloted vehicles could enhance their performance. Second, the all NEP cargo delivery option could significantly reduce the amount of mass required in LEO even with the additional chemical OTV stages for orbit raising and vehicle resupply. Further, the cargo can be delivered in a shorter time period than the current scenario which is completed over ten years. This permits a more robust build-up of the Mars system infrastructure. The versatility of NEP could also allow the use of the vehicle as an OTV about Mars or a power source at Phobos for up to three years following the cargo mission.

ALL NUCLEAR CARGO DELIVERY



1 AND 1/2 ROUND TRIPS DELIVERS COMPLETE SRD MARS EVOLUTION MANIFEST

Conclusions

The power and propulsion system selection is the most critical aspect of devising a nuclear electric vehicle concept. The systems that were selected for this study are based on various performance parameters including low mass and technical readiness. A nuclear power system concept employing SP-100 reactor technology and Brayton cycle conversion appears to be attractive for the long-life, steady-state, multi-megawatt power requirements associated with electric propulsion cargo missions. Ion thrusters, scaled in power level but based on state-of-the-art ring cusp designs, were selected for the propulsion system. Additional thrusters would be required to meet lifetime requirements for Mars missions. The systems were combined into an overall vehicle and detailed mission analyses were performed.

The most telling figure of merit in assessing the system options is mission performance. In comparison to other possible transportation system options for Mars cargo delivery, NEP was found to offer 50% IMEO savings at the expense of somewhat longer trip times. The longer trip times would be of minimal consequence to the cargo, provided that the vehicle reach its destination as the cargo is needed.

The baseline vehicle with a 5 MWe, 10 year life power system can deliver 400 MT to Phobos and return 100 MT to GEO with an initial mass in LEO of 835 MT. Lower or higher power level does not appear to offer significant mission leverage. A 5 MWe, 3 year life power system was used in assessing a one-way expendable mission delivering the same 400 MT payload with a 57% payload mass fraction in LEO. Perhaps the most attractive mission scenario is one in which the entire Mars cargo manifest is delivered in 1 and 1/2 round trips beginning in 2007. This scenario could ease transportation requirements and provide an efficient use of invested mass to orbit.

CONCLUSIONS

- NEP OFFERS SIGNIFICANT MASS SAVINGS OVER CHEMICAL SYSTEMS FOR MARS CARGO DELIVERY
  - HIGHER Isp AND PAYLOAD FRACTION ALLOWS FOR LARGER PAYLOADS
  - THE SOMEWHAT LONG TRIP TIMES SHOULD NOT IMPACT CARGO
- BASELINE VEHICLE DELIVERS 400 MT TO PHOBOS, RETURNS 100 MT TO GEO WITH 835 MT MASS TO LEO
- A MISSION SCENARIO WHICH USES NEP FROM THE OUTSET COULD SIGNIFICANTLY EASE TRANSPORTATION REQUIREMENTS
  - 1 AND 1/2 ROUND TRIPS TO DELIVER COMPLETE MARS MANIFEST
- A ONE WAY EXPENDABLE NEP SCENARIO PROVIDES A SIMPLIFIED AND EFFECTIVE MEANS OF CARGO TRANSFER
  - 702 MT TO LEO DELIVERS 400 MT TO PHOBOS IN JUST OVER 2 YEARS
- LOWER OR HIGHER POWER LEVEL DOES NOT APPEAR TO SHOW SIGNIFICANT MISSION LEVERAGE